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HANDLING QUALITIES REQUIREMENTS AND CONTROL DESIGN FOR HIGH-SPEED ROTORCRAFT

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**Technology Development Directorate
Combat Capabilities Development Command
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14. ABSTRACT The U.S. Department of Defense has established the Future Vertical Lift initiative to develop a family of next-generation vertical lift aircraft that will fly farther, faster, and more efficiently than the current fleet of rotorcraft. To accomplish these goals, advanced rotorcraft configurations beyond the single main rotor/tail rotor design must be considered. Two advanced configurations currently being flight tested in a technology demonstrator program are a lift offset coaxial rotorcraft with a pusher propeller and a tiltrotor. The U.S. Army Combat Capabilities Development Command Aviation & Missile Center Technology Development Directorate has developed generic, high-fidelity flight-dynamics models of these two configurations to provide the government with independent control-system design, handling-qualities analysis, and simulation research capabilities for these types of aircraft in support of the Future Vertical Lift initiative.					
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ABSTRACT (CONCLUDED)

In an effort to begin defining high-speed handling qualities requirements for these types of advanced rotorcraft configurations, full flight envelope inner-loop explicit model following control systems were designed for both configurations using a multi-objective optimization approach to meet a comprehensive set of stability, handling qualities, and performance requirements. The requirements were chosen from both helicopter and fixed-wing criteria. The control laws for both aircraft were evaluated in a piloted simulation experiment at the NASA Ames Vertical Motion Simulator using a series of high-speed handling qualities demonstration maneuvers. The results from the simulation experiment show overall assigned Level 1 handling qualities for both aircraft which correlates well with the predicted Level 1 handling qualities they were designed to, thus supporting the choice of handling qualities requirements selected during the design process.

In addition, outer-loop control laws were also developed for both aircraft in an effort to improve handling qualities and reduce pilot workload. A dynamic inversion control law architecture was used for the outer loops, and again a comprehensive set of stability, handling qualities, and performance requirements was used to optimize the parameters of the control laws. The outer-loop control laws for both aircraft were evaluated in a piloted simulation experiment at the Pennsylvania State University Flight Simulator facility using a subset of the high-speed handling qualities demonstration maneuvers used in the first evaluation. Overall, the outer-loop control laws provided Level 1 handling qualities for the Break Turn and High-Speed Acceleration/Deceleration maneuvers, with improvements seen over the inner-loop control laws. In addition, the outer-loop control laws improved performance and reduced workload during a formation flying task that was developed for this project. However, for pitch attitude capture and tracking tasks, the outer-loop control laws degraded handling qualities as compared to the inner-loop control laws. This was due to the pitch attitude dropback behavior of the outer-loops, even though they were tuned to meet the dropback specification. The results suggest that the amount of allowable dropback should be decreased for high-speed rotorcraft and the outer-loop functionality might be provided as a selectable mode.

The set of handling qualities criteria used to optimize the inner- and outer-loop control systems presented here are proposed as a set of requirements to tune high-speed rotorcraft control systems to achieve Level 1 handling qualities.

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List of Symbols

Variables

β_0	Collective flapping angle [deg]
β_{1c}	Longitudinal flapping angle [deg]
β_{1s}	Lateral flapping angle [deg]
$\Delta\theta'_{1c}$	Differential phased lateral cyclic [deg]
$\Delta\theta'_{1s}$	Differential phased longitudinal cyclic [deg]
$\Delta\theta_0$	Differential collective [deg]
δ_a	Aileron deflection [deg]
δ_e	Elevator deflection [deg]
δ_{nac}	Nacelle angle [deg] (0 deg = airplane mode, 90 deg = helicopter mode)
δ_r	Rudder deflection [deg]
Δ_{SP}	Swashplate control phasing angle [deg]
DB_θ	Pitch attitude dropback [deg]
LOS	Lift Offset [ft or %]
NR	Rotor speed [%]
ν_β	Nondimensional flap frequency [rad/sec]
ν_ζ	Nondimensional lag frequency [rad/sec]
Ω	Rotor rotation speed [rad/sec]
ω	Natural frequency [rad/sec]

ω_{BW}	Piloted bandwidth frequency [rad/sec]
ω_{co}	Pilot cutoff frequency [rad/sec]
ω_c	Broken-loop 0 dB crossover frequency [rad/sec]
A, B, C, D	State-space representation (standard form)
P	Covariance matrix
u	Input vector
x	State vector
y	Output vector
ψ	Rotor azimuth angle [deg]
ρ	Air density [slugs/ft ³]
σ	Standard deviation
τ	Time constant or time delay [sec]
θ	Pitch attitude [deg]
θ'_{1c}	Symmetric (or single rotor) phased lateral cyclic [deg]
θ'_{1s}	Symmetric (or single rotor) phased longitudinal cyclic [deg]
θ_0	Symmetric collective [deg]
$\theta_{0\text{PP}}, \delta_{\text{prop}}$	Pusher propeller collective [deg]
$\theta_{1\text{CPP}}$	Pusher propeller monocyclic [deg]
φ	Frequency response phase angle [deg]
ζ	Damping ratio [-]
h	Altitude [ft]
J_{LOES}	Lower-order equivalent system mismatch cost [-]
J_{MF}	Model following cost [-]
K	Control system gain
L_p	Example of dimensional stability derivative, $L_p \equiv \partial L / \partial p$

L_{δ_a}	Example of dimensional control derivative, $L_{\delta_a} \equiv \partial L / \partial \delta_a$
P	Power [hp or ft-lb/sec]
R	Rotor radius [ft]
s	Laplace Variable
T_γ	Flight path lag [sec]
T_{θ_2}	Flight path-attitude lag [sec]
V	Airspeed [ft/sec or kts]
V_H	Maximum level flight speed at maximum continuous power [kts]
W	Weight [lb]

Subscripts

aux	Auxiliary input (pusher propeller for Coaxial-Pusher, nacelle angle for tiltrotor)
CAS	Calibrated airspeed
cm	Commanded response
col	Collective
dir	Directional
dr	Dutch roll
HO	Higher-order
lat	Lateral
lon	Longitudinal
L	Lower rotor
OL	Outer loop
ped	Pedal
ph	Phugoid
RB	Rigid-body

r	Roll mode
sp	Short period
ss	Steady-state
s	Pilot stick input or spiral mode
TAS	True airspeed
TC	Turn compensation
U	Upper rotor

Acronyms

ABC	Advancing Blade Concept
BACH	Bank Angle Capture and Hold
BWLR	Bedford Workload Rating
CAP	Control Anticipation Parameter
CG	Center of Gravity
DI	Dynamic Inversion
DI	Dynamics Inversion
DRB	Disturbance Rejection Bandwidth
DRP	Disturbance Rejection Peak
EMF	Explicit Model Following
FFS	Force Feel System
FVL	Future Vertical Lift
FVL	Stability and Control Augmentation System
HQR	Cooper-Harper Handling Qualities Rating
ID	Identification
KCAS	Knots Calibrated Airspeed
KTAS	Knots True Airspeed

L1, L2, L3	Level 1, 2, 3
LOES	Lower-Order Equivalent System
MTE	Mission Task Element
NRTC	The National Rotorcraft Technology Center
OAT	Outside Air Temperature
OLOP	Open Loop Onset Point
PACH	Pitch Attitude Capture and Hold
PIO	Pilot Induced Oscillation
PVS	Pilot-Vehicle System
qLPV	quasi-Linear-Parameter-Varying
RCAH	Rate-Command/Attitude-Hold
RMS	Root Mean Square
SL	Sea Level
SOS	Sum-of-Sines
TCL	Thrust Control Lever
VMS	NASA Ames Vertical Motion Simulator

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Chapter 1 | Introduction

With the development of advanced high-speed rotorcraft, through the U.S. Department of Defense (DoD) Future Vertical Lift (FVL) initiative and Joint Multi-Role (JMR) Technology Demonstrator (TD) flight test program, new high-speed handling qualities requirements are needed to ensure safe and low-workload piloting in the transition and high-speed regimes. This research aims to investigate the handling qualities requirements for advanced high-speed rotorcraft in transition and high-speed flight. To that end, full flight envelope flight control systems were developed for a generic lift offset coaxial helicopter with pusher propeller (herein referred to as coaxial-pusher) and a generic tiltrotor. Piloted simulation handling qualities evaluations were done with both generic aircraft models at the NASA Ames Vertical Motion Simulator as well as the Pennsylvania State University Flight Simulator facility using high-speed handling qualities demonstration maneuvers. This chapter will cover background and motivation for this work, a review of the relevant literature, and the objectives and contributions of this research.

1.1 Background and Motivation

Since the advent of vertical lift, there has been an operational need to combine the ability to take off and land vertically with the ability to cruise at sustained high speed. In fact, an effort to develop the next generation of vertical lift aircraft that will fly farther, faster, and more efficiently than the current fleet of rotorcraft, is currently the U.S. Army's third modernization priority [1]. In 1986, nearly 50 years after the first practical helicopter took flight, a Westland Lynx set the current level-flight speed record for conventional single main rotor helicopters, reaching 216 kts. The maximum achievable speed for a conventional helicopter is limited not just by its installed engine power, but also by such factors as the advancing blade tip speed and retreating blade dynamic stall of the rotor, vibration, stability, and maximum transmission torque [2]. Therefore, in order to go faster and break a speed record that has stood for over 30 years, advanced rotorcraft configurations beyond the single main rotor and tail rotor design must be considered.

In the context of this work, “advanced rotorcraft configurations” are defined broadly as those configurations employing lift offset rotors (where the advancing blade generates more lift than the retreating blade), compound lift (e.g., through use of a wing), compound thrust (e.g., through use of propellers or jet engines), and/or tilting wings or nacelles to achieve speeds in excess of 200 kts at mission gross weights.

1.1.1 Past

The list of advanced rotorcraft configuration technology demonstrators developed is long and includes designs with compound lift (e.g., wing), compound thrust (e.g., pusher propeller or jet engines), lift offset coaxial rotors (also known as Advancing Blade Concept), tilting proprotors, and tilting wings. Examples of advanced rotorcraft configuration that have been flown in the past include the Piasecki 16H-1A Pathfinder II (shown in Fig. 1.1), which has a single articulated main rotor, a low-mounted wing, and a ring-tail pusher propeller that provides anti-torque and forward propulsion. The Pathfinder II made its initial flight in 1965 and reached a maximum speed of about 195 kts during flight testing.



Figure 1.1. Piasecki 16H-1A Pathfinder II, early example of compound rotorcraft.

At around the same time of the Pathfinder II flight test program, the U.S. Army initiated the Advanced Aerial Fire Support System (AAFSS) program, to acquire an armed helicopter capable of speeds of over 200 kts. The program led to the development of the Lockheed AH-56A Cheyenne (shown in Fig. 1.2) which featured a hingeless, variable RPM main rotor, a low-mounted wing, a conventional tail rotor, and a rear-mounted pusher propeller. The aircraft had a maximum flight speed of 212

kts, but suffered from several deficiencies affecting mission accomplishment [3]. Among these deficiencies were several handling qualities related issues including: loss of control within the flight envelope resulting from blade moment stall (resulting in assigned Handling Qualities Ratings, HQRs, 10); inability to effectively perform slow-speed ($V < 120$ kts), low-level mission tasks under adverse weather conditions due to the combined effect of the lateral-directional control system and static lateral-directional stability characteristics (HQRs 7); high pilot workload during transition from hover to forward flight (HQRs 6); and excessive pilot effort required to control aircraft attitude and maintain altitude in a rapid reversal of lateral acceleration (HQRs 6) [3]. The Cheyenne program was canceled by the Army in August 1972.



Figure 1.2. Lockheed AH-56A Cheyenne, early attempt at a high-speed attack helicopter that suffered from handling qualities related issues.

In the early 1970s, the Army awarded Sikorsky Aircraft a contract to develop an experimental coaxial compound helicopter to demonstrate the Advancing Blade Concept (ABC), or lift offset. In a lift offset configuration, the advancing blade generates more lift than the retreating blade, and the roll moment on the rotor is nonzero. Operating with a lift offset is useful for avoiding the drag penalty associated with retreating blade stall and thus increases the potential speed of the aircraft, but is only practical in compound or coaxial configurations in order to balance the entire aircraft rolling moment. Sikorsky developed the S-69/XH-59A (shown in Fig. 1.3), which included two rigid counter-rotating rotors spaced 2.5 ft apart and a pair of turbojets to provide compound thrust. The XH-59A first flew in July 1973.

The cockpit had conventional helicopter controls and the aircraft had a dual channel stability augmentation system (SAS) [4]. The two rotors could be actuated in symmetric and differential collective and cyclic pitch. Symmetric collective, lateral cyclic, longitudinal cyclic, and differential collective were used for control, while



Figure 1.3. Sikorsky Aircraft S-69/XH-59A, successfully demonstrated the Advancing Blade Concept (ABC).

differential lateral and longitudinal cyclic could be used for trim through electric beeper switches to investigate rotor lift sharing and rotor moment distributions [4].

As a result of the rigid rotors of the XH-59A, a much larger swashplate phase angle Δ_{sp} (the angular difference around the rotor azimuth between the point at which a control input to a rotor blade is made and the point at which maximum displacement of the blade in response to that control input occurs) was required than for an articulated rotor [4] ($\Delta_{sp} \approx 50 - 60$ deg for the XH-59A as compared to $\Delta_{sp} \approx 8 - 12$ deg for articulated rotors).

The first XH-59A was damaged in an incident in August 1973 caused by insufficient longitudinal control at low speed (around 30 kts) [5]. Follow-on analysis led to control system changes consisting of nearly doubling the longitudinal and lateral control range and sensitivities and providing a flight adjustable swashplate phase angle Δ_{sp} with a range of 0–70 deg. A real-time tip clearance monitoring system was also developed to display minimum rotor tip separation to the pilots in real-time, with 10 inches established as the minimum allowable tip clearance for the XH-59A.

The XH-59A eventually attained a speed of 238 kts and demonstrated good handling qualities [4]. The high control power available resulted in a very maneuverable aircraft without a tendency to over control [4]. The test program ended in 1981, and it would be another 26 years before an ABC aircraft flew again.

At the same time as the development of compound rotorcraft like those described above, work was ongoing to develop tiltrotor aircraft. The Bell XV-3 (shown in Fig. 1.4) was developed in the early 1950s and first flew in August 1955. During a flight test in October 1965, a severe rotor instability occurred as the rotor shaft was moved to 17 degrees forward from vertical which resulted in extremely high cockpit

vibrations and caused the pilot to black out and the aircraft to crash (seriously injuring the pilot) [6]. To study and solve the rotor/pylon/wing dynamic instability problem, Bell and the Government conducted testing in the then NACA Ames Aeronautical Laboratory 40-by-80-foot wind tunnel. The testing led to replacement of the original three-bladed, 25 ft diameter articulated rotor with a 23 ft diameter two-bladed stiff-inplane rotor, the addition of external struts to stiffen the wing, and a significant increase in the stiffness of the rotor controls [6]. Overall, the XV-3 suffered from handling qualities issues (including lateral instabilities during hover in ground effect, low control response, and poor control harmony) and high pilot workload issues (due to a complex gear shifting process required to reduce rotor RPM after converting to the airplane mode) [6]. But the data and experience from the XV-3 program were key elements used to successfully develop the Bell XV-15.



Figure 1.4. Bell XV-3, successfully demonstrated the tiltrotor concept.

The Bell XV-15 (shown in Fig. 1.5) first flew in May 1977. Unlike the XV-3, which had its engines in the fuselage with drive shafts transferring power out to tilting proprotors at the wingtips, the XV-15 had its engines mounted in rotating wingtip nacelles, directly coupled to the rotors. The flight control system for the XV-15 consisted of the stability and control augmentation system (SCAS) and force feel system (FFS), which combined both rotor and aerosurface controls to give minimum pilot effort in all modes of flight and provide conventional control characteristics [7]. The XV-15 SCAS was designed to provide Level 1 handling qualities in airplane mode at altitude with SCAS off and FFS on, and in hover with SCAS and FFS on. Military and FAA specifications (MIL-H-8501A, MIL-F-83300, MIL-F-8785B, AGARD-577, and FAR-25) were used as guidelines in establishing handling qualities criteria since no specification existed for tiltrotor aircraft [7]. The SCAS gains were scheduled with nacelle angle to provide desirable characteristics throughout the flight



Figure 1.5. Bell XV-15, first experimental tiltrotor to successfully demonstrate high-speed performance.

envelope. Overall, the XV-15 exhibited excellent handling qualities and its conversion procedure was made easier by use of a nacelle tilt angle versus airspeed schedule that the pilots followed [8]. However, the gust response in airplane mode was described as unusual—instead of the vertical acceleration felt in most airplanes as a result of gusts and turbulence, the XV-15 had more longitudinal response caused by transient changes in blade angle of attack [8].

1.1.2 Present

In 1981, the DoD began the Joint-Service Vertical Take-Off/Landing Experimental (JVX) aircraft program in an effort to address the need for vertical lift, speed, and range. Partnered with Boeing Vertol, Bell submitted a proposal for an enlarged version of the XV-15, which led to the development of the Bell Boeing V-22 Osprey (shown in Fig. 1.6), the first (and to date only) operational tiltrotor aircraft. The development of the V-22, spanning nearly 20 years from first flight to fielding, is indicative of the challenges associated with advanced vertical lift configurations. In fact, two of the several V-22 crashes that occurred during development were related to the flight control system. The fifth prototype crashed on its first flight (June 11, 1991) because of incorrect wiring in the flight control system [9] and a subsequent crash in December 2000 occurred because of a hydraulic leak and an undiscovered error in the aircraft's control software [10]. The V-22 also experienced several Pilot Augmented Oscillations (PAOs) during its development resulting from biodynamic interference [11] and a roll Pilot Induced Oscillation (PIO) in a landing attempt on a ship deck during sea trials 1999 [12].

Another operational aircraft capable of hover and high-speed flight is the F-35 Joint Strike Fighter (JSF) B model, the short takeoff and vertical landing (STOVL) variant.



Figure 1.6. Bell Boeing V-22 Osprey, first production tiltrotor aircraft.

The Lockheed Martin X-35B concept demonstrator aircraft, from which the F-35B was developed, had to demonstrate good flying qualities from hover through supersonic flight [13]. Flying qualities criteria were taken from MIL-F-83300 for hover while in forward flight MIL-STD-1797A requirements were used. However, an integrated flight propulsion controls working group had to be established, consisting of representatives from both government and Lockheed Martin, to establish a consolidated set of flying qualities design guidelines and evaluation tasks covering hover, transition, and forward flight. The work in Ref. 13 demonstrates the importance piloted simulation played in the development of the flight control system for the X-35B.

More recently, there has been renewed interest in advanced rotorcraft configuration to achieve high speed. In 2009, the DoD established the Future Vertical Lift (FVL) initiative to develop a family of next-generation vertical lift aircraft that will fly farther, faster, and more efficiently than the current fleet of rotorcraft. This was followed by internal research and development (R&D) efforts by the manufactures to develop advanced configurations that could meet the new requirements. In 2010, the Sikorsky X2 Technology Demonstrator experimental helicopter (shown in Fig. 1.7) with two rigid, coaxial lift offset rotors (i.e., ABC) and a pusher propeller, achieved a speed of 250 kts, setting a new record for rotorcraft.

Then in June 2013, the Eurocopter (now Airbus Helicopters) X³ experimental compound helicopter (Fig. 1.8) set a new record, when it reached 255 kts in level flight. The X³ is a single main rotor helicopter with compound thrust provided by two propellers. To avoid drag associated with advancing blade tip Mach number and retreating blade stall, the rotor is slowed down in forward flight and offloaded by the small wing.

Recognizing the challenges inherent in advanced rotorcraft configurations, the DoD established the Joint Multi-Roll (JMR) Technology Demonstrator (TD) program to mitigate the risk associated with the development of FVL [14, 15]. Five companies were selected in 2014 to participate in the JMR-TD effort (Bell Helicopter, Boeing, Sikorsky, AVX, and Karem), with two air vehicles being developed—the Bell V-280



Figure 1.7. Sikorsky Aircraft X2 Technology Demonstrator, continued development of Advancing Blade Concept (ABC).

Valor tiltrotor (Fig. 1.9) and the Sikorsky Boeing SB>1 Defiant (Fig. 1.10). Two of the key performance parameters that the JMR-TD aircraft are expected to demonstrate are speeds in excess of 230 kts and Level 1 handling qualities as specified in ADS-33E-PRF [16] and MIL-STD-1797B [17], and both JMR-TD aircraft are currently undergoing flight tests to demonstrate their speed capabilities as well as agility and handling qualities.

In addition, Sikorsky has been developing S-97 Raider (Fig. 1.11) as a high-speed scout attack compound helicopter based on the advancing blade concept (ABC). The S-97 has a full authority fly-by-wire flight control system, with passive side arm controllers and a single centrally mounted collective. The control response type provided by the FCS is rate command with attitude hold in both pitch and roll and proportional collective in the vertical axis, with full time trim engaged [18].

In 2017, the S-97 suffered a hard landing which resulted in damage to the outer section of all of its rotor blades and minor injuries to the two experimental test pilots on board [18]. As the flight control software transitioned from ground to air mode during takeoff, the pilot stick sensitivity was greater than expected and the aircraft exhibited an extreme roll oscillation. The rotor blades of the coaxial counter-rotating rotors contacted each other due to gyroscopic moments generated by the high roll rate. As noted in the NTSB report [18], once the incident timeline was established, the issue of increased pilot stick sensitivity could be consistently replicated in the simulator. This demonstrates the continuing challenges associated with advanced rotorcraft configurations and the importance of a validated model of the aircraft and



Figure 1.8. Airbus Helicopters (formerly Eurocopter) X³, fastest non-jet augmented compound helicopter.



Figure 1.9. Bell V-280 Valor tiltrotor, one of two Joint Multi-Role Technology Demonstrator aircraft.

control system for off-line and piloted simulation.

In Europe, Leonardo is currently working on the development of the AW609 tiltrotor. The aircraft began as a joint venture between Bell Helicopter and AgustaWestland, and first flew in March 2003. In 2015, severe lateral-directional oscillations during a high-speed dive resulted in a fatal crash of an AW609. The aircraft's excessive sideslip forced its proprotors to hit its wings multiple times, damaging the hydraulic and fuel lines, and causing an in-flight breakup and fire. The crash investigation report from Italy's flight safety national agency (ANSV) [19] points to two main causes for the crash: the inability of the fly-by-wire flight control system (FCS) control laws to



Figure 1.10. Sikorsky Boeing SB-1 Defiant Advancing Blade Concept (ABC) rotorcraft, one of two Joint Multi-Role Technology Demonstrator aircraft.



Figure 1.11. Sikorsky S-97 Raider, high-speed scout and attack compound helicopter based on the Advancing Blade Concept (ABC).

maintain controlled flight and a project simulator that “did not foresee the event in any way.” Again, this demonstrates the challenges inherent to advanced rotorcraft design and the importance of a validated flight dynamics simulation model for flight control system development and testing.

1.1.3 Future

In the U.S., the FVL program is continuing with the Future Attack Reconnaissance Aircraft (FARA), a high-speed capable armed scout attack helicopter meant to replace the retired OH-58D Kiowa Warrior (e.g., Fig. 1.13), and the Future Long-Range Assault Aircraft (FLRAA), a high-speed capable assault aircraft meant to replace



Figure 1.12. Leonardo AW609, aimed to be the first production tiltrotor for civil use.

the UH-60 Black Hawk. For FARA, two concepts will be down-selected in mid-2020 for detailed design and testing, with the first prototype flight projected for late 2022. FARA would transition to an acquisition program of record (PoR) in 2024, and reach initial operating capability (IOC) in 2028 [20]. FLRAA is expected to reach IOC in 2030 [20]. Both FARA and FLRAA aircraft will be required to have Level 1 handling qualities.



Figure 1.13. AVX-L3 Future Attack Reconnaissance Aircraft (FARA) prototype proposal, one of five concepts under development for the FARA program.

In Europe, the Clean Sky 2 Joint Undertaking (CSJU), launched in 2014, is a public-private partnership between the European Commission and the European aeronautics industry tasked with achieving environmental performance targets. Through the Clean Sky 2 Fast Rotorcraft Innovative Aircraft Demonstrator Platform (IADP), the EU is funding two high-speed rotorcraft: the Airbus RACER compound helicopter (Fig. 1.14) and the Leonardo Next-Generation Civil Tiltrotor (NGCTR) (Fig. 1.15). For the RACER aircraft, Airbus plans to begin flight demonstrations in 2020, and

focus on assessing key performance objectives including speed, handling qualities, stability, and aerodynamics [21]



Figure 1.14. Airbus RACER (Rapid and Cost-Effective Rotorcraft), one of two aircraft being developed as part of the Clean Sky 2 Fast Rotorcraft Innovative Aircraft Demonstrator Platform.



Figure 1.15. Leonardo Next-Generation Civil Tiltrotor, one of two aircraft being developed as part of the Clean Sky 2 Fast Rotorcraft Innovative Aircraft Demonstrator Platform.

Electric Vertical Takeoff and Landing (eVTOL) configurations are also an emerging area of advanced vertical lift, primarily designed for use as person air vehicles (PAV) for urban air mobility (UAM). Figures 1.16 through 1.18 show three of the many examples of such configurations—The Airbus A³ Vahana which is a piloted eVTOL aircraft, the Boeing NeXt autonomous eVTOL aircraft, and Bell Nexus hybrid-electric VTOL aircraft. Although the maximum speeds of these aircraft is typically less than the “high speeds” considered in this research, many of the eVTOL configurations have redundant controls or reconfigure in flight (e.g., with tilting wings or nacelles).

Therefore, although the handling qualities requirements studied here may not be directly applicable to eVTOL configurations, many of the control system design recommendations will be applicable.



Figure 1.16. Airbus Vahana eVTOL aircraft, eight-propeller tilt-wing demonstrator.



Figure 1.17. Boeing NeXt autonomous eVTOL prototype.

1.1.4 Challenges

Advanced rotorcraft configurations, such as coaxial compound helicopters and tiltrotors, bring with them new, unique challenges and opportunities in flight control and handling qualities. For example, the ability to decouple response characteristics that have traditionally been coupled in helicopters (such as independent thrust control), reconfiguration in flight (such as moving nacelles), and redundant controls (such as rotors and aero-surfaces). Designing flight control systems that provide good handling qualities for these configurations will depend on three key areas:

1. What response types during hover, transition, and forward flight do pilots prefer?



Figure 1.18. Bell Nexus hybrid-electric VTOL aircraft concept.

2. What are the qualitative handling-qualities metrics (i.e., Mission Task Elements or handling qualities demonstration maneuvers) that will be useful to assess differences between response types and control systems?
3. What are the quantitative handling-qualities metrics and their associated boundaries that the control systems should be designed to meet?

In terms of response types (#1), The U.S. Army’s rotorcraft handling-qualities specification Aeronautical Design Standard-33E-PRF (ADS-33E) calls for a rate command response type in forward flight, while the DoD flying qualities specification MIL-STD-1797B inherently requires an aircraft response type be lower order, similar to a classical fixed-wing aircraft (through the use of lower-order equivalent systems specifications). Many of the advanced rotorcraft configuration being considered have redundant controls. The use of traditional cockpit controls and response types, therefore, may cause high pilot workload as the pilot is required to manage thrust vectoring and flight path control manually across a wide speed envelope ranging from hover to high-speed forward flight.

Item #2 above was recently addressed by a U.S. Army National Rotorcraft Technology Center (NRTC) project titled “Rotorcraft Handling Qualities Requirements for Future Configurations and Missions,” which was a collaborative effort between industry, academia, and government. Roughly a dozen Mission Task Elements (MTEs) were developed and tested as part of the NRTC project, and subsequently, seven were down-selected for recommendation for inclusion in an ADS-33E update. These seven MTEs will make a good basis for assessing flight control systems for advanced rotorcraft configurations at high speed, and the research conducted here was the first time that these MTEs were used for a formal handling qualities evaluation.

Finally, Item #3 will require careful selection of handling qualities and control system specifications from existing rotorcraft and fixed-wing criteria. It is still unknown,

and something this research seeks to answer, which specification will be applicable for different parts of the flight envelope and for different response types.

1.2 Previous Research

Development of handling-qualities requirements has a long history for both fixed-wing aircraft and rotorcraft, and requirements from both communities will be important for advanced high-speed rotorcraft configurations. In addition, there has been a significant amount of work on flight controls for compound rotorcraft, primarily looking at using redundant controls for decreased power usage and increased maneuvering performance. The literature review provided here is divided into three main topics: 1) Rotorcraft handling qualities requirements, 2) Fixed-wing handling qualities requirements, and 3) Flight controls for advanced rotorcraft configuration.

1.2.1 Rotorcraft Handling Qualities Requirements

The first helicopter handling-qualities specification was MIL-H-8501A [22], published in 1961. The requirements in MIL-H-8501A considered time-domain parameters such as stick force gradients, frequency and damping of oscillatory modes, and angular displacement during step responses. Both the U.S. Army and Navy continued to use MIL-H-8501A into the 1970s and 80s, using it to procure the UH-60, AH-64, SH-2, SH-60, and CH-53 [23].

In the 1960s, the U.S. Air Force sponsored work on a vertical/short take-off and landing (V/STOL) aircraft handling-qualities specification, which results in the adoption of MIL-F-83300 [24] in 1970. MIL-F-83300 was structured similar to the fixed-wing handling-qualities requirement of the time (MIL-F-8785B, [25]), and used the same parameters for many of the requirements. The Army and Navy did not adopt MIL-F-83300 for use as a helicopter handling-qualities specification, but the Navy did use it for the V-22.

Aeronautical Design Standard (ADS)-33 Handling Qualities Requirements for Military Rotorcraft is the current rotorcraft handling qualities specification. It was initially adopted in 1987 [26], and introduced several new concepts in handling-qualities testing. Among these new concepts was a list of stylized maneuvers (or Mission Task Elements, MTEs) designed to evaluate the aircraft handling qualities and pilot workload to both single and multi-axis inputs. Thus, the requirements were divided into two categories: quantitative or predicted handling-qualities requirements and qualitative or assessed handling-qualities requirements.

The MTEs are mission oriented maneuvers including precision tasks, aggressive tasks, and tasks in both Good and Degraded Visual Environments (GVE and DVE). The original MTEs were geared towards the scout/attack mission, and provided desired performance standards only (with no adequate standards).

Several refinements were made to the standard, as more data and experience were gathered using ADS-33, leading to the release of ADS-33B [27] in 1988 followed by ADS-33C [28] in 1989, which saw the shift of quantitative requirements from the time domain to the frequency domain. In 1990–91, ADS-33C was assessed using an AH-64A Apache by the U.S. Army Aviation Qualification Test Directorate (AQTD) [29]. Two of the major conclusions of the AH-64A flight test assessment of ADS-33C given in Ref. 29 are: 1) The maneuvers were not defined enough to allow for repeatable results, and 2) Adequate standards were needed for each maneuver to be able to fully utilize the Cooper-Harper Handling Qualities Rating (HQR) scale [30]. The results and experience of the AH-64A flight tests contributed to the next version of ADS-33, version D [31,32], which included significant improvements to the flight test maneuvers section. However, the maneuvers were still focused on the scout and attack missions.

Between 1994–2000, the Aeroflightdynamics Directorate (now the U.S. Army Combat Capabilities Development Command Aviation & Missile Center Technology Development Directorate, TDD) and AQTD conducted simulation and flight tests to support the development of criteria that are applicable to cargo and utility helicopters. These included CH-47D Chinook flight tests in 1994–95 [33,34] and UH-60A Black Hawk flight tests in 1999 [35,36]. The results of the CH-47D and UH-60A flight tests were used to expand the MTEs to cover the utility and cargo missions with and without externally slung loads, and resulted in the latest update to ADS-33, version E [16], released in 2000.

The latest version of ADS-33 still focuses on the hover/low-speed regime (with limited coverage of high speed above about 140–150 kts) and conventional configurations. The ADS-33E MTEs are based on legacy vehicles and were developed at an early stage of rotorcraft fly-by-wire (FBW) technology [37]. Therefore, the current high-speed MTEs in ADS-33E are not well vetted for high-speed flight or for modern complex configurations. For additional details on the background and evolution of ADS-33, the reader is referred to Refs. 35,37, and 23

Now, with the development of advanced high-speed and complex-configuration vertical lift aircraft, through the DoD FVL initiative and JMR-TD flight test program, the next evolution of ADS-33 will require new high-speed handling-qualities requirements. As with past revisions of ADS-33, these new handling-qualities requirements must include the stylized maneuvers, or MTEs, to qualitatively assess handling qualities in flight or simulation, as well as quantitative requirements useful for control system design.

To develop the new qualitative requirements, the U.S. Army National Rotorcraft Technology Center (NRTC) recently completed a multi-year project titled “Rotorcraft Handling Qualities Requirements for Future Configurations and Missions” [38–45], which was a collaborative effort between industry, academia, and government. Roughly a dozen MTEs were developed and tested as part of this project. Subsequently, seven MTEs were down-selected for recommendation for inclusion in ADS-33:

1. Break Turn [42],

2. High-Speed Acceleration [43],
3. High-Speed Deceleration [43],
4. Bank Angle Capture and Hold [44],
5. Pitch Attitude Capture and Hold [44],
6. Roll Sum-of-Sines Tracking [45], and
7. Pitch Sum-of-Sines Tracking [45].

The development and initial testing of the proposed MTEs was limited to simulator studies only. Subsequently, a flight-test evaluation of the Break Turn and High-Speed Acceleration/Deceleration MTEs was conducted using a UH-60M to help standardize the parameters and cueing requirements, and ensure the MTEs focus on handling qualities and not aircraft performance [46]. The flight test showed that both MTEs proved to be well suited for evaluating high-speed helicopter handling qualities using the timing requirements, and desired and adequate tolerances from the NRTC studies. These NRTC MTEs were used to assess the handling qualities of advanced rotorcraft configurations in the research presented here.

1.2.2 Fixed-Wing Handling Qualities Requirements

Fixed-wing handling qualities research and specification have a long history, covered in detail in Refs. 23, 47–50. There have been many debates over the years about the best metrics and approaches to characterize the aircraft responses and assess handling qualities, and no single approach has emerged as the preferred approach [50]. The debates include, for example, time- versus frequency-domain criteria, lower-order equivalent system (LOES) modeling versus non-parametric frequency responses, and aircraft only versus pilot-in-the-loop analysis.

Initial fixed-wing aircraft handling-qualities requirements (MIL-F-8785, 1954 [51]) were placed on control characteristics and simple limits on modal responses. In the 1960s, handling-qualities specifications shifted to the modal parameters themselves (e.g., short period frequency and damping, roll mode time constant, etc.), and were formally issued in MIL-F-8785B(ASG) (1969, [25]). In 1980, MIL-F-8785C [52] was adopted and included the equivalent systems approach to be able to apply the existing handling qualities databases and criteria to higher-order (i.e., augmented) aircraft responses. In the 1990s, with the adoption of MIL-STD-1797A [53], there was a shift from explicit requirements to providing a list of alternate criteria with background information meant to help the handling qualities engineer select the most appropriate criteria. Today, MIL-STD-1797B [17] is the current fixed-wing handling-qualities specification. Like its predecessor, MIL-STD-1797A, it provides several alternate criteria. For example, in addition to the five LOES-based requirements for the short-period pitch response, seven alternate criteria are given, including time-domain requirements like the Gibson dropback criteria, frequency-domain requirements like the pitch attitude bandwidth and phase delay criteria, and pilot-in-the-loop requirements

like the Neal-Smith criteria.

In 2000, Gibson, Lodge, and Fieldings from BAE Systems in the UK, published their work on handling qualities design criteria for short take-off, vertical landing (STOVL) aircraft [54]. They considered that the concepts behind the Gibson criteria that were developed and successfully applied to conventional take-off and landing (CTOL) aircraft, could be applied to STOVL aircraft in jet-borne flight. They noted that the existing design criteria of MIL-F-83300 were largely based on mechanical control systems with rate command response types, and that such criteria were inappropriate for fly-by-wire designs capable of highly-augmented response types. They carried out a series of piloted simulation experiments to determine the desirable and undesirable characteristics for a series of response types. They used a motion based simulator and an aircraft model composed of fourth-order transfer functions for each of the six rigid body degrees of freedom. Three right hand inceptor (RHI) response types were investigated: height rate command, pitch attitude command, and fore/aft translation rate command. The results of the work were intended to set boundaries for control law design, with an emphasis on time response boundaries for basic handling qualities, supplemented by frequency response boundaries for the prevention of PIO induced by higher-order effects. However, the results were based on the observations of a single engineering pilot.

The application of fixed-wing handling qualities criteria to a tiltrotor in airplane mode is obvious. However, do these criteria apply to high-speed compound helicopters as well? This remains an open research question, and the research presented here aims to address it.

1.2.3 Advanced Rotorcraft Configuration Flight Controls

1.2.3.1 Tiltrotor

In 2002, Meyer and Padfield published a paper [55] that attempted to begin defining predicted handling-qualities criteria for tiltrotor since a generic handling-qualities specification for tiltrotor aircraft does not exist. Their aim was to select from existing fixed-wing and rotorcraft handling-qualities criteria to form a good starting point from which further development/refinement can take place. The paper reviewed existing criteria from MIL-STD-1797A, a proposed update to MIL-STD-1797A [56], MIL-F-83300, ADS-33E, and FAR/JAR-29 and found that there are no specific tiltrotor handling-qualities criteria and no criteria for conversion mode available in open literature. Furthermore, they postured that for a notional civil tiltrotor search and rescue (SAR) mission, continuous flight in partially converted configuration would be beneficial for loitering and searching. They developed a roll-step maneuver (somewhat similar to the ADS-33E Slalom MTE) and conducted motion-based simulator evaluations using an XV-15 FLIGHTLAB model developed at the University of Liverpool which includes the XV-15 SCAS. The focus of the simulator trials was to: 1)

Understand and define what constitutes Level 2 handling qualities and 2) Establish the Level 2/Level 3 boundaries based on any handling-qualities cliffs. This was done by assessing the impact of increasing speed and changing attitude bandwidth on the roll-step maneuver. Three discrete flight conditions were tested all within the conversion corridor: $\delta_{\text{nac}} = 90$ deg, $V = 60$ kts; $\delta_{\text{nac}} = 75$ deg, $V = 100$ kts; and $\delta_{\text{nac}} = 60$ deg, $V = 140$ kts. Their results suggest that the XV-15 with its core SCAS is Level 2 for the maneuver tested, and that no handling-qualities cliffs emerged across a wide range of configurations ($\omega_{\text{BW}} = 0.8 - 2.5$ rad/sec, $\tau_p = 0.02 - 0.25$ sec)

In 2007, Cameron and Padfield published a paper [57] aiming to fill in criteria gaps and develop a database of piloted simulation results in support of the European civil tiltrotor program. They investigated open-loop/bare-airframe pitch and flight-path handling qualities for degraded mode evaluations in airplane mode only. They noted that large pitch rate overshoot, leading to large pitch attitude dropback DB_θ , is more severe in tiltrotors than conventional fixed- or rotary-wing aircraft due to proprotors making positive contributions to both the pitch damping derivative M_q and the static stability derivatives M_w , resulting in a large value of numerator zero in the pitch rate transfer function T_{θ_2} . The objective of Cameron and Padfield's research was to develop a more unified short period mode handling-qualities criteria to replace the lower-order equivalent systems Control Anticipation Parameter (CAP) [17], bandwidth [16, 17], and dropback [17] specifications, which properly integrates the dropback phenomenon. Qualitative results and initial simulator testing was done using three tiltrotor models available at the University of Liverpool (XV-15, Eurotilt, and EuroFAR), all tested in airplane mode at $V = 200$ kts. However, the majority of the simulator testing was done with a simple, short-period approximation model. Two tasks were flown in the simulator: 1) Flight path capture task (Heave-hop MTE) and 2) Pitch attitude tracking task based on the Drogue Tracking Task. The results of the simulator study suggest that low values of T_{θ_2} (≈ 0.5 sec) are preferable for the flight path capture task, while for the drogue tracking task low values of T_{θ_2} got very few Level 1 ratings and more Level 3 ratings. In addition, for the drogue tracking task, as values of T_{θ_2} increased, the majority of the configurations tested became easier. This is explained by the fact that for the drogue tracking tasks, pilot preferred to have high pitch acceleration as they were constantly applying longitudinal stick inputs, and a steady state pitch rate was never achieved. However, for the flight path capture task, pilots applied step input until a steady state pitch rate was achieved, then removed their input to capture the desired pitch attitude, thus the capture task is a lower bandwidth task. For both tasks, pilots did not like a large amount of dropback as this made the aircraft pitch response less predictable. A new criteria was proposed with boundaries drawn on a plot of CAP vs. $\text{DB}_\theta/\alpha_{\text{ss}}$ (with $\text{DB}_\theta/\alpha_{\text{ss}} = 1 - T_\gamma/T_{\theta_2}$). However, the proposed boundaries were based on a limited number of configurations all tested at one airspeed.

In 2008, continuing their research into tiltrotor handling qualities, the University of Liverpool conducted a workshop/competition to provide experience and education

in integrated flight control and handling qualities for tiltrotors. Teams were provided specifications and linear models of the XV-15, with the objective of designing flight control systems that provide Level 1 handling qualities across the flight envelope for a predefined SAR mission. The Pennsylvania State University (Penn State) participated in the workshop and published a paper on their results in 2009 [58]. The handling-qualities requirements used came from ADS-33E for hover/low-speed and conversion and from MIL-F-8785C (for Class II aircraft, Category A flight phase) for airplane mode. Linear models of the XV-15 were derived from a nonlinear FLIGHTLAB model at airspeeds ranging from hover to 240 kts in 20 kt increments, spanning helicopter, conversion, and airplane modes. The linear models included the control gearing of the XV-15, and so control allocation was not part of this study. Nonlinear dynamic inversion control laws were developed that were scheduled with airspeed only (and not nacelle tilt angle). The control laws provided Attitude-Command/Attitude-Hold (ACAH) response type in the pitch and roll axes in hover/low-speed, Rate-Command/Attitude-Hold (RCAH) response type in the pitch and roll axes at high speed, RCAH in yaw with turn coordination at high speed, and open-loop control of rotor collective pitch in all configurations. A comprehensive set of specifications was evaluated for the closed-loop XV-15 plus dynamic inversion control system which showed overall predicted Level 1 handling qualities. The only requirements evaluated which did not meet Level 1 were roll and yaw attitude quickness at low speed, attributed to large roll and yaw moments of inertia of the XV-15 (with its engines and rotors on the ends of the wings) and actuator rate limiting. The control laws were then tested in piloted simulation using the University of Liverpool HELIFLIGHT motion-based simulator and several MTEs from ADS-33E. In the helicopter configuration, the control laws shows substantial improvements in handling qualities, going from a Level 3 bare-airframe to a Level 1 for the augmented aircraft. Unfortunately, the model was not tested in airplane mode due to an instability in forward flight (possibly due to an integrator windup issues). Overall, the dynamic inversion controller with its model following two degree-of-freedom (2 DOF) architecture provided an effective design tool for tiltrotor flight control.

In 2017, Vigano, Riccardi, and Leonello studied augmented control laws for a tiltrotor in low- and high-speed flight [59]. Their paper summarized a research project by Leonardo Helicopter Division to develop control laws that use sidestick inceptors, ensure satisfactory decoupling between the various tiltrotor control axes, and provide smooth operation across all flight phase. The system they developed provided an F-35 Unified Control Laws-like response type, where the thrust control lever (TCL, left-hand inceptor) provided longitudinal Translational Rate Command (TRC) in hover/low-speed and Linear Acceleration Command/Speed Hold (LACSH) in airplane mode. The sidestick (right-hand inceptor) provided Rate Command/Height Hold (RCHH) heave axis control in hover/low-speed and flight path Rate-Command/Attitude-Hold (RCAH) in airplane mode. The pedals provided Rate-Command/Direction-Hold (RCDH) in hover/low-speed and Rate-Command (RC) with Turn-Coordination (TC)

in airplane mode. An airplane-style command strategy throughout the flight envelope (as described) was considered to be the most promising solution for a next generation tiltrotor.

Design and testing was done using a FLIGHTLAB model of a medium-size tiltrotor. The baseline control laws were modified to include auto-trim functions for the roll, pitch, and thrust channels as new active inceptors were used in a unique-trim configuration. The existing thrust and power management system of the FLIGHTLAB model and existing gearing law between equivalent stick demands and the physical tiltrotor control surfaces (i.e., original ganged control allocation matrix) were not modified.

One issue identified by the authors was the selection of an appropriate flight path command scheme for airplane mode. They recognized that flight path rate command was preferred to attitude command when using a sidestick with unique-trim to provide better agility. However, since they noted a lack of knowledge and available data on flight path RCAH systems, they used a C^* response type, given by:

$$C^* = \Delta n_z + \frac{V_{co}}{g} q \quad (1.1)$$

where:

Δn_z is the incremental load factor (obtained by washing out measured vertical acceleration in order to remove steady-state gravity contributions),

V_{co} is the crossover speed, selected as 200 kts in this case (slightly lower than 210 kts typically used for jetliners), and

q is the pitch rate.

The commanded C^* (C_{cm}^*) is based on the relationship of pitch rate to normal acceleration ($q = \Delta n_z g / V$) and is given by:

$$C_{cm}^* = q_{cm} \frac{V_{co}}{g} \left(\frac{V}{V_{co}} + 1 \right) \quad (1.2)$$

Feedback on C^* error consisted of proportional and integral (PI) gains. The authors note that the integral path is roughly equivalent to pitch attitude and vertical speed feedback which provides favorable characteristic of strongly damped phugoid and ensures flight path stability (at the expense of speed stability). The controller governing and decoupling flight path and airspeed is multivariable (2×2), and the authors applied H_∞ optimization techniques to their design. The paper shows time histories of the aircraft pitch and throttle responses in airplane mode, although no analysis was done in a conversion configuration. In addition, the system was not evaluated against any handling-qualities specifications or in piloted simulation. However, the authors argue that a selected control strategy should apply a certain set of response types to each cockpit control axis throughout the flight envelope, that the most promising solution identified so far makes use of active sidesticks allocated

according to an airplane control strategy, and that the C^* control scheme appears promising for a tiltrotor when flying in airplane mode and during the last portion of conversion corridor.

1.2.3.2 Compound Rotorcraft

A significant amount of research has been done at Penn State regarding compound helicopter flight control design and handling qualities analysis. In 2013, Ozdemir, Horn, and Thorsen [60] designed and tested a “Fly-to-Optimal” (FTO) controller to perform inflight optimization of redundant control effectors on a compound rotorcraft to minimize power required. The aircraft model used was a modified UH-60A with reduced drag, a low-mounted wing, and a pusher propeller, modeled in GENHEL-PSU. The objectives of this work were to optimize the redundant controls available on the compound helicopter as well as rotor speed to improve power required. The work built on the Adaptive Performance Optimization (APO) approach, which performs low-frequency sweeps of the redundant controls and assess changes in power required, but improved the optimization method and extended it to multi-variable optimization by performing a two-dimensional search. The flight control system used was a dynamic inversion (DI) architecture with inner-loop control of roll, pitch, yaw, heave, and RPM and a high-speed autopilot/outer-loop control of forward speed and altitude. The outer loop also used a DI architecture, using a second-order linear model of airspeed and climb rate extracted from the higher-order linear model of the aircraft. The research only investigated the control laws and FTO at high-speed, and control allocation for the outer loop was determined manually by using pitch attitude to control airspeed and collective to control climb rate for $V < 180$ kts (with propeller pitch as the redundant control to be optimized by FTO), and pitch attitude to control climb rate and propeller pitch to control airspeed for $V > 180$ kts (with collective as the redundant control to be optimized by FTO). The results showed that the control laws were able to effectively hold flight condition while the FTO method was effective at improving performance. The authors also noted that it was critical to have an accurate and well-designed flight control system to maintain trimmed flight condition during FTO.

In 2014, Thorsen, Horn, and Ozdemir investigated control allocation for the same compound rotorcraft to optimize performance in maneuvering flight (acceleration, pull-up, and turning maneuvers) [61]. They leveraged the propeller and symmetric wing flaps as redundant controls, and used trim analysis of quasi-steady maneuvers to derive control allocation that minimizes power required. Thorsen’s control architecture was similar to that of Ozdemir [60] described above, using a DI inner loop to control roll, pitch, and yaw, and a DI outer loop to control airspeed and climb rate based on a second-order linear model extracted from the higher-order linear model of the aircraft. In addition, the outer loops in this case commanded longitudinal and normal load factors (g-command), and incorporated minimum power schedules developed

from off-line simulation that allocated the controls based on airspeed and commanded longitudinal and normal load factors. Thus, for changes in commanded airspeed, command accelerations were converted to a combination of pitch attitude and propeller pitch to follow the minimum power curve, while for pull-up or turn maneuvers, vertical load factor command was converted to a combination of collective, flaps, and pitch attitude to follow the minimum power curve. The results shown were for piloted simulations of a level acceleration and an accelerated climbing turn. Overall, the results showed that the g-command controller had significant power savings when using redundant controls following a minimum power path in basic accelerations.

Thorsen, Horn, and Ozdemir then expanded on the work to use redundant controls on a compound helicopter for power minimization in quasi-steady maneuvers by using the redundant controls to enhance the transient response and handling qualities of the aircraft [62]. While the focus of the previous work [61] was on minimizing power required, the focus of ref. 62 was instead on establishing methods of control that enhance handling qualities via use of redundant controls surfaces. Three redundant controls were used in this work: propeller pitch and symmetric flap deflection (as before), and differential flap deflection. The control laws used were the same as the g-command DI controller in their previous research. Trim values of the redundant controls were scheduled based on performance optimization for minimum power required (as in Ref. 61). The controller could then reallocate control during maneuvers (transient allocation) to enhance maneuvering performance. The transient allocation was not optimized in this study, but simply employed to show potential of redundant controls in shaping transient response characteristics and impacting handling qualities. The study only focused on high-speed and looked at three handling-qualities metrics: agility quickness [63], roll attitude quickness, and turn quickness [63]. The results shown were step and pulse responses to assess quickness in the different axes, and suggested that use of redundant controls can enhance transient response through increased quickness during transient flight.

In 2016, Thorsen and Horn [64] again expanded on the compound helicopter control laws by developing a Unified Control Architecture (inspired by work that was done on the Vectored Thrust Aircraft Advanced flight Controls [VAAC] program [65] and eventually used on the F-35B [66]) and evaluating it in maneuvering flight. This work was motivated by the desire to unify all the controls in one control architecture with intuitive pilot control strategy and seamless flight regime transitions. An energy management system was also developed and integrated into the controller in Ref. 64 to limit longitudinal acceleration, vertical speed, and bank angle to allow for more carefree maneuvering. The flight control system used an inner-loop nonlinear dynamic inversion (NDI) with full rigid-body state feedback to control bank angle, pitch attitude, and yaw rate. The outer loop used a DI architecture based on a second-order linear model extracted from the higher-order linear model of the aircraft as in Refs. 60–62. This time though, three outer-loop command modes were investigated which commanded longitudinal and vertical velocities and/or accelerations. In all cases and throughout

the entire flight envelope, the pilot left hand inceptor (LHI) was a fore/aft throttle-style inceptor used to control the longitudinal axis, while the right hand inceptor (RHI) was a conventional sidestick used to control the vertical axis. Thorsen and Horn's work expanded on using the DI outer loop for hover and low-speed flight. As before, the DI outer loop was used to allocate collective, propeller pitch, and pitch attitude. Below $V = 120$ kts, propeller pitch was not used for maneuvering and the DI outer loop which regulated two states (airspeed and climb rate) had two inputs (square system, as desired). Above $V = 155$ kts, collective was not used for maneuvering and again the DI outer loop was a square system, as desired. Between $V = 120 - 155$ kts, all three controls were active, and the DI outer loop had a redundant control. Therefore, a weighted pseudo-inverse was used for control allocation of the outer loops between $V = 120 - 155$ kts. Piloted simulator evaluations were conducted to investigate level accelerations from hover to maximum speed, vertical climbs in hover and at high speed, and a pull-up and a high-speed transient figure-8 turn. Overall, the results showed good controller tracking in all controller modes, with minor issues with vertical speed tracking during mode transitions. The weighted pseudo inverse technique to transition outer-loop control effectors worked well. In general, the fore/aft throttle LHI as opposed to a conventional helicopter collective stick felt intuitive for speed commands, but a mechanical detent at the midpoint would have been favorable. Finally, the authors noted that although the fore/aft LHI concept for rotorcraft showed promise, it should be assessed in more detail.

In 2017, Horn, Thorsen, and Conti studied handling-qualities requirements for high-speed compound rotorcraft [38]. They noted that the modern methodology for handling qualities analysis involves two paths: 1) Development of demonstration maneuvers (MTEs) for subjective pilot evaluations, and 2) Definition of quantitative criteria for aircraft dynamics to predict handling qualities (useful for FCS design). Their research was focused on the first path (MTE development), as they studied two new MTEs: Break Turn and Decelerating Approach with Precision Approach Path Indicator (PAPI) Lights. They used a generic winged compound model based on GENHEL-PSU, and designed a nonlinear dynamic inversion (NDI) controller. The controller provided an RCAF response type in the roll axis, ACAH in the pitch axis, turn coordination in yaw, open loop collective, and direct propeller pitch control with airspeed scheduled trim and upper/lower constraints. The roll axis command model was varied to generate four configurations of varying bandwidth and phase delay values ($\omega_{BW} = 1.35 - 2.57$ rad/sec), which were tested using the Break Turn MTE at $V_{BR} = 170$ kts. Correlation of the predicted handling qualities and assigned HQRs for the Break Turn showed the right trend, but the assigned HQRs were all Level 2 with some into Level 3. The higher assigned HQRs were attributed to cueing limitations in the simulator. A novel and important development of this work was refining the tolerance and timing requirements of the Break Turn MTE. Specifically, developing a variable time to complete that is a function of aircraft performance as a way to separate aircraft performance from handling qualities testing.

In 2018, Yuan, Thomson, and Chen studied propeller control strategies for coaxial compound helicopters to improve performance and flight dynamic characteristics [67]. They assessed the bandwidth/phase delay, eigenvalue damping, and inter-axis coupling criteria from ADS-33E, but evaluated the bare-airframe only (no control system). The different rotors and propeller thrust allocation schemes they investigated were not just used for trim but also represented the allocation of longitudinal control between the rotor and propeller. They noted that for some configurations, inter-axis coupling became an issue at high speed.

1.3 Research Objectives and Contributions

There has been a significant amount of research devoted to developing high-speed handling-qualities requirements and flight control system design guidance for rotorcraft, as described above. The majority of the research though was done based on one aircraft configuration (e.g., winged compound, tiltrotor, etc.) and generally at a limited number of flight conditions or aircraft configuration. In addition, with the exception of Ref. 58, only a very limited (in most cases one or two) handling-qualities specifications were investigated.

The handling-qualities requirements and MTEs are supposed to be mission focused (and not aircraft configuration focused). However, whether or not this holds across advanced rotorcraft configurations (e.g., compound versus tiltrotor) remains to be seen. That is why the research done here investigated two configurations.

Furthermore, as discussed above, the NRTC recently completed a project to develop high-speed Mission Task Elements to assess handling qualities and pilot workload of advanced high-speed rotorcraft in simulation or flight test. However, before this project, there had not been a formal piloted handling qualities evaluation of advanced flight control laws using the newly developed high-speed MTEs.

Therefore, the objectives and contributions of this report are to:

1. Create real-time simulation models of a generic lift offset coaxial compound helicopter with pusher propeller (coaxial-pusher) and a generic tiltrotor aircraft by combined linear models and trim data of each aircraft extracted from the comprehensive rotorcraft simulation code HeliUM into stitched simulation models.
2. Use a weighted pseudo-inverse to design inner-loop (moment demand) control allocation using each configuration's redundant controls.
3. Use a handling-qualities optimization based approach to design inner-loop (rate-command) explicit model following (EMF) control laws for both aircraft configurations across their entire flight envelopes. A major contribution was the

selection of applicable criteria throughout the flight envelope and a compilation of a comprehensive set of stability, handling-qualities, and performance specifications for advanced high-speed rotorcraft flight control design.

4. Extend the work by Thorsen [64] to design outer-loop (flight path and airspeed command) control laws for both configurations across their entire flight envelopes. This included developing a control allocation scheme to distribute longitudinal and vertical acceleration commands to the aircraft pitch attitude, collective, and propeller pitch (coaxial-pusher) or nacelle angle (tiltrotor). A weighted pseudo-inverse approach was used, as was done in Ref. [64] for a limited speed range, but now extended to the whole flight envelope.
5. Perform a formal handling-qualities evaluation of both inner- and outer-loop control laws for both configuration using the newly developed high speed Mission Task Elements. Such an evaluation has not been previously done for high speed rotorcraft control laws.
6. Ultimately, this work seeks to establish which response types pilots prefer during transition and high-speed flight for a range of tasks, and which rotorcraft and fixed-wing handling-qualities criteria are applicable to the different response types.

1.4 Organization

This report is organized as follows. Chapter 2 provides an overview of the generic coaxial-pusher and tiltrotor flight dynamics models developed in the comprehensive rotorcraft simulation code HeliUM. First, detailed descriptions of the physical aircraft parameters used in HeliUM are provided. This is followed by a discussion of the linear models extracted from HeliUM, including description of trim, key stability and control derivatives, the rotor modes retained in the linear models, and the eigenvalues of the linear models. Subsequently, the model stitching architecture used to simulate the models in real-time is discussed. The control allocation method used to distribute the desired moments to the multiple control effectors is shown for each aircraft. Finally, primary on-axis frequency responses are provided for each aircraft at a range of airspeeds.

Chapter 3 provides a description of the inner-loop flight control systems that were designed and optimized for both coaxial-pusher and tiltrotor aircraft. First, a description of the control law architecture is provided, including discussion of all major elements of the block diagram. Next, the flight-control and handling-qualities specifications used to drive the control system optimization are given. After a brief discussion of the optimization strategy used, the results of the control system optimization is provided for both aircraft in the form of predicted handling qualities.

Chapter 4 discusses the results of a piloted handling-qualities simulation experiment investigating the inner-loop control laws of both aircraft. After a brief description of the simulation experimental setup, the results of the experiment are provided first for the coaxial-pusher and then for the tiltrotor. The chapter concludes with a discussion of the results.

Chapter 5 describes the outer-loop control laws that were designed for both coaxial-pusher and tiltrotor aircraft. First, the design approach and architecture used for the outer loops is discussed, including the outer-loop control allocation of longitudinal and vertical acceleration commands to pitch attitude, collective, and propeller pitch/nacelle tilt angle. Next, the design specifications used for control law optimization are provided. Control law optimization results are presented for both coaxial-pusher and tiltrotor, first at one flight condition, and then at several flight conditions spanning their respective flight envelopes. Finally, a discussion of the outer loops is provided.

Chapter 6 describes the results of a piloted handling-qualities simulation experiment that was carried out at the Pennsylvania State University Flight Simulator facility to test the outer-loop control laws described in Chapter 5. A description of the experimental setup is provided, including a descriptions of the facility and the tasks flown by the pilots. The results for both the coaxial-pusher and tiltrotor configuration are provided followed by a discussion.

Chapter 7 describes the results of a piloted handling-qualities simulation experiment to test different variations of the inner-loop pitch-axis command model response types for both aircraft. Variations in the pitch-axis command model that were tested are described. Then, results for the simulation evaluation are provided.

Chapter 8 describes a robustness analysis that was done for the inner- and outer-loop control laws of both aircraft using an Unscented Transform approach. First, a description of the bare-airframe parameter perturbations is provided. Next, a brief overview of the Unscented Transform is provided followed by a simple example of its application to assessing control system robustness. Results are then shown for both the coaxial-pusher and tiltrotor inner- and outer-loop control laws, followed by a discussion.

Finally, Chapter 9 provides the conclusions of this research, control system design guidance for advanced high-speed rotorcraft, and recommendations for future work.

Chapter 2 | Flight Dynamics Models

This chapter provides an overview of the generic lift offset coaxial compound helicopter (herein referred to as coaxial-pusher) and tiltrotor flight dynamics models. First, detailed descriptions of the physical aircraft parameters used in the comprehensive rotorcraft simulation code HeliUM are provided. Next, variations in trim controls and attitudes with airspeed are provided. This is followed by a discussion of the linear models extracted from HeliUM, including description of key stability and control derivatives, the rotor modes retained in the linear models, and the eigenvalues of the linear models. Subsequently, the model stitching architecture used to simulate the models in real-time is discussed. Next, the control allocation method used to distribute the desired moments to the multiple control effectors of each aircraft is shown. Finally, primary on-axis frequency responses are provided for each aircraft at a range of airspeeds, followed by a discussion.

2.1 Flight Dynamics Modeling

2.1.1 Overview

The flight dynamics models for both configurations were developed using HeliUM [68, 69]. HeliUM uses a finite-element approach to model flexible rotor blades with coupled nonlinear flap/lag/torsion dynamics to capture structural, inertial, and aerodynamic loads along each blade segment. Blade, wing, and fuselage aerodynamics come from nonlinear lookup tables, and the rotor airwakes are modeled using a dynamic inflow model [70]. A multi-body like modeling approach is used to build the aircraft configuration from its independent components (e.g., fuselage, wing, nacelle, etc.) [68], which allows modeling of arbitrary aircraft configurations with multiple rotors. Extensive validation of HeliUM (including with flight data) covering many rotorcraft configurations has previously been done. HeliUM coaxial-pusher modeling has been validated using Sikorsky X2 Technology Demonstrator flight data and the Sikorsky X2 GenHel model [71]. HeliUM tiltrotor modeling has been validated against XV-15 flight data, the Bell GTRSIM XV-15 model, and a CAMRAD II [72] (comprehensive

analysis tool) model of the Large Civil Tiltrotor (LCTR) [68]. The excellent results of all these validation efforts give confidence in the modeling fidelity and approach used here for the generic aircraft models.

To be able to realistically simulate these aircraft, power required was included in the models. Power requirements in HeliUM are obtained from summing power requirements of the individual main rotors and any tail rotors, with an additional 5% added to account for transmission and accessory losses.

2.1.2 Coaxial-Pusher

2.1.2.1 Description

The coaxial-pusher configuration was derived from a previous rotorcraft sizing trade-off study [73], which gives the overall dimensional and weight characteristics as well as key rotor and aircraft aerodynamic properties. The only configuration change made to the original design of Ref. 73, was to relocate the pusher propeller to be vertically aligned with the aircraft center-of-gravity (CG). This change was made to be more consistent with the X2 and SB>1 configurations. The coaxial-pusher aircraft is shown in Figure 2.1.



Figure 2.1. Generic coaxial-pusher rendering (reproduced from [74]).

Public domain data of the Sikorsky XH-59A Advancing Blade Concept (ABC) [4] and Sikorsky X2 Technology Demonstrator were used to develop the mass and stiffness properties for the generic coaxial-pusher hingeless rotor system. Physical blade properties of the XH-59A were tuned such that the first non-dimensional flap and lag mode frequencies are similar to those of the X2, a state-of-the-art aircraft that represents the latest technology advancements in rotor design. Blade twist and chord properties come from the rotor aerodynamic optimization results in Ref. 73. The same blade airfoils were used as the XH-59A ABC.

The inflow model for this configuration immerses each rotor in the wake of the other rotor (using a simple “dynamic climb” approximation) in hover and slow-speed flight using influence coefficients as defined in Ref. [75]. The blending of the inflow diminishes as speed increases, and the two rotors are treated independently at airspeed above $V_{\text{TAS}} = 120$ kts. Note that the biplane effect between the two rotors, which reduces the induced power of coaxial rotors at moderate airspeed [76, 77] is ignored here for simplicity due to the emphasis on flight dynamics and not performance for this modeling effort.

Aerodynamics for the H-tail come from open-source lookup tables of XV-15 vertical and horizontal stabilizers obtained from wind tunnel tests that include effects of elevator and rudder deflections [78].

Finally, the pusher propeller is treated as a momentum theory Bailey rotor [79]. It generates a thrust and torque which are transmitted to the aircraft CG. This type of modeling for the pusher propeller has been validated with X2 and SB>1 models and provides a high level of flight-dynamics fidelity.

A set of key aircraft properties is given in Table 2.1. The aircraft mission representative gross weight is 35,200 lbs. Maximum speed is limited to $V_{\text{TAS}} = 240$ kts using notional engines that provide a total of 8,000 hp. The rotors are four-bladed with a radius of $R = 30.55$ ft each. The vertical separation between the rotors is 7% of the rotor diameter, or 4.28 ft. The blades of both rotors are preconed by $\beta_p = 2.0$ deg at the hub to reduce steady flapwise bending stresses.

Table 2.1. Coaxial-Pusher Configuration Data

<i>Aircraft Data</i>	
Gross Weight	35,200 lbs
Max Continuous Power (SL)	8,000 hp
V_{MCP} (KTAS, 6k95)	240 kts
<i>Rotor Data</i>	
Radius	30.55 feet
Number of Blades/Rotor	4
Rotational Speed	23.7 – 19.0 rad/sec
Vertical Separation (S/D)	7%
Precone	2 deg
Twist	–9 deg
<i>Pusher Data</i>	
Radius	6.6 feet
Number of Blades	6
Rotational Speed	136 rad/sec

2.1.2.2 Trim

To have a MATLAB Simulink simulation model that is capable of faster-than-real-time execution speeds, linear models and trim data were extracted from HeliUM and used to develop a stitched model of the coaxial-pusher configuration. The stitched model will be described in more detail in Sec. 2.3, but the coaxial-pusher trim data and linear models are presented here.

To trim the nonlinear HeliUM model, a set of ganged phased controls was used for the coaxial rotors. First, the swashplate cyclic controls (lateral cyclic pitch θ_{1c} and longitudinal cyclic pitch θ_{1s}) for each rotor must be phased since the phase lag between cyclic pitch and flap is significantly less than 90 deg for these very stiff rotors. Figure 2.2 shows the control phasing notation for the upper and lower rotors, where θ'_{1c} and θ'_{1s} are the phased lateral and longitudinal cyclic pitch inputs respectively, and Δ_{SP} is the phase (or mixing) angle. In general, the phase angle Δ_{SP} can be different for lateral cyclic ($\Delta_{SP_{1c}}$) and for longitudinal cyclic ($\Delta_{SP_{1s}}$), as well as different between the upper and lower rotors (denoted by subscripts U and L). For the upper rotor (Figure 2.2, top), the blade pitch θ_U is given by:

$$\theta_U(\psi_U) = \theta_{0U} + \theta'_{1cU} \cos(\psi_U - \Delta_{SP_{1cU}}) + \theta'_{1sU} \sin(\psi_U - \Delta_{SP_{1sU}}) \quad (2.1)$$

while for the lower rotor (Figure 2.2, bottom), the blade pitch θ_L is given by:

$$\theta_L(\psi_L) = \theta_{0L} + \theta'_{1cL} \cos(\psi_L - \Delta_{SP_{1cL}}) + \theta'_{1sL} \sin(\psi_L - \Delta_{SP_{1sL}}) \quad (2.2)$$

Typical values of phase angle for an articulated rotor are $\Delta_{SP} = 8-12$ deg [80]. The optimal phase angle value depending on airspeed. For these very stiff coaxial-pusher rotors, a single value of $\Delta_{SP} = 70$ deg was chosen as a compromise between hover and forward flight, and used for both upper and lower rotor lateral and longitudinal cyclic. This value of phase angle is similar to the values used for the XH-59A ($\Delta_{SP} = 50-60$ deg) [4].

The phased controls for the two rotors were then ganged into symmetric (θ_0 , θ'_{1c} , and θ'_{1s}) and differential ($\Delta\theta_0$, $\Delta\theta'_{1c}$, and $\Delta\theta'_{1s}$) rotor controls:

$$\theta_0 = \frac{\theta_{0U} + \theta_{0L}}{2} \quad (2.3)$$

$$\theta'_{1c} = \frac{\theta'_{1cU} - \theta'_{1cL}}{2} \quad (2.4)$$

$$\theta'_{1s} = \frac{\theta'_{1sU} + \theta'_{1sL}}{2} \quad (2.5)$$

$$\Delta\theta_0 = \frac{\theta_{0U} - \theta_{0L}}{2} \quad (2.6)$$

$$\Delta\theta'_{1c} = \frac{\theta'_{1cU} + \theta'_{1cL}}{2} \quad (2.7)$$

$$\Delta\theta'_{1s} = \frac{\theta'_{1sU} - \theta'_{1sL}}{2} \quad (2.8)$$

with the symmetric controls (θ_0 , θ'_{1c} , and θ'_{1s}) used to trim the HeliUM model.

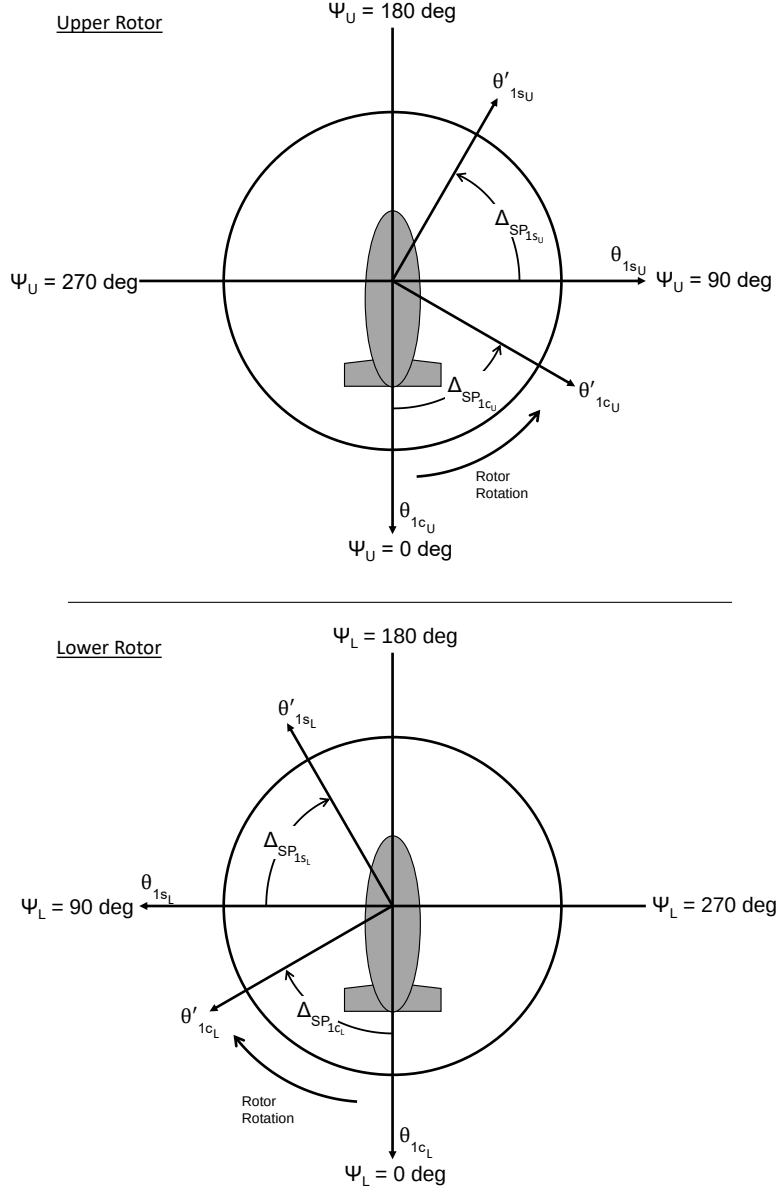


Figure 2.2. Rotor control phasing Δ_{SP} definition for the upper rotor (top) and lower rotor (bottom) (coaxial-pusher).

Note that the single, fixed phase angle $\Delta_{SP} = 70$ described here is used only to trim the nonlinear model and to then extract linearized models. Once the linearized models were generated, a pseudo inverse control allocation scheme was developed for

use in a control system, which led to different (optimal) phasing of the controls, as described in Sec. 2.2.

The aircraft is trimmed using both the symmetric rotor controls described above and the pusher propeller collective ($\theta_{0_{PP}}$). The pusher propeller provides a redundant control degree of freedom in the longitudinal axis, which allows specific trim attitude targets to be reached. The nominal trim condition is ship level (i.e., $\theta = 0$ deg) at all airspeeds.

Aerosurface controls (elevator δ_e and rudder δ_r) as well as pusher propeller mono-cyclic $\theta_{1_{CPP}}$ (which is used to generate yawing moments with the pusher propeller) are used for control but not to trim the aircraft.

Figure 2.3 shows trim values as a function of airspeed for standard sea level conditions. Rotor speed is reduced linearly from NR = 100% ($\Omega = 23.7$ rad/sec) to NR = 80% ($\Omega = 19.0$ rad/sec) between $V_{TAS} = 160 - 220$ kts, and remains at NR = 80% above $V_{TAS} = 220$ kts to avoid Mach effects on each rotor's advancing blade. A 20% reduction in rotor speed was similarly done for the Sikorsky X2 technology demonstrator aircraft [81].

Trim symmetric collective θ_0 has its maximum value at hover and decreases as airspeed increases. This is typical of conventional helicopters [2], where trim collective approaches its minimum value at the minimum drag airspeed and then begins to increase again with increasing airspeed as the rotor is required to provide more propulsive power. However, for the coaxial-pusher configuration, trim symmetric collective continues to decrease with airspeed (increasing slightly between $V = 160 - 220$ kts as rotor speed is decreased), since propulsive power is provided by the pusher propeller.

By constraint, elevator δ_e and pitch attitude θ are fixed at $\delta_e = 0$ and $\theta = 0$ for all airspeeds, and pitching moment is trimmed using symmetric longitudinal cyclic θ'_{1s} .

Figure 2.4 shows the trim power required P_{req} for the coaxial-pusher as a function of airspeed at standard sea level (SL) conditions and at an altitude of $h = 6,000$ ft and an outside air temperature (OAT) of $T = 95$ F (6k95). The figure also shows the power available P_{avail} provided by a pair of notional engines at sea level and 6k95. Power available for the notional engines is scaled with altitude and OAT using a simplified power available model based on air density ρ [2]:

$$P_{avail} = P_{avail_{SL}} \frac{\rho}{\rho_{SL}} \quad (2.9)$$

The notional engines were sized to provide sufficient power to hover at 6k95 and reach speeds of roughly $V_{TAS} = 240$ kts, resulting in $P_{avail_{SL}} = 4,000$ hp per engine.

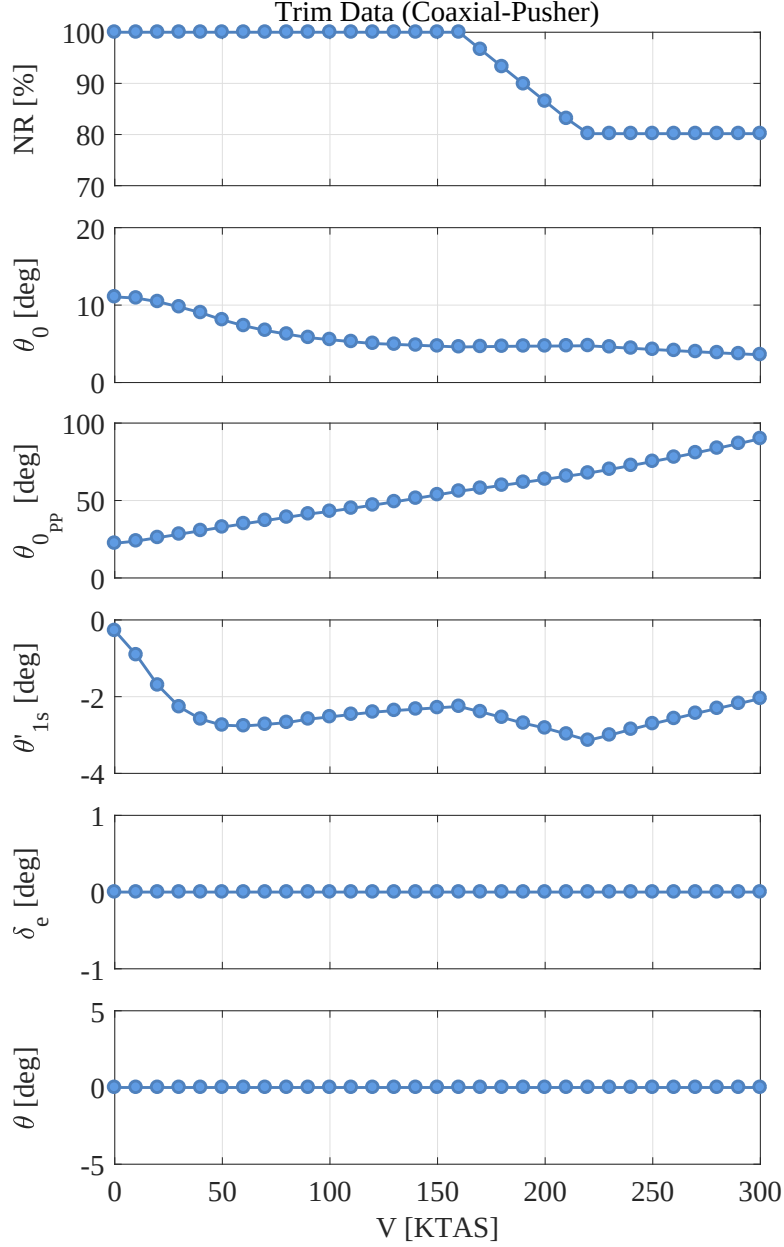


Figure 2.3. Trim rotor speed (NR), symmetric collective pitch (θ_0), pusher propeller collective pitch ($\theta_{0_{PP}}$), symmetric longitudinal cyclic pitch (θ'_{1s}), elevator (δ_e), and pitch attitude (θ) as a function of true airspeed (coaxial-pusher, sea level).

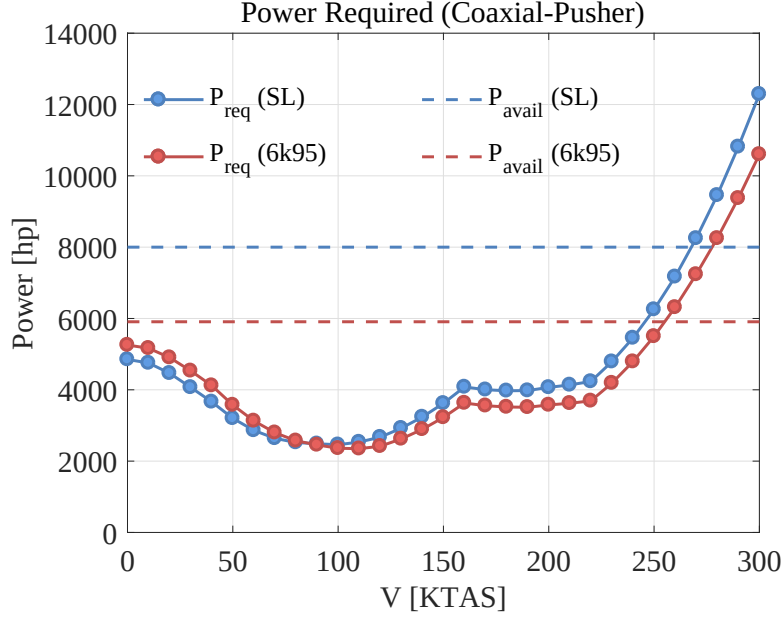


Figure 2.4. Power required and available at sea level (SL) and 6,000 ft 95 F (6k95) (coaxial-pusher).

An additional parameter for the coaxial-pusher configuration is lift offset (LOS) of the rotors, defined for a single rotor as [76]:

$$\text{LOS} = \frac{M_x}{TR} \quad (2.10)$$

where:

- M_x is the rotor rolling moment
- T is the rotor thrust, and
- R is the rotor radius.

Figure 2.5 shows the average lift offset of the upper and lower rotors as a function of airspeed. Lift offset is $\text{LOS} = 0\%$ in hover and increases with airspeed, reaching a maximum value of about $\text{LOS} = 20\%$ at $V_{\text{TAS}} = 210$ kts. At greater airspeeds ($V_{\text{TAS}} > 210$ kts), lift offset reduces with increased airspeed. This is because as trim collective θ_0 is reduced with increasing airspeed (Figure 2.3), the outer portion of the advancing blade eventually reaches negative angles of attack, and the lift distribution along the blade moves inboard.

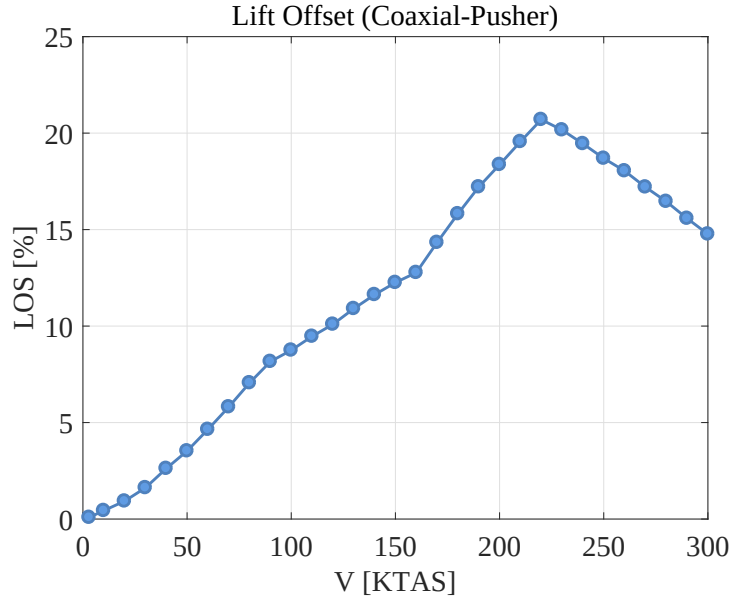


Figure 2.5. Lift offset (LOS) as a function of true airspeed (coaxial-pusher, sea level).

2.1.2.3 Linear Models

Once the HeliUM model was trimmed at specific airspeeds, it was linearized using a numerical linearization algorithm. Linear state-space models of the form:

$$\begin{aligned}\dot{\mathbf{x}} &= \mathbf{A}\mathbf{x} + \mathbf{B}\mathbf{u} \\ \mathbf{y} &= \mathbf{C}\mathbf{x} + \mathbf{D}\mathbf{u}\end{aligned}\tag{2.11}$$

were extracted at airspeeds ranging from hover to $V_{\text{TAS}} = 300$ kts in increments of $V_{\text{TAS}} = 10$ kts, and at altitudes from sea level to 25,000 ft in increments of 5,000 ft. For brevity, only data at sea level is presented here.

The linear models contain 48 states ($\mathbf{x}$):

- Nine fuselage states (9),
- Four (one collective, two cyclic, and one reactionless) second-order rotor states for each of the two blade modes retained in the linear model per rotor (32),
- Three (average, cosine, and sine) inflow states per rotor (6),
- One inflow state for the pusher propeller (1).

And 10 inputs ($\mathbf{u}$):

- Symmetric lateral cyclic (θ'_{1c})
- Symmetric longitudinal cyclic (θ'_{1s})
- Symmetric collective (θ_0)

- Differential collective ($\Delta\theta_0$)
- Differential lateral cyclic ($\Delta\theta'_{1c}$)
- Differential longitudinal cyclic ($\Delta\theta'_{1s}$)
- Pusher propeller collective (θ_{0PP})
- Pusher propeller monocyclic (θ_{1cPP})
- Elevator (δ_e)
- Rudder (δ_r)

In addition to all of the states being outputs, power required and tip clearance (discussed later in Sec. 2.1.2.4) outputs were also included.

The remainder of this section will discuss trends in the linear models as a function of airspeed by looking at the stability and control derivatives, blade modes, and eigenvalues.

To study the overall trends in the fuselage stability and control derivatives with respect to the inputs and fuselage states, the bare-airframe model (Eq. 2.11) was reduced to the rigid-body six degrees-of-freedom (6 DOF) quasi-steady dynamics:

$$\begin{aligned}\dot{\mathbf{x}}_{\text{RB}} &= \hat{\mathbf{A}}_{\text{RB}}\mathbf{x}_{\text{RB}} + \hat{\mathbf{B}}_{\text{RB}}\mathbf{u} \\ \mathbf{y} &= \hat{\mathbf{C}}_{\text{RB}}\mathbf{x}_{\text{RB}} + \hat{\mathbf{D}}_{\text{RB}}\mathbf{u}\end{aligned}\tag{2.12}$$

using singular perturbation theory [82]. This is done by first dividing the model (Eq. 2.11) into the rigid-body states $\mathbf{x}_{\text{RB}}$ and higher-order (rotor and inflow) states $\mathbf{x}_{\text{HO}}$:

$$\begin{aligned}\begin{bmatrix} \dot{\mathbf{x}}_{\text{RB}} \\ \dot{\mathbf{x}}_{\text{HO}} \end{bmatrix} &= \begin{bmatrix} \mathbf{A}_{\text{RB}} & \mathbf{A}_{\text{RH}} \\ \mathbf{A}_{\text{HR}} & \mathbf{A}_{\text{HO}} \end{bmatrix} \begin{bmatrix} \mathbf{x}_{\text{RB}} \\ \mathbf{x}_{\text{HO}} \end{bmatrix} + \begin{bmatrix} \mathbf{B}_{\text{RB}} \\ \mathbf{B}_{\text{HO}} \end{bmatrix} \mathbf{u} \\ \mathbf{y} &= \begin{bmatrix} \mathbf{C}_{\text{RB}} & \mathbf{C}_{\text{HO}} \end{bmatrix} \begin{bmatrix} \mathbf{x}_{\text{RB}} \\ \mathbf{x}_{\text{HO}} \end{bmatrix} + \mathbf{D}\mathbf{u}\end{aligned}\tag{2.13}$$

Next, all higher-order states are assumed to reach their quasi-steady values more quickly than the rigid-body states, such that in the time scale of interest:

$$\dot{\mathbf{x}}_{\text{HO}} = 0\tag{2.14}$$

and the matrices of the quasi-steady 6 DOF (Eq. 2.12) can be solved for as:

$$\begin{aligned}\hat{\mathbf{A}}_{\text{RB}} &= \mathbf{A}_{\text{RB}} - \mathbf{A}_{\text{RH}}\mathbf{A}_{\text{HO}}^{-1}\mathbf{A}_{\text{HR}} \\ \hat{\mathbf{B}}_{\text{RB}} &= \mathbf{B}_{\text{RB}} - \mathbf{A}_{\text{RH}}\mathbf{A}_{\text{HO}}^{-1}\mathbf{B}_{\text{HO}} \\ \hat{\mathbf{C}}_{\text{RB}} &= \mathbf{C}_{\text{RB}} - \mathbf{C}_{\text{HO}}\mathbf{A}_{\text{HO}}^{-1}\mathbf{A}_{\text{HR}} \\ \hat{\mathbf{D}}_{\text{RB}} &= \mathbf{D} - \mathbf{C}_{\text{HO}}\mathbf{A}_{\text{HO}}^{-1}\mathbf{B}_{\text{HO}}\end{aligned}\tag{2.15}$$

Figure 2.6 shows the key quasi-steady rigid-body stability derivatives of the coaxial-

pusher as a function of airspeed. Figures 2.7 and 2.8 show the key lateral/directional and longitudinal/heave quasi-steady rigid-body control derivatives, respectively, as a function of airspeed.

2.1.2.3.1 Stability Derivatives

The speed stability derivative M_u (Figure 2.6, first row, first column) starts at a positive value in hover due to rotor blowback for increased forward speed. However, this affect is diminished as airspeed increases, and the value of M_u approaches zero above $V_{TAS} = 120$ kts, which is typical of fixed-wing aircraft.

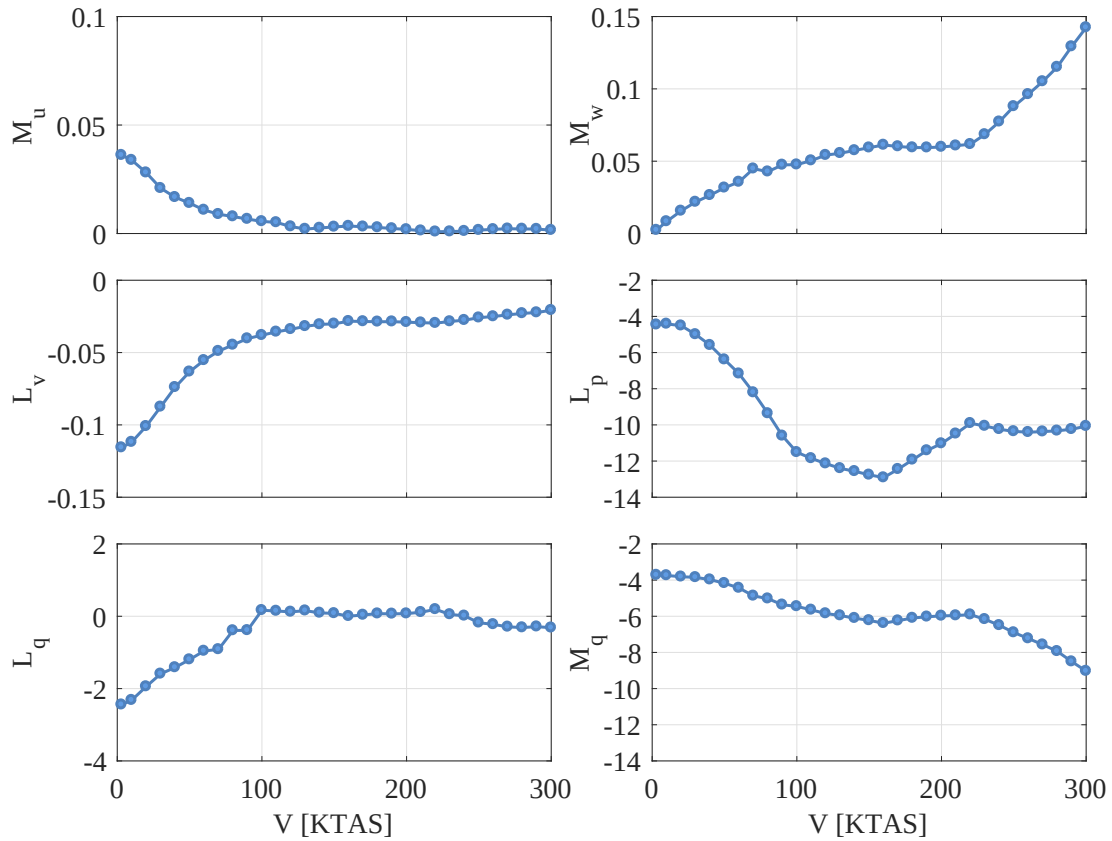


Figure 2.6. Rigid body stability derivatives as a function of true airspeed (coaxial-pusher, sea level).

The dihedral effect derivative L_v (Figure 2.6, second row, first column) is similar to the M_u derivative, although negative in sign due to the body axis system sign convention. It begins at a large value in hover, and decreases in magnitude as airspeed increases. The negative value of L_v , as well as the positive value of N_v (not shown),

has a stabilizing effect on lateral/directional static stability [80].

The longitudinal static stability derivative M_w (Figure 2.6, first row, second column) starts at $M_w \approx 0$ for hover, and becomes larger (more positive) as airspeed increases. Since $M_w > 0$, this aircraft is statically unstable, and the instability becomes more pronounced as airspeed increases, as will be seen in the eigenvalue plots shown later.

The damping derivatives L_p and M_q (Figure 2.6, second row, second column, and third row, second column, respectively) are both negative and increase in magnitude with increasing airspeed. Between $V_{TAS} = 160 - 220$ kts, both derivatives decrease linearly as rotor speed is decreased.

Roll moment due to pitch, given by the roll coupling derivative L_q (Figure 2.6, third row, first column) is large around hover and decreases with increasing airspeed. Above $V_{TAS} = 100$ kts, the coupling is significantly reduced and L_q is small, common for helicopters in forward flight (e.g., SH-2G identification results in Ref. [83]).

2.1.2.3.2 Lateral/Directional Control Derivatives

The primary roll control derivative is rolling moment due to symmetric lateral cyclic $L_{\theta'_{1c}}$ (Figure 2.7, first row, first column). Symmetric lateral cyclic θ'_{1c} becomes more effective at generating rolling moment as airspeed increases above $V_{TAS} = 40$ kts. Between $V_{TAS} = 160 - 220$ kts $L_{\theta'_{1c}}$ decreases due to decreasing rotor speed.

Large rolling moments are also generated by differential collective at higher airspeeds as seen by the $L_{\Delta\theta_0}$ derivative (Figure 2.7, first row, second column). However, this is an undesirable coupling effect as differential collective is intended for yaw control.

The yawing moment generated by differential collective is also a function of airspeed as seen by the $N_{\Delta\theta_0}$ derivative (Figure 2.7, third row, first column), with its largest value in hover. The $N_{\Delta\theta_0}$ derivative approaches a value of zero for airspeeds greater than $V_{TAS} = 100$ kts. This is because the amount of yawing moment generated through differential collective is proportional to the rotor torque, which for the coaxial-pusher is highest at hover and decreases as airspeed increases (proportional to the trim symmetric collective θ_0 curve in Figure 2.3).

Side force and yawing moment due to rudder both increase proportional to dynamic pressure or V^2 with increasing airspeed as seen from the Y_{δ_r} (not shown) and N_{δ_r} (Figure 2.7, fourth row, second column) derivatives. The rudder is only effective at generating yawing moments at high airspeeds. This leaves a gap in the mid-airspeed range where yawing moment due to differential collective is reduced and the rudder is not effective enough.

Differential longitudinal cyclic $\Delta\theta'_{1s}$ also generates a yawing moment by producing differential torque on the two rotors. It is effective at speeds greater than $V_{TAS} = 50$ kts as seen by the $N_{\Delta\theta'_{1s}}$ derivative (Figure 2.7, third row, second column). However, it also generates an off-axis rolling moment at low ($V_{TAS} < 50$ kts) and high ($V_{TAS} > 150$ kts) airspeeds, as seen by the $L_{\Delta\theta'_{1s}}$ derivative (Figure 2.7, second row, first column).

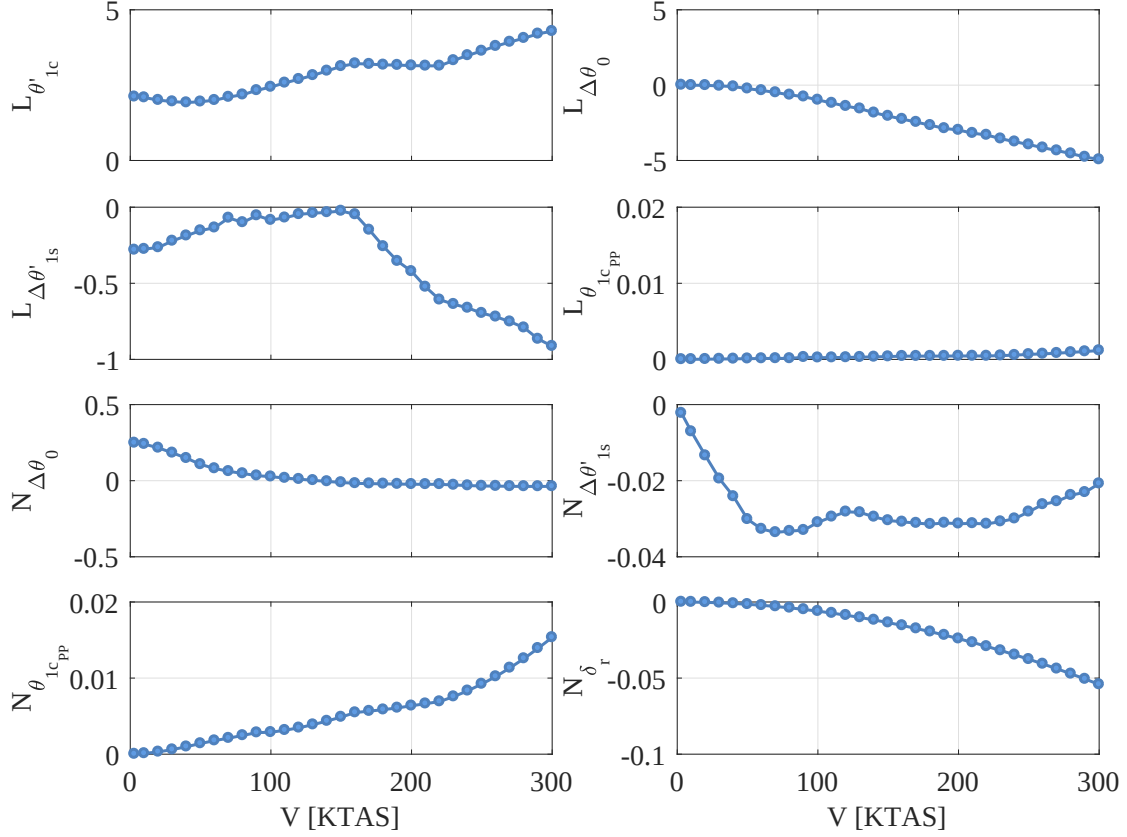


Figure 2.7. Rigid body lateral/directional control derivatives as a function of true airspeed (coaxial-pusher, sea level).

To meet yaw control power and roll-sideslip coupling requirements, a fourth yaw control was introduced—pusher propeller monocyclic θ_{1cPP} . The yaw moment generated by the pusher propeller monocyclic is proportional to the thrust being generated by the pusher propeller and therefore increases with increasing airspeed as seen by the $N_{\theta_{1cPP}}$ derivative (Figure 2.7, fourth row, first column). There is also near zero rolling moment generated by pusher propeller monocyclic as seen by the $L_{\theta_{1cPP}}$ derivative (Figure 2.7, second row, second column), since the pusher propeller is vertically aligned with the CG.

2.1.2.3.3 Longitudinal/Heave Control Derivatives

The primary pitch control derivative is pitching moment due to symmetric longitudinal cyclic $M_{\theta'_{1s}}$ (Figure 2.8, third row, second column). This derivative value has a similar variation with airspeed as rolling moment due to symmetric lateral cyclic $L_{\theta'_{1c}}$,

however $M_{\theta'_{1s}}$ is smaller in magnitude than $L_{\theta'_{1c}}$ at all airspeeds due to the aircraft's larger pitch inertia I_{yy} than roll inertia I_{xx} ($I_{yy}/I_{xx} \approx 8$).

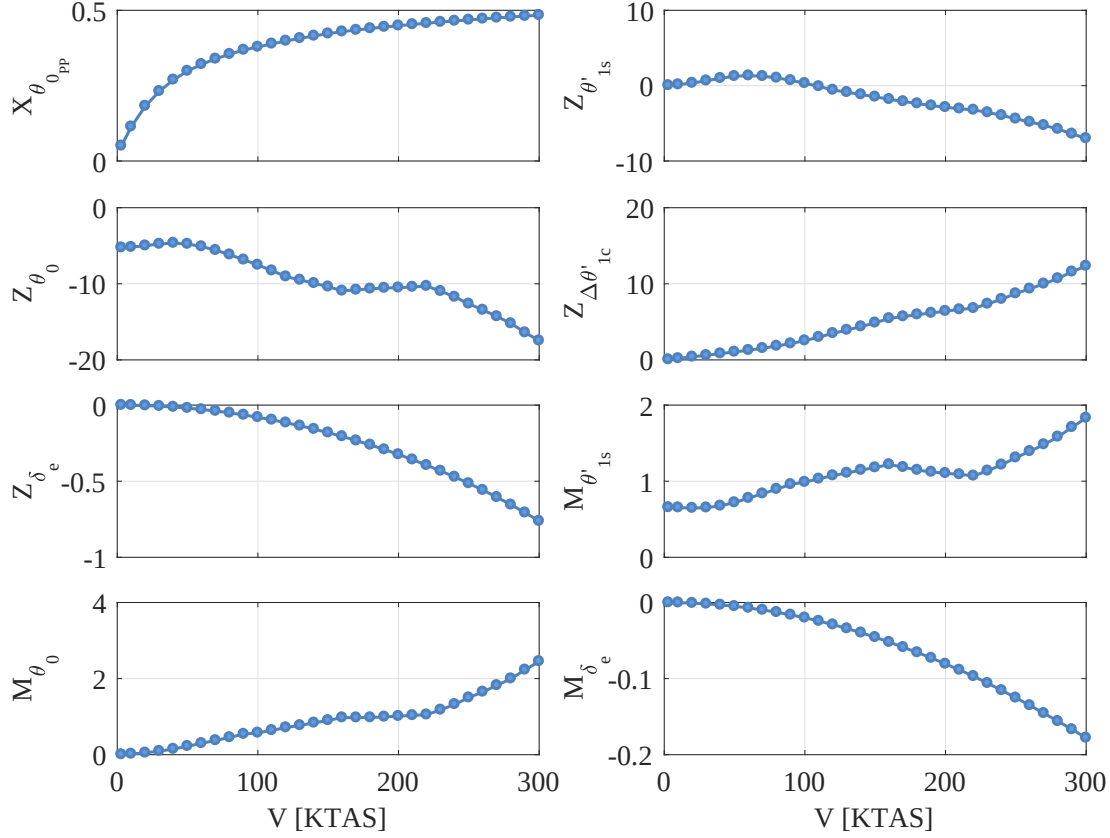


Figure 2.8. Rigid body longitudinal/heave control derivatives as a function of true airspeed (coaxial-pusher, sea level).

Symmetric longitudinal cyclic θ'_{1s} also generates vertical force, as seen by the $Z_{\theta'_{1s}}$ derivative (Figure 2.8, first row, second column). This derivative is $Z_{\theta'_{1s}} = 0$ in hover and increases in magnitude above $V_{TAS} = 100$ kts. The primary control for vertical force is symmetric collective θ_0 as seen by the Z_{θ_0} derivative (Figure 2.8, second row, first column). Symmetric collective also generates a pitching moment (M_{θ_0}), with increasing magnitude as airspeed increases. This can potentially cause undesirable pitch-heave coupling at high speed if symmetric collective is used for control.

The vertical force generated by differential lateral cyclic $\Delta\theta'_{1c}$ is a strong function of airspeed, as seen by the $Z_{\Delta\theta'_{1c}}$ derivative (Figure 2.8, second row, second column). Differential lateral cyclic $\Delta\theta'_{1c}$ controls lift offset, and as it is increased, the lift vector on each rotor is moved outboard on the advancing blade and increases in magnitude. Due to the counter rotating rotors, the total rolling moment generated by differential

lateral cyclic $L_{\Delta\theta'_{1c}}$ (not shown) is nearly zero. However, this also increases the inter-hub mast moment.

Axial force is provided by the pusher propeller collective θ_{0PP} as seen by the $X_{\theta_{0PP}}$ derivative (Figure 2.8, first row, first column). The amount of force generated per degree of pusher propeller collective is small at low airspeeds and increases with airspeed. Pitching moment due to pusher propeller collective is nearly zero due to the pusher propeller being vertically aligned with the CG. However, rolling moment due to pusher propeller collective is significant due to the increased torque on the propeller with increased thrust.

Normal force and pitching moment due to elevator both increase proportional to dynamic pressure or V^2 with increasing airspeed as seen from the Z_{δ_e} (Figure 2.8, third row, first column) and M_{δ_e} (Figure 2.8, fourth row, second column) derivatives .

2.1.2.3.4 Blade Modes

The first two rotor blade modes (coupled flap-lag) are retained in the linearized coaxial-pusher models. Figure 2.9 shows the natural frequencies of the two modes in the rotating frame as a function of rotor speed, with the primary motion (first flap bending 1F or first lag bending 1L) labeled. Note that the mode frequencies shown here are for constant collective θ_0 value. At $\Omega/\Omega_0 = 1$, the lag mode frequency is $\nu_\zeta = 1.33/\text{rev}$ (stiff in-plane), while the flap mode frequency is $\nu_\beta = 1.49/\text{rev}$. For the range of rotor speed used $\Omega/\Omega_0 = 0.8 - 1$, the two blade modes are away from any integer rotor harmonics (e.g., 1/rev, 2/rev, etc.).

Figure 2.10 shows the flap and lag blade deflections contributing to each of the two blade modes for the nominal rotor speed (Ω_0) and reduced rotor speed ($0.8\Omega_0$). At the nominal rotor speed Ω_0 , the lower frequency Mode 1 is primarily composed of the first lag mode with some contribution of the first flap mode (Figure 2.10, upper plot, solid lines). The higher frequency Mode 2 at the nominal rotor speed Ω_0 is primarily composed of first flap mode with some contribution of the first lag mode (Figure 2.10, lower plot, solid lines).

At the reduced rotor speed $0.8\Omega_0$, the flap and lag modes become nearly fully coupled, with Mode 1 being anti-symmetric flap-lag motion (Figure 2.10, upper plot, dashed lines) and Mode 2 being symmetric flap-lag motion (Figure 2.10, lower plot, dashed lines). This coupling of the modes as the rotor is slowed down can be seen from the convergence of the two mode lines in Figure 2.9 at around $\Omega/\Omega_0 = 0.8$. As the rotor is slowed down further ($\Omega/\Omega_0 < 0.8$), the modes decouple and switch dominant motion. This behavior was similarly seen in the XH-59A [4].

The blade modes are stable for all airspeeds as will be shown in the eigenvalue analysis next.

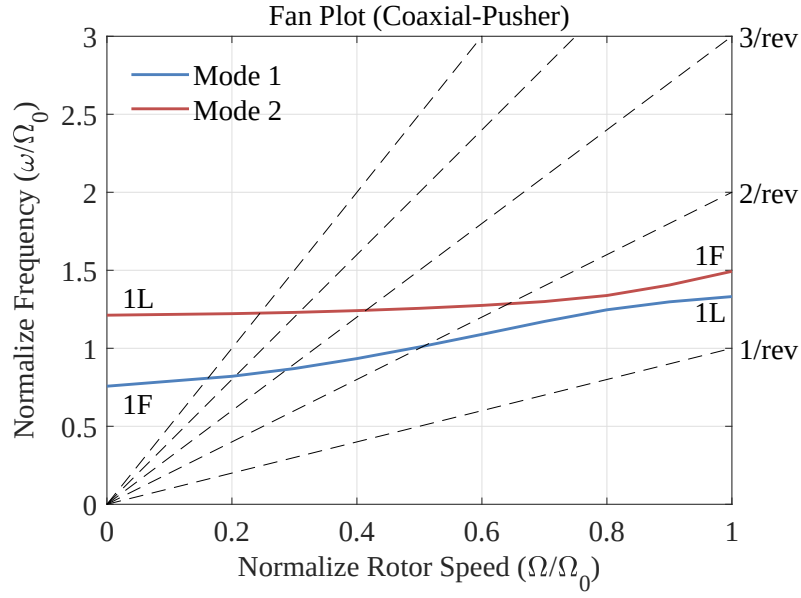


Figure 2.9. Blade mode fan diagram (coaxial-pusher).

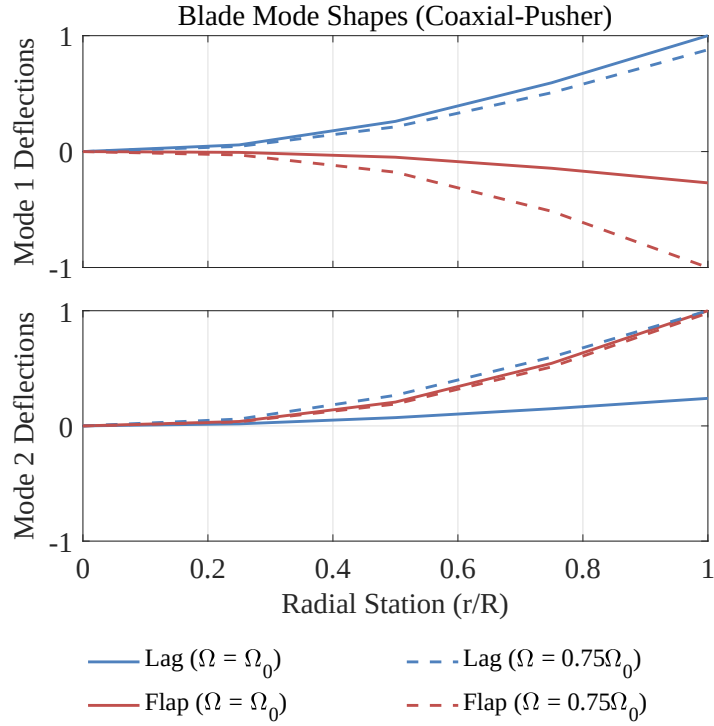


Figure 2.10. Flap and lag deflections for blade Modes 1 and 2 (coaxial-pusher).

2.1.2.3.5 Eigenvalues

Figures 2.11 and 2.12 show a zoomed-out and zoomed-in view, respectively, of the eigenvalues of the full order (48-state) linear state-space models as a function of airspeed. The collective, progressive, regressive, and reactionless rotor modes can be seen in Figure 2.11. These rotor modes correspond to blade Mode 1 and Mode 2, and are labeled with their dominant motion (flap and lag) as discussed in the previous section. Note that there are two of each mode (one for each of the two rotors), for a total of 16 modes.

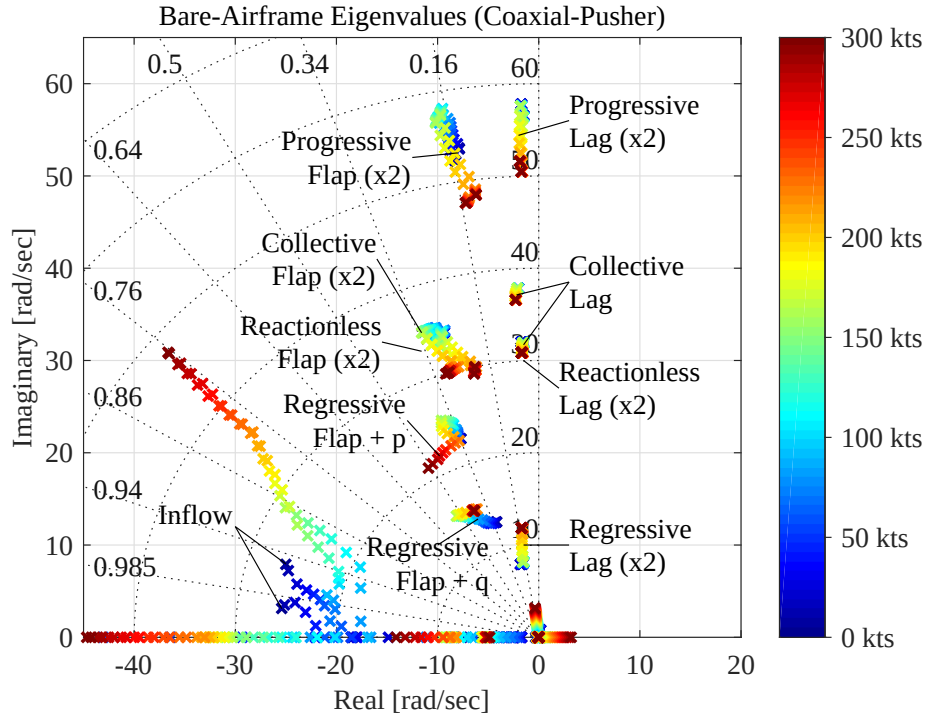


Figure 2.11. Eigenvalues as a function of true airspeed (coaxial-pusher, sea level).

The rotor modes are all stable, with the lag mode being more lightly damped than the flap mode, as expected. Furthermore, the rotor mode frequencies are a function of rotor speed, mainly varying between $V_{TAS} = 160 - 220$ kts where the rotor is slowed down. The two (low-frequency) regressive flap modes couple with the fuselage roll and pitch motion.

In addition, the six inflow modes can be seen in Figure 2.11, which are a strong functions of airspeed with natural frequencies roughly equal to the rotor speed and damping ratios between $\zeta = 0.76 - 1.0$ throughout.

Figure 2.12 shows the fuselage eigenvalues of the coaxial-pusher as a function of airspeed. At hover, there are low-frequency unstable complex modes (hovering cubic) in both the lateral (labeled Dutch Roll in Figure 2.12) and longitudinal (labeled

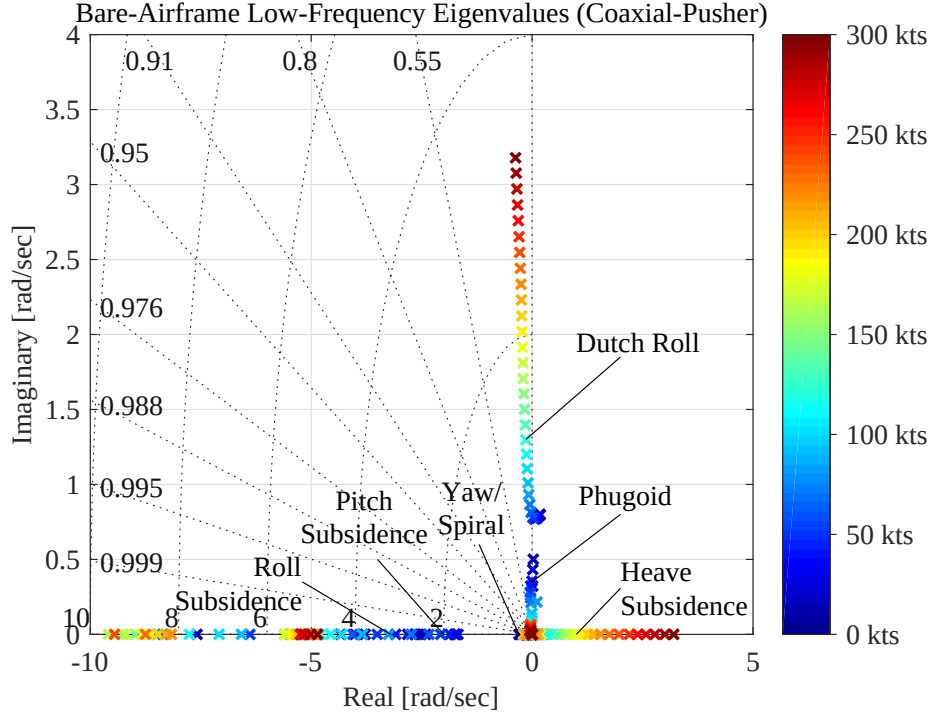


Figure 2.12. Low-frequency eigenvalues as a function of true airspeed (coaxial-pusher, sea level).

Phugoid in Figure 2.12) axes. In the lateral axis, this mode stabilizes and increases in frequency as airspeed increases, becoming the lightly damped Dutch roll mode. In the longitudinal axis, the complex pair of the phugoid mode is also unstable at hover and low speed, but reduces in frequency and becomes stable as airspeed increases.

A real roll mode $1/T_r$ (labeled Roll Subsidence in Figure 2.12) is present which increases in frequency with increasing airspeed, from about $1/T_r = 5$ rad/sec at hover to $1/T_r = 10$ rad/sec at $V_{TAS} = 300$ kts. These values of roll mode inverse time constant correspond to the values of the L_p derivative seen in Figure 2.6.

Real pitch and heave modes (labeled Pitch and Heave Subsidence in Figure 2.12) are present which both increase in frequency with increasing airspeed. At hover/low-speed, they are both stable and the pitch and heave motions are decoupled. As airspeed increases, the pitch and heave motions of both modes couple. In addition, the heave subsidence mode moves into the right half-plane (becomes unstable), as expected from the positive values of M_w seen in Figure 2.6.

Finally, a low frequency real yaw/spiral mode can be in Figure 2.12.

2.1.2.4 Tip Clearance Output

Ensuring sufficient separation between the two rotors of the coaxial-pusher configuration during dynamic maneuvers is important. Therefore, tip clearance was calculated

from the trim data and linear models. This was done by first performing a state transformation on the trim data and linear models to transform states corresponding to coupled flap/lag blade Modes 1 and 2 into states corresponding to decoupled flap and lag modes. Then, the total (trim plus perturbation) flap states (β_0 , β_{1c} , and β_{1s}) and rotor radius R were used to determine the height z of the tip path plane of each rotor:

$$z_U(\psi) = R(\beta_{0U} + \beta_{1cU} \cos \psi + \beta_{1sU} \sin \psi) \quad (2.16)$$

$$z_L(\psi) = -R(\beta_{0L} + \beta_{1cL} \cos \psi - \beta_{1sL} \sin \psi) \quad (2.17)$$

Finally, the minimum vertical separation between the two tip path planes around the azimuth was determined as:

$$\Delta z_{\min} = \min_{\psi} [z_U(\psi) - z_L(\psi) + S] \quad (2.18)$$

where S is the vertical separation between the rotors, equal to 7% of the rotor diameter ($S = 4.28$ ft), and included as an output in the quasi-Linear-Parameter-Varying (qLPV) stitched simulation model of the coaxial-pusher (described in Sec. 2.3).

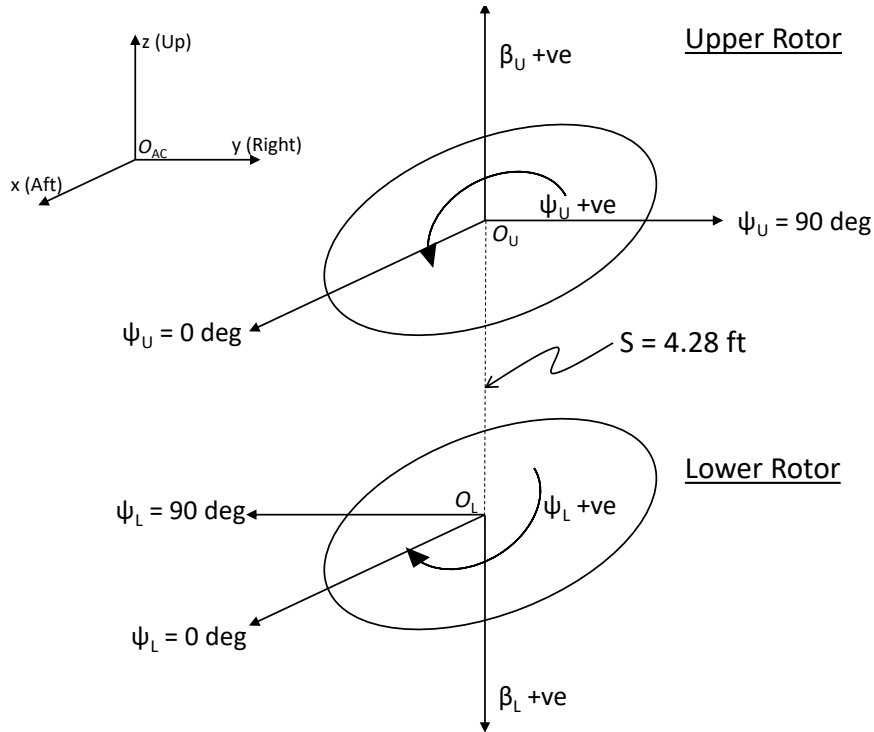


Figure 2.13. Upper and lower rotor flapping sign convention (coaxial-pusher).

Figure 2.14 shows the trim minimum rotor separation $\Delta z_{\min}$ as a function of airspeed. Trim minimum rotor separation is a function of lift offset (shown in

Figure 2.5), as well as rotor speed NR and blade stiffness properties. The trend seen here matches well the tip clearance versus airspeed trends seen for the X2 in flight [84].

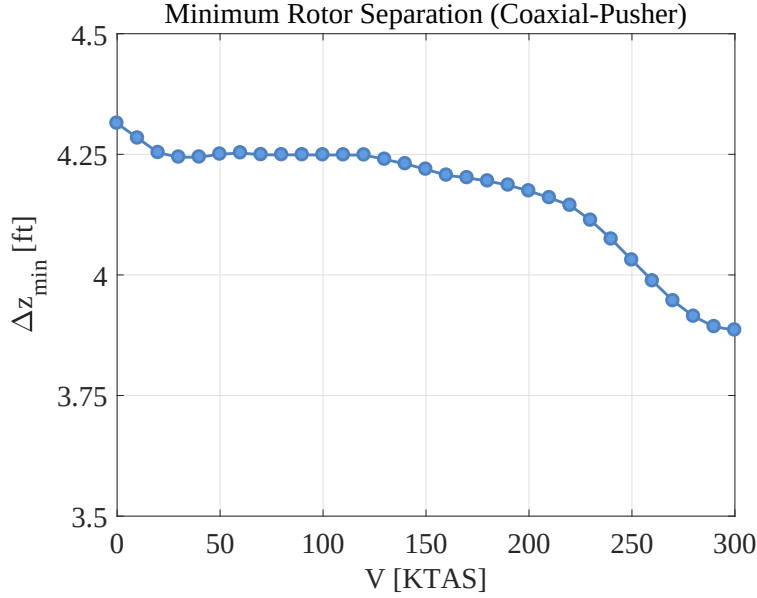


Figure 2.14. Minimum rotor separation $\Delta z_{\min}$ as a function of true airspeed (coaxial-pusher, sea level).

2.1.3 Tiltrotor

2.1.3.1 Description

The generic tiltrotor configuration was derived from scaling geometric, inertial, and structural properties of the XV-15 [85–87], V-22 [88], and the notional NASA Large Civilian Tilt-Rotor 2 (LCTR2) [89, 90]. The resulting tiltrotor aircraft is shown in Figure 2.15, and is similar to the configuration used in the control allocation study in Ref. [91].

The generic tiltrotor has a stiff in-plane hingeless rotor system with non-dimensional flap and lag modes similar to the LCTR2. The same airfoils were used as in the LCTR2. Baseline blade properties for twist and chord are derived from the XV-15 and tuned to be consistent with trends from more advanced tiltrotors like the LCTR2, resulting in increased stiffness and reduced weight than compared to conventional tiltrotors (e.g., XV-15, V-22, AW609).

The wings are scaled geometrically with data from the other tiltrotors, but do not have any forward sweep to remain consistent with the newer V-280 design. The wings have both inboard flaps and outboard ailerons similar to the XV-15. The flaps are

schedule with airspeed and nacelle angle to extend in hover and retract in forward flight.

The inflow model for this configuration leaves the wake of each rotor isolated. The wake impinges on the wing and results in an additional down force in hover.

Aerodynamics for the V-tail come from lookup tables of representative airfoils. The V-tail flaperons can deflect symmetrically (equivalent elevator deflection δ_e) to produce a pitching moment or asymmetrically (equivalent rudder deflection δ_r) to produce a yawing moment.

A set of key aircraft properties is given in Table 2.2. The aircraft mission representative gross weight is 32,100 lbs. Maximum speed is limited to $V_{TAS} = 280$ kts using notional engines that provide a total of 9,400 hp. The rotors are four-bladed with a radius of $R = 17.8$ ft each. The blades of both rotors are precone by $\beta_p = 2.5$ deg at the hub to reduce steady flapwise bending stresses. Since the rotors act as propellers in forward flight (i.e., proprotors), the blades have a significant amount of pretwist $\theta_{tw} = -44$ deg.

The nacelle tilt angle range for the tiltrotor is from $\delta_{nac} = 0$ deg (airplane mode) through $\delta_{nac} = 90$ deg (helicopter mode), and up to $\delta_{nac} = 95$ deg. The nacelles are modeled as rigid bodies with a rotational degree-of-freedom pivoted around the wing tips.

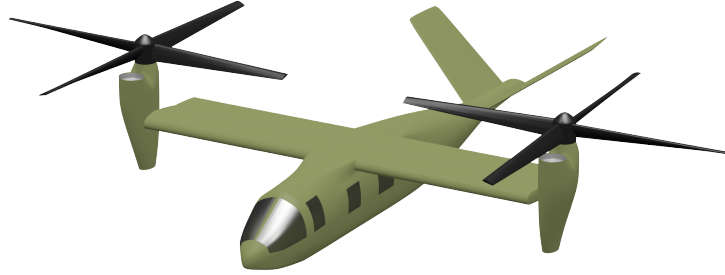


Figure 2.15. Generic tiltrotor rendering (reproduced from [74]).

2.1.3.2 Trim

Similarly to the coaxial-pusher HeliUM model, the tiltrotor HeliUM model was used to generate a set of linear models and trim data at the flight conditions spanning the conversion corridor shown in Figure 2.16. Figure 2.17 shows the trim controls and pitch attitude values for the tiltrotor as a function of airspeed and nacelle angle. As with the coaxial-pusher, the tiltrotor rotor controls are phased and ganged into

Table 2.2. Tiltrotor Configuration Data

<i>Aircraft Data</i>	
Gross Weight	32,100 lbs
Max Continuous Power (SL)	9,400 hp
V_{MCP} (KTAS, 6k95)	280 kts
Wing Span	45 ft
Wing Sweep	0 deg
Nacelle Range	0 – 95 deg
<i>Rotor Data</i>	
Radius	17.8 feet
Number of Blades/Rotor	4
Rotational Speed	40 – 30 rad/sec
Precone	2.5 deg
Twist	–44 deg

symmetric (collective θ_0 , lateral cyclic θ'_{1c} , and longitudinal cyclic θ'_{1s}) and differential (collective $\Delta\theta_0$, lateral cyclic $\Delta\theta'_{1c}$, and longitudinal cyclic $\Delta\theta'_{1s}$) controls. For the tiltrotor, a phase angle of $\Delta_{SP} = 50$ deg was chosen to minimize the off-axis flapping response.

The rotor speed NR is scheduled with nacelle angle (Figure 2.17, top plot) and is reduced from NR = 100% ($\Omega = 40$ rad/sec) to NR = 75% ($\Omega = 30$ rad/sec) when the nacelles are at $\delta_{nac} = 0$ deg (airplane mode).

Symmetric longitudinal cyclic θ'_{1s} (Figure 2.17, second plot) is used for trim for nacelle angles $\delta_{nac} > 0$ deg, but phased out in airplane mode ($\delta_{nac} = 0$ deg).

Trim collective θ_0 (Figure 2.17, third plot) begins at a similar value for all nacelle angles in hover. Below minimum drag speed $V_{TAS} \approx 70$ kts (region where induced drag dominates), less trim collective is required as airspeed is increased. Additionally, for a given airspeed, less trim collective is required for lower nacelle angles. This is because the aircraft is trimmed at a higher pitch attitude θ for lower nacelle angles (Figure 2.17, last plot), which translates to larger angle-of-attack α on the wing. Therefore, for lower nacelle angles, the lift share of the wing is higher and the rotor has to generate less lift. Above $V_{TAS} \approx 70$ kts (region where profile drag dominates), both trends are reversed. More trim collective is required for higher airspeeds, to overcome the increased drag. In addition, more trim collective is required for lower nacelle angles. This is because at high speed, a lower angle-of-attack α (achieved with higher nacelle angles) results in reduced profile drag.

Finally, unlike the coaxial-pusher, elevator deflection δ_e is used to trim the tiltrotor (Figure 2.17, fourth plot).

Figure 2.18 shows the trim power required P_{req} for the tiltrotor as a function of

airspeed at standard sea level conditions and at an altitude of $h = 6,000$ ft and OAT of $T = 95$ F (6k95). The figure also shows the power available P_{avail} provided by a pair of notional engines at sea level and 6k95. The notional engines were sized to provide sufficient power to hover at 6k95 and reach speeds of roughly $V_{\text{TAS}} = 280$ kts, resulting in $P_{\text{avail}_{\text{SL}}} = 4,700$ hp per engine.

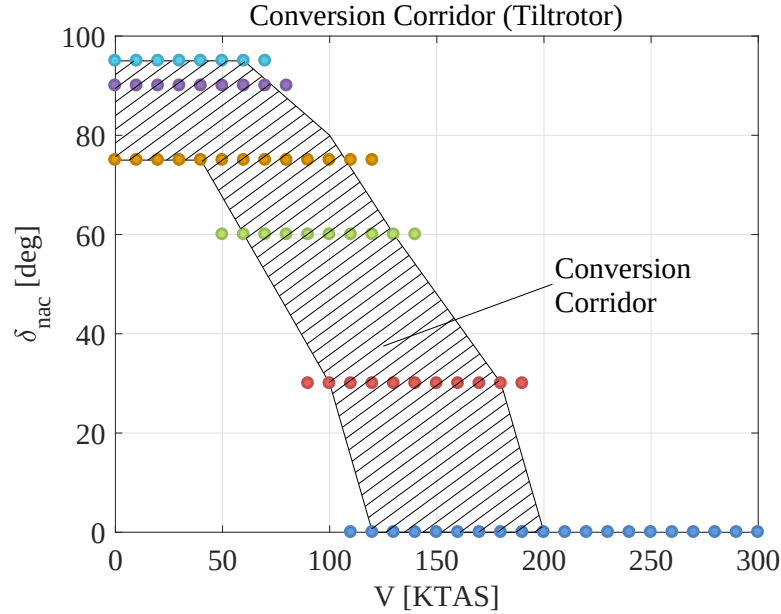


Figure 2.16. Conversion corridor and linear model points (tiltrotor, sea level).

2.1.3.3 Linear Models

Once trimmed, the HeliUM tiltrotor model was linearized at specific airspeeds and nacelle angles spanning the conversion corridor, shown as the points in Figure 2.16. This was repeated for several altitudes from sea level to 25,000 ft in increments of 5,000 ft. For brevity, only data at sea level is presented here.

The tiltrotor linear models contain 51 states:

- Nine fuselage states (9),
- Four (one collective, two cyclic, and one reactionless) second-order rotor states for each of the two blade modes retained in the linear model per rotor (32),
- Three (average, cosine, and sine) inflow states per rotor (6),
- Second-order nacelle rotational dynamics comprised of nacelle angle and nacelle angular rate per nacelle (4).

And 10 inputs:

- Symmetric lateral cyclic (θ'_{1c})

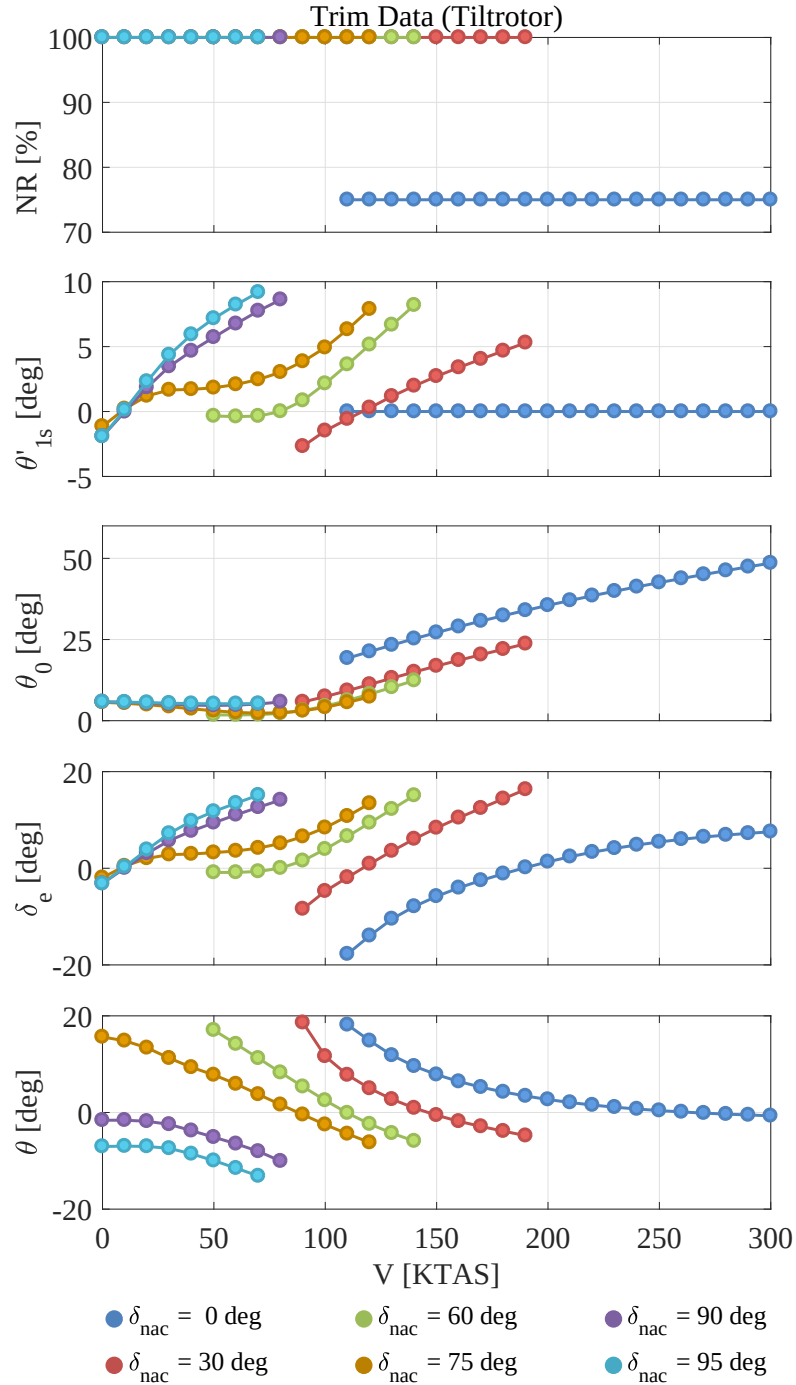


Figure 2.17. Trim rotor speed (NR), symmetric longitudinal cyclic pitch (θ'_{1s}), symmetric collective pitch (θ_0), elevator (δ_e), and pitch attitude (θ) as a function of true airspeed and nacelle angle (tiltrotor, sea level).

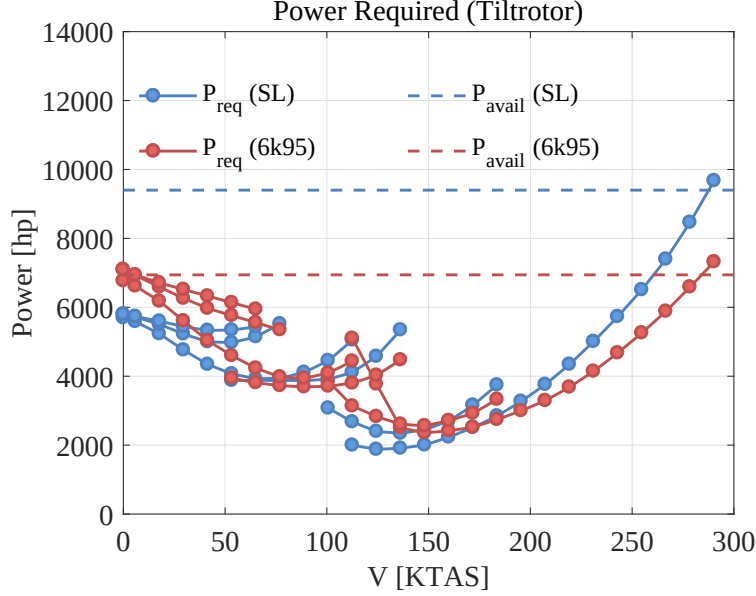


Figure 2.18. Power required and available at sea level (SL) and 6,000 ft 95 F (6k95) (tiltrotor).

- Differential collective ($\Delta\theta_0$)
- Symmetric longitudinal cyclic (θ'_{1s})
- Symmetric collective (θ_0)
- Differential longitudinal cyclic ($\Delta\theta'_{1s}$)
- Aileron (δ_a)
- Elevator (δ_e)
- Rudder (δ_r)
- Nacelle 1 torque (Q_{nac_1})
- Nacelle 2 torque (Q_{nac_2})

The nacelle torque inputs are the torques required to maintain the nacelles at their current angles by the nacelle actuators. A simple proportional-integral-derivative (PID) control system is used to convert commanded nacelle angles δ_{nac_1} and δ_{nac_2} to the required torques Q_{nac_1} and Q_{nac_2} , modeling the nacelle angle actuators.

Before further analysis was done, the two individual nacelle angle inputs and states were transformed to symmetric (δ_{nac}) and differential ($\Delta\delta_{nac}$) nacelle angles, with only symmetric nacelle angle deflections considered here.

As in the case of the coaxial-pusher model, in addition to all of the states being outputs, a power required output was also included in the linear models.

The remainder of this section will discuss trends in the linear models as a function of airspeed by looking at the quasi-steady rigid-body stability and control derivatives (as described in Sec. 2.1.2.3), blade modes, and eigenvalues.

2.1.3.3.1 Stability Derivatives

The speed stability derivative M_u (Figure 2.19, first row, first column) starts at a positive value in hover for all nacelle angles. As in the case of the coaxial-pusher, the value of M_u approaches zero as airspeed increases. At high airspeeds with the nacelles at $\delta_{\text{nac}} = 0$ deg, $M_u \approx 0$ which is typical of fixed-wing aircraft.

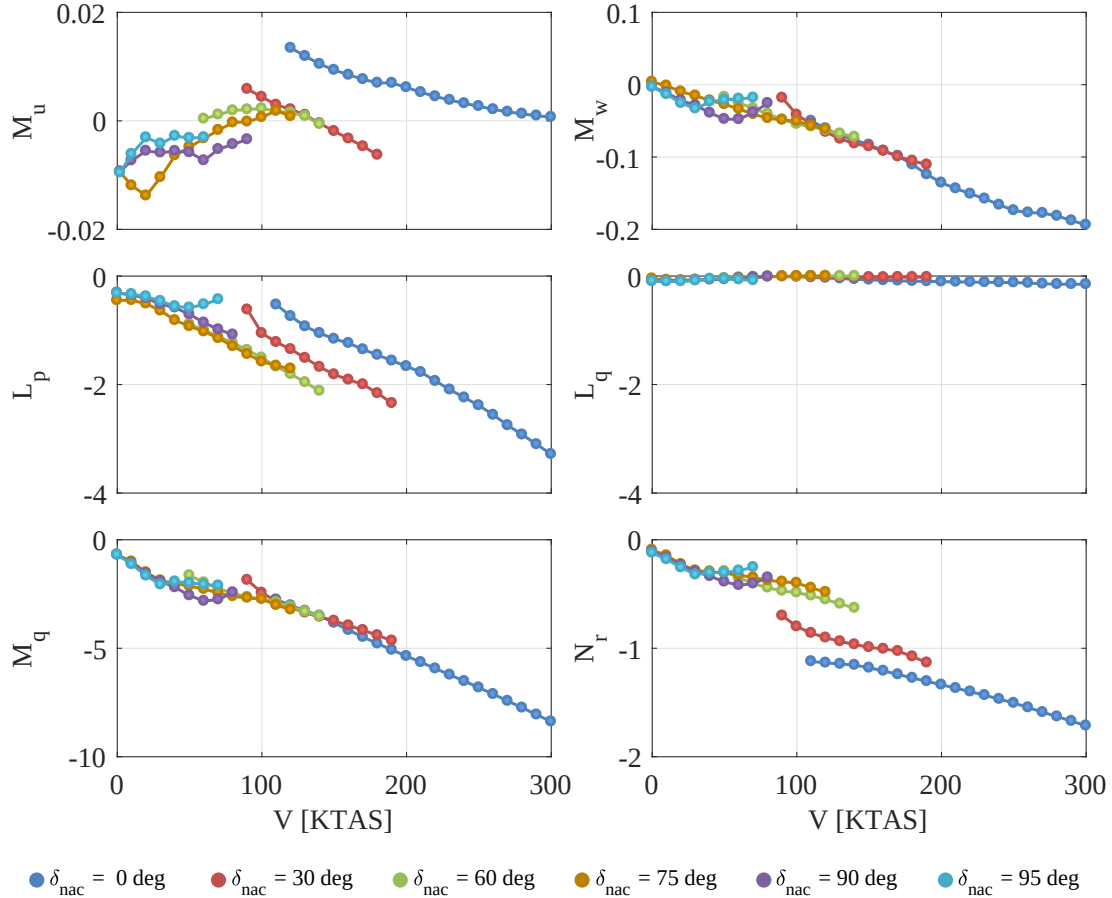


Figure 2.19. Rigid body stability derivatives as a function of true airspeed (tiltrotor, sea level).

The longitudinal static stability derivative M_w (Figure 2.19, first row, second column) starts at $M_w \approx 0$ for hover, and becomes larger in magnitude (more negative) as airspeed increases. Since $M_w < 0$, this aircraft is statically stable in forward flight.

The rate damping derivatives L_p , M_q , and N_r (Figure 2.19, second row, first column; third row, first column; third row, second column) are all negative and increase in magnitude with increasing airspeed. At airspeeds above $V_{\text{TAS}} \approx 100$ kts, the roll rate damping derivative L_p value is a function of nacelle angle δ_{nac} , with larger magnitude

(more negative) for lower nacelle angle. This is because the contribution of the rotors to L_p decreases as the nacelle angle decreases and the rotors become less aligned with the airflow resulting from aircraft rolling motion. This is in contrast to the pitch rate damping derivative M_q values which are similar for different nacelle angle δ_{nac} for $V_{\text{TAS}} > 100$ kts, since the main contribution is from the tailplane.

The coupling derivatives L_q and M_p (Figure 2.19, second row, second column, L_q only shown) are essentially zero for the tiltrotor due to its symmetric configurations, as is the case for the smaller XV-15 tiltrotor aircraft [83].

2.1.3.3.2 Lateral/Directional Control Derivatives

In hover, rolling moment can be generated using differential collective $\Delta\theta_0$ as well as symmetric lateral cyclic θ'_{1c} , as seen by the values of the $L_{\Delta\theta_0}$ (Figure 2.20, first row, first column) and $L_{\theta'_{1c}}$ (Figure 2.20, first row, second column) control derivatives. The values of $L_{\Delta\theta_0}$ increase in magnitude with increasing airspeed, and decrease with decreasing nacelle angle. However, $L_{\Delta\theta_0}$ is still significant in airplane mode ($\delta_{\text{nac}} = 0$ deg) due to the differential torque effect. As the nacelles are rotated down towards airplane mode ($\delta_{\text{nac}} = 0$ deg), differential collective $\Delta\theta_0$ generates more yawing moment, with $N_{\Delta\theta_0}$ (Figure 2.20, third row, second column) remaining constant with airspeed but increasing in magnitude with decreasing nacelle angle.

Differential longitudinal cyclic $\Delta\theta'_{1s}$ is the primary yaw control in hover/low-speed with $N_{\Delta\theta'_{1s}}$ (Figure 2.20, fourth row, first column) having the largest magnitude of the yaw control derivatives for airspeeds below $V_{\text{TAS}} \approx 90$ kts.

Rolling moment due to ailerons increases proportionally to dynamic pressure or V^2 with increasing airspeed as seen from the L_{δ_a} derivative (Figure 2.20, second row, second column). Similarly, yawing moment due to rudder increases proportionally to V^2 as seen by the N_{δ_r} derivative (Figure 2.20, fourth row, second column). Unique to the V-tail configuration of the tiltrotor, effective rudder deflection δ_r also generates large rolling moment, with roughly equal magnitudes of L_{δ_r} (Figure 2.20, third row, first column) and N_{δ_r} . Furthermore, the roll generated due to effective rudder deflection δ_r is adverse ($L_{\delta_r} < 0$, $N_{\delta_r} > 0$), with a left-wing-down roll moment being generated for nose-right yaw moment. This undesired behavior should be addressed with the flight controls through control mixing.

Longitudinal and heave responses due to lateral/directional inputs are essentially zero due to the symmetric rotor configuration.

2.1.3.3.3 Longitudinal/Heave Control Derivatives

The primary pitch control derivative at hover/low-speed is pitching moment due to symmetric longitudinal cyclic $M_{\theta'_{1s}}$ (Figure 2.21, third row, second column). This derivative has a similar variation with airspeed and nacelle angle as rolling moment due to symmetric lateral cyclic $L_{\theta'_{1c}}$. Symmetric longitudinal cyclic also generates

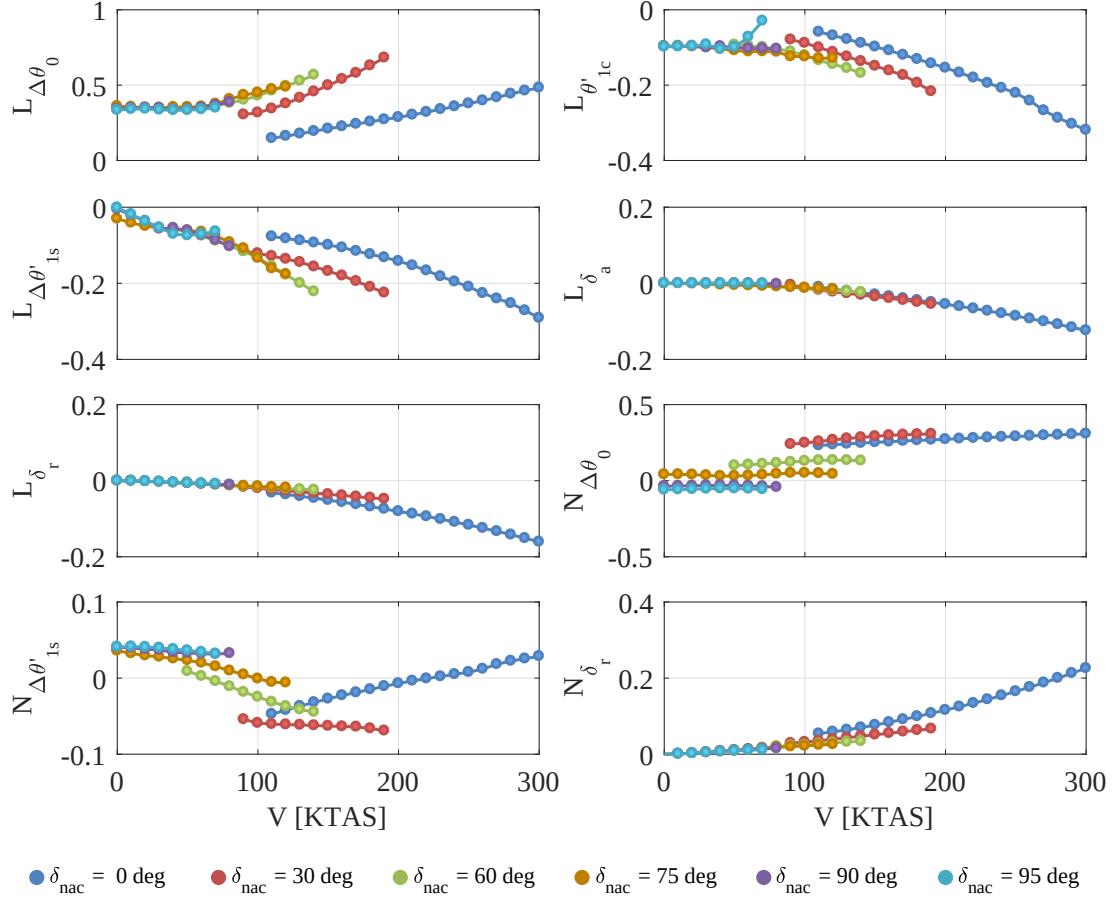


Figure 2.20. Rigid body lateral/directional control derivatives as a function of true airspeed (tiltrotor, sea level).

z-axis force, as seen by the $Z_{\theta'_{1s}}$ derivative (Figure 2.21, first row, second column), with its value increasing with increasing airspeed.

In hover/low-speed, the primary control for z-axis force is symmetric collective θ_0 , as seen by the Z_{θ_0} derivative (Figure 2.21, second row, first column). As airspeed is increased and nacelle angle is reduced to $\delta_{nac} = 0$, the primary control for pitching moment and z-axis force is effective elevator δ_e , with its effectiveness increasing proportionally to dynamic pressure or V^2 with increasing airspeed, as seen by M_{δ_e} (Figure 2.21, fourth row, first column) and Z_{δ_e} (Figure 2.21, second row, second column).

Increasing nacelle angle tilts the thrust vector up, creating a negative z-axis force, as seen by the negative values of the $Z_{\delta_{nac}}$ derivative (Figure 2.21, third row, first column). The magnitude of $Z_{\delta_{nac}}$ is proportional to the amount of thrust generated by the rotors, which increases with airspeed. Similarly, the pitching moment generated by

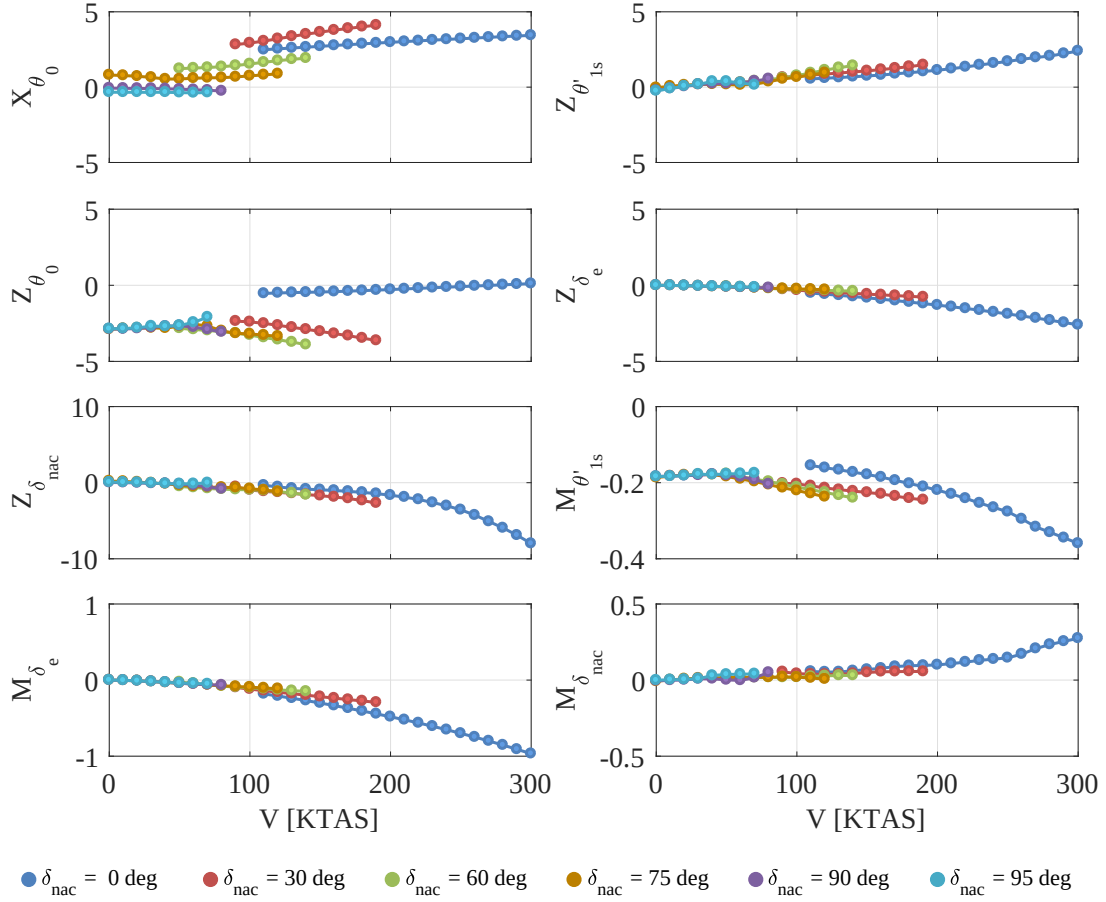


Figure 2.21. Rigid body longitudinal/heave control derivatives as a function of true airspeed (tiltrotor, sea level).

the nacelles $M_{\delta_{nac}}$ (Figure 2.21, fourth row, second column) increases with airspeed.

2.1.3.3.4 Blade Modes

As with the coaxial-pusher linear models, two coupled flap-lag rotor blade modes are retained in the linearized tiltrotor models. Figure 2.22 shows the natural frequencies of the two coupled modes as a function of rotor speed, with the primary motion (first flap bending 1F or flight lag bending 1L) labeled.

Figure 2.23 shows the flap and lag blade deflection contributing to each of the two blade modes for the nominal rotor speed ($\Omega = \Omega_0 = 40$ rad/sec) and reduced rotor speed ($\Omega = 0.75\Omega_0 = 30$ rad/sec). The tiltrotor flap and lag modes are more tightly coupled as compared to the coaxial-pusher (Figure 2.10). This is primarily due to the large amount of pretwist of the tiltrotor's proprotor blades ($\theta_{tw} = -44$

deg, Table 2.2) as compared to the coaxial-pusher rotor blades ($\theta_{tw} = -9$ deg, Table 2.1). Pretwist introduces structural coupling between flap and lag deflections [92], which causes Mode 1 to have nearly equal magnitude deflections in the flap and lag directions for both $\Omega = \Omega_0$ and $\Omega = 0.75\Omega_0$ (Figure 2.23, upper plot).

A second differences between the coaxial-pusher and tiltrotor blade modes is that the tiltrotor blade Mode 2 is at significantly higher normalized frequency. This is due to the larger chordwise and beamwise stiffness of the tiltrotor blades.

At both the nominal ($\Omega/\Omega_0 = 1$) and reduced ($\Omega/\Omega_0 = 0.75$) rotor speeds, Modes 1 and 2 are de-conflicted from any of the integer rotor harmonics (e.g., 1/rev, 2/rev, etc.). However, Mode 2 does cross the 3/rev harmonic at around $\Omega/\Omega_0 = 0.8$. Dwelling in this region of rotor speed can be avoided by introducing hysteresis in the rotor speed nacelle angle schedule.

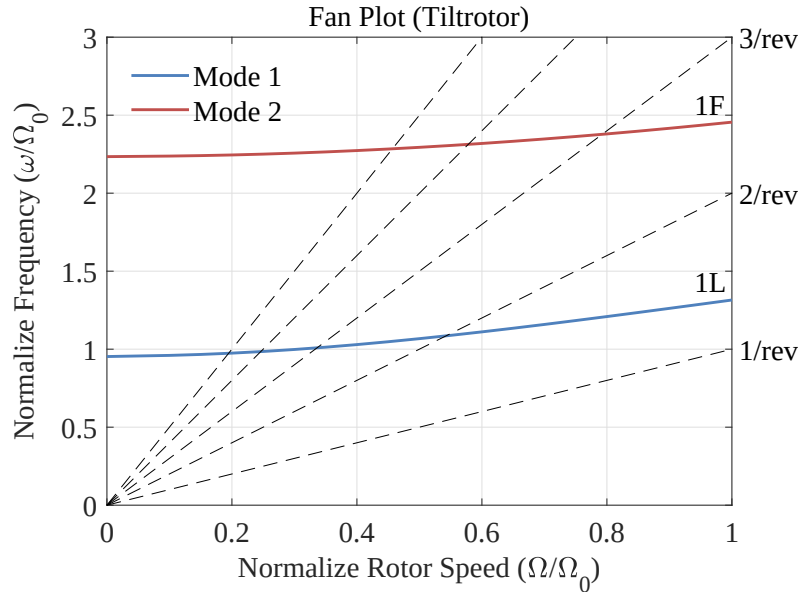


Figure 2.22. Blade mode fan diagram (tiltrotor).

2.1.3.3.5 Eigenvalues

Figures 2.24 and 2.25 show a zoomed-out and zoomed-in view, respectively, of the eigenvalues of the full-order (51-state) linear state-space models plus nacelle controllers (2-state) as a function of airspeed and nacelle angle. The collective, progressive, regressive, and reactionless rotor modes can be seen in Figure 2.24. These rotor modes correspond to blade Mode 1 and Mode 2 discussed in the previous section, and are labeled by their dominant motion (flap or lag). Note that there are two of each rotor mode (for the two rotor), for a total of 16 modes.

The tiltrotor rotor modes are stronger functions of airspeed than the coaxial-pusher rotor modes (Figure 2.24), with the tiltrotor modes varying the most in airplane mode

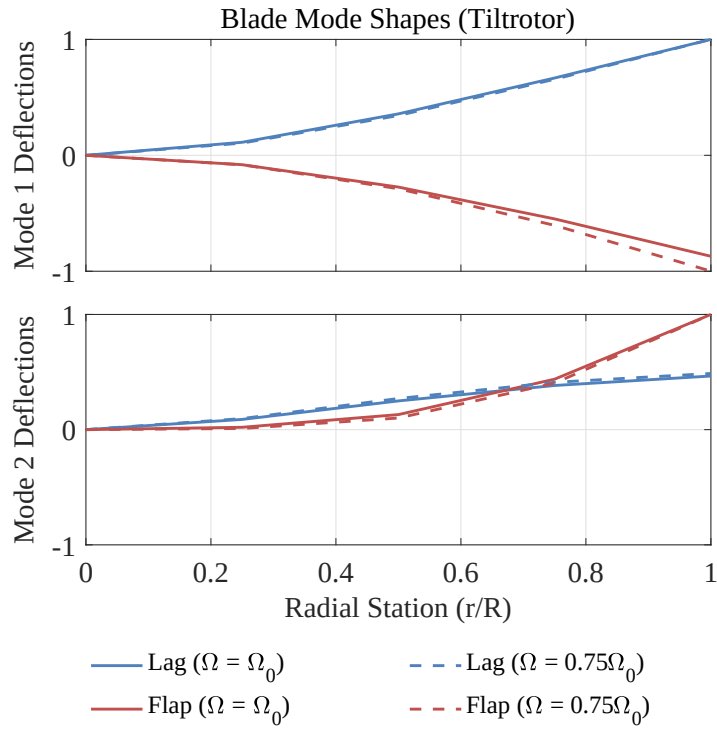


Figure 2.23. Flap and lag deflections for blade Modes 1 and 2 (tiltrotor).

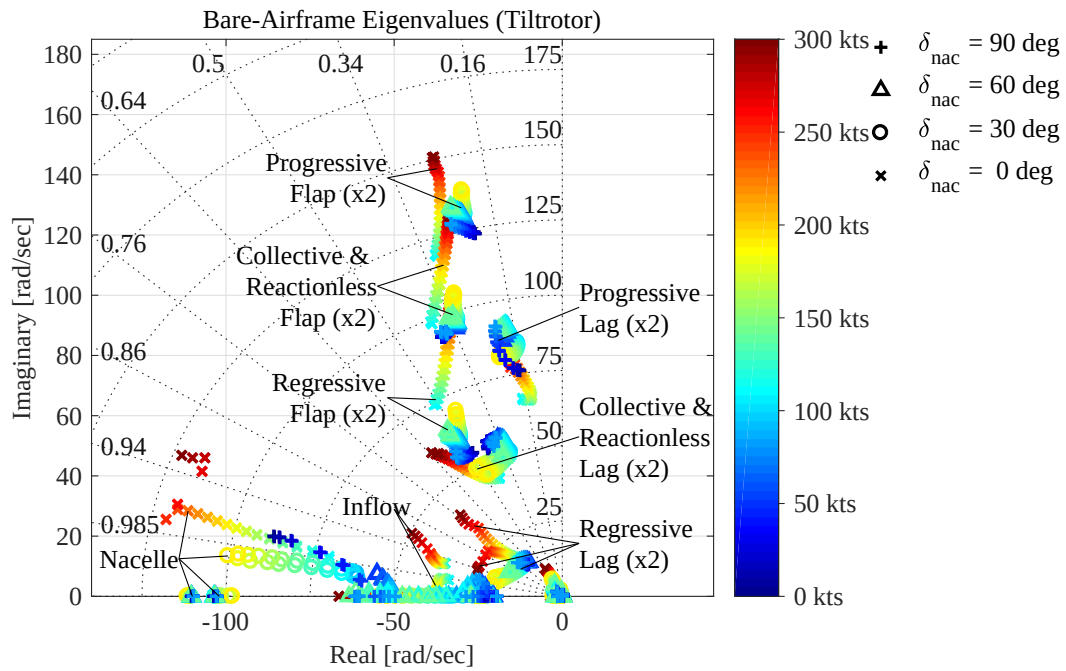


Figure 2.24. Eigenvalues as a function of true airspeed (tiltrotor, sea level).

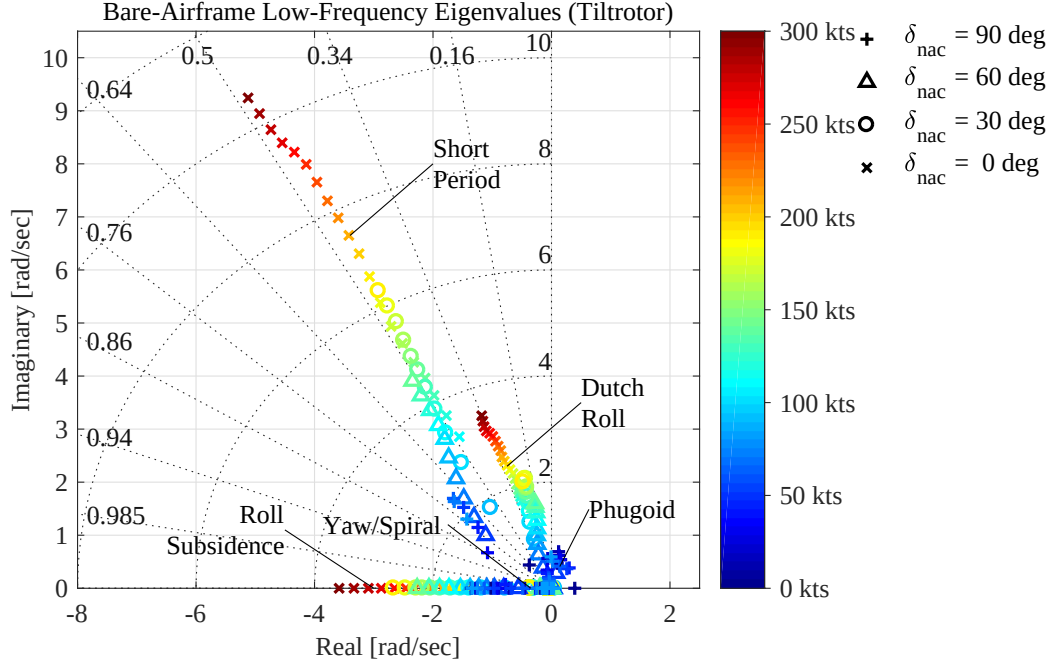


Figure 2.25. Low-frequency eigenvalues as a function of true airspeed (tiltrotor, sea level).

($\delta_{\text{nac}} = 0$ deg). This is because the tiltrotor has a much wider trim collective θ_0 range (Figure 2.17, third plot) as compared to the coaxial-pusher (Figure 2.3, third plot). The rotor modes are all stable, with the lower-frequency Mode 1 being more lightly damped than the higher-frequency Mode 2.

Inflow state modes can be seen in Figure 2.24 around the rotor speed frequency ($\omega = 30 - 40$ rad/sec). In addition, high-frequency nacelle angle modes can be seen at around $\omega = 60 - 120$. These modes are a consequence of modeling the nacelles as rigid bodies with a rotational degree-of-freedom. Note that these nacelle modes are well damped ($\zeta > 0.9$) due to the inclusion of the nacelle controller described above.

Figure 2.25 shows the fuselage eigenvalues of the tiltrotor as a function of airspeed and nacelle angle. At hover, there are low-frequency unstable complex modes (hovering cubic) in both the lateral (marked Dutch Roll in Figure 2.25) and longitudinal (marked Phugoid in Figure 2.25) axes. In the lateral axis, this mode stabilizes and increases in frequency as airspeed increases, becoming the lightly damped Dutch roll mode ($\zeta_{\text{dr}} = 0.16 - 0.34$). In the longitudinal axis, the oscillatory phugoid mode reduces in frequency and becomes stable as airspeed increases. Eventually, as airspeed continues to increase, the phugoid mode becomes critically damped and turns into two real modes.

In hover/low-speed, there are two real pitch and heave modes. As airspeed increases, these two modes combine to form the oscillatory short period mode (labeled in Figure 2.25). As expected from the negative values of M_w seen in Figure 2.19, the short period mode of the tiltrotor is stable and well damped ($\zeta_{\text{sp}} \approx 0.5$). The short

period mode increases in frequency with increasing airspeed.

A real roll subsidence mode ($1/T_r$) is present which increases in frequency with increasing airspeed, from about $1/T_r = 0.3$ rad/sec at hover to $1/T_r = 3.5$ rad/sec at $V = 300$ kts. These values of roll mode inverse time constant correspond to the values of the effective L_p derivative seen in Figure 2.19.

Finally, as with the coaxial-pusher, there is a low frequency real yaw/spiral mode, as seen in Figure 2.25.

2.2 Control Allocation

2.2.1 Overview

A control allocation scheme is required for both the coaxial-pusher and tiltrotor aircraft, since both have redundant controls. A weighted pseudo-inverse method [91, 93] is used to allocate the demanded roll, pitch, and yaw moments $\mathbf{d} = [\delta_{lat} \ \delta_{lon} \ \delta_{ped}]^T$ to each aircraft's control actuator commands $\mathbf{u}_{cmd}$:

$$\mathbf{u}_{cmd} = \mathbf{W}^{-1} \hat{\mathbf{B}}_{RB_{pqr}}^T \left(\hat{\mathbf{B}}_{RB_{pqr}} \mathbf{W}^{-1} \hat{\mathbf{B}}_{RB_{pqr}}^T \right)^{-1} \mathbf{d} \quad (2.19)$$

where $\mathbf{W}$ is a diagonal weighting matrix composed of the individual w_i weightings (Eq. 2.20) and $\hat{\mathbf{B}}_{RB_{pqr}}$ is the control effectiveness matrix, composed of the $\dot{p}$, $\dot{q}$, and $\dot{r}$ rows of the quasi-steady rigid-body control matrix (Eq. 2.15).

Since rate limiting of control surfaces increases equivalent delay in the aircraft response and can result in pilot induced oscillations (PIO) [48], a weighting scheme based on the rate limit of each actuator $\dot{u}_{i_{max}}$ was used:

$$w_i = \frac{1}{\dot{u}_{i_{max}}} \quad (2.20)$$

This weighting scheme drives the control allocation to prioritize, or utilize more, actuators with higher rate limits. Table 2.3 lists the actuator position and rate limits for both coaxial-pusher and tiltrotor. Note that since the actuators used in the weighted pseudo inverse control allocation all have similar ranges, the control derivative matrix $\hat{\mathbf{B}}_{RB_{pqr}}$ was not scaled to percent control deflection, however that is the recommended approach.

The quasi-steady control matrix, derived by reducing out the rotor, inflow, and nacelle angle states from the linearized coaxial-pusher and tiltrotor models (as described in Sec. 2.1.2.3), is used to get the effective rigid-body control derivative matrix $\hat{\mathbf{B}}_{RB_{pqr}}$. In the full-order system, the controls only affect the rotor modes which then affect rigid body motion through the stability matrix $\mathbf{A}$. Therefore using the full-order control matrix $\mathbf{B}$ would result in incorrect control allocation.

Table 2.3. Actuator Position and Rate Limits (Coaxial-Pusher and Tiltrotor)

Actuator	Position Limit	Rate Limit
	$u_{i_{\max}}$ [deg]	$\dot{u}_{i_{\max}}$ [deg/sec]
<i>Both Aircraft</i>		
Main Rotor	± 20	40
Elevator	± 20	80
Rudder	± 30	70
<i>Coaxial-Pusher Only</i>		
Pusher Prop. Collective	$-10 - 100$	5
Pusher Prop. Monocyclic	± 10	20
<i>Tiltrotor Only</i>		
Aileron	± 40	80

The following sections present the control allocation for both aircraft as a function of airspeed on radar charts, with each spoke corresponding to a bare-airframe input. The control allocation to the individual swashplate actuators of the two rotors was phased and ganged into the symmetric and differential rotor controls presented earlier (Sec. 2.1.2.2). Each radar chart has as many spokes as the number of available controls for a given moment demand. The magnitude of moment demand allocated to each control input corresponds to where each colored line crosses the spokes, normalized by the largest value across all airspeeds (and nacelle angles in the case of the tiltrotor).

2.2.2 Coaxial-Pusher Control Allocation

Figure 2.26 shows the coaxial-pusher roll, pitch, and yaw control allocation as a function of airspeed. Recall from Sec. 2.1.2.2 that the coaxial-pusher controls are phased (denoted by the primed cyclic pitch inputs θ'_{1c} and θ'_{1s}) using a constant phase angle $\Delta_{sp} = 70$ deg. As shown here, the control allocation introduces some additional phasing to decouple the roll and pitch motion of the aircraft by allocating commands to differential lateral and longitudinal cyclic inputs $\Delta\theta'_{1c}$ and $\Delta\theta'_{1s}$. This results in the optimal phase angle which is shown in Figure 2.27 as a function of airspeed.

In hover, roll moment demand is allocated to symmetric lateral cyclic θ'_{1c} . Some differential longitudinal cyclic $\Delta\theta'_{1s}$ is also used, which phases the symmetric lateral cyclic input. For airspeeds faster than $V_{TAS} = 100$ kts, the amount of phasing required is decreased. As airspeed is increased, differential collective $\Delta\theta_0$ and rudder δ_r are used to remove the bare-airframe yaw-due-to-roll control coupling.

In hover, pitching moment demand is allocated to symmetric longitudinal cyclic

θ'_{1s} and symmetric collective θ_0 . Note that some differential lateral cyclic $\Delta\theta'_{1c}$ is used, which phases the symmetric longitudinal cyclic input. As airspeed increases, less pitching moment demand is allocated to symmetric collective θ_0 , with that control being phased out by about $V_{TAS} = 120$ kts. As airspeed continues to increase, more pitching moment demand is allocated to the elevator δ_e .

Yaw moment demand is allocated to differential collective $\Delta\theta_0$ at hover. As airspeed increases, pusher propeller monocyclic θ_{1cPP} , rudder δ_r and differential longitudinal cyclic $\Delta\theta'_{1s}$ are phased in. Differential longitudinal cyclic $\Delta\theta'_{1s}$ is phased out at around $V_{TAS} = 110$ kts, and pusher propeller monocyclic θ_{1cPP} is phased out at around $V_{TAS} = 150$ kts. At airspeeds faster than $V = 150$ kts, yaw moment is allocated to the rudder only.

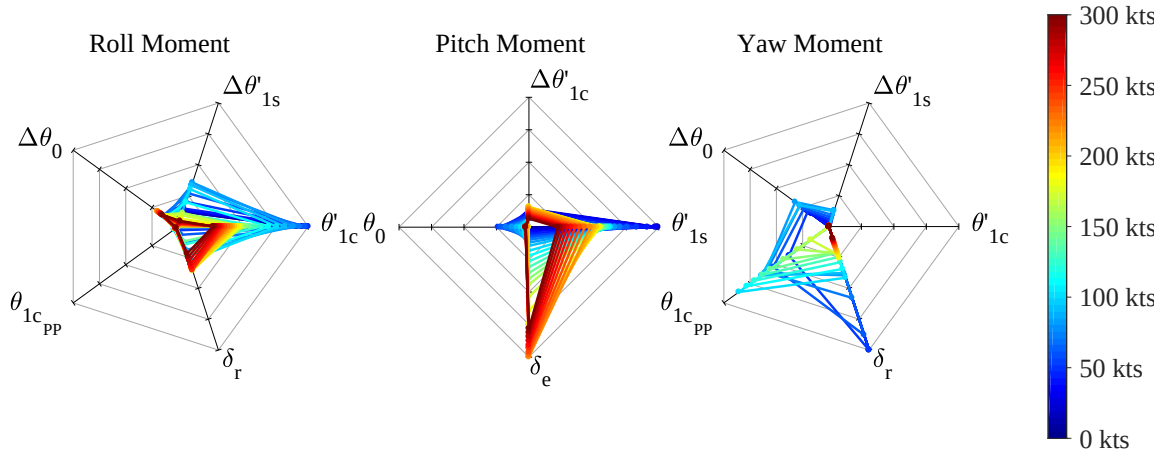


Figure 2.26. Roll, pitch, and yaw moment control allocation as a function of true airspeed (coaxial-pusher).

2.2.3 Tiltrotor Control Allocation

Figure 2.28 shows the tiltrotor control allocation as a function of airspeed and nacelle angle. In hover with the nacelles at $\delta_{nac} = 90$ deg, roll control is achieved using differential collective $\Delta\theta_0$ and symmetric lateral cyclic θ'_{1c} . Differential longitudinal cyclic $\Delta\theta'_{1s}$ is used to remove the yaw-due-to-roll control coupling. As airspeed is increased with the nacelles at $\delta_{nac} = 90$ deg, effective rudder δ_r is also used to remove the yaw-due-to-roll control coupling. This is equivalent to the control crossfeed in the XV-15 [83].

With the nacelles at $\delta_{nac} = 60$ deg, roll control is also achieved using differential collective $\Delta\theta_0$ and symmetric lateral cyclic θ'_{1c} , with aileron δ_a fading in with increased airspeed. With the nacelles tilted down ($\delta_{nac} < 90$ deg), differential longitudinal cyclic $\Delta\theta'_{1s}$ is used to generate roll moment as well as yaw moment. Furthermore, effective rudder δ_r deflections are used for roll moment as well (due to the V-tail configuration).

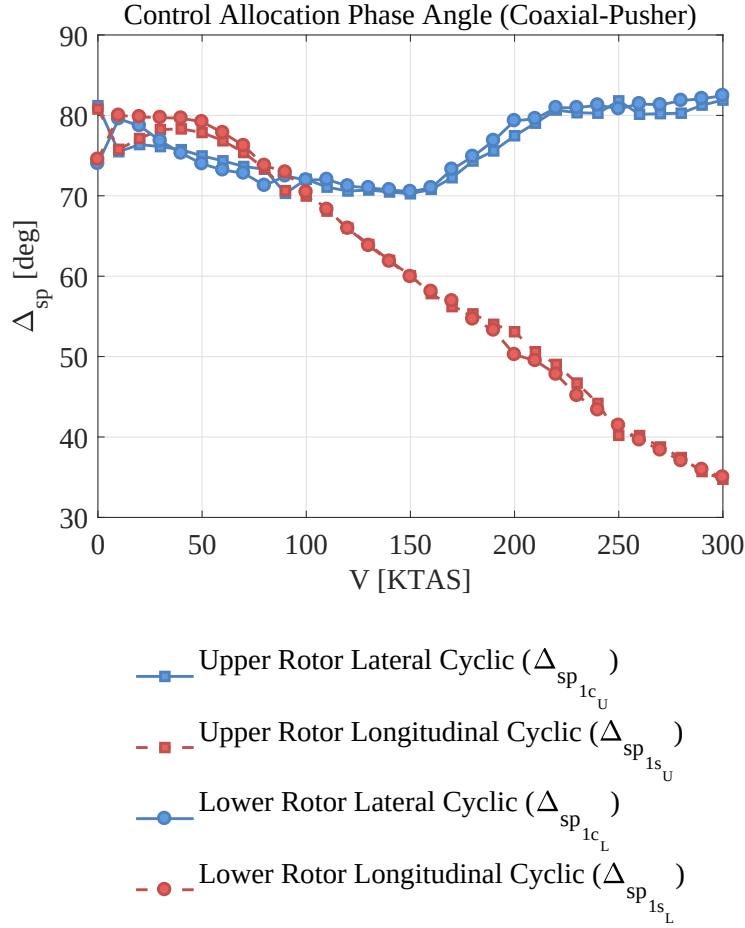


Figure 2.27. Phase angle Δ_{sp} provided by control allocation as a function of true airspeed (coaxial-pusher).

For $\delta_{nac} = 30$ deg, roll control is provided primarily through differential longitudinal cyclic $\Delta\theta'_{1s}$, ailerons δ_a , and rudder δ_r . In airplane mode ($\delta_{nac} = 0$ deg), the weighted pseudo inverse control allocation solution was replaced with a control ganging method. Roll control is provided through ailerons δ_a and differential longitudinal cyclic $\Delta\theta'_{1s}$. Differential longitudinal cyclic $\Delta\theta'_{1s}$ is used to supplement the ailerons in airplane mode to provide sufficient control authority to meet the MIL-STD-1797B [17] time to bank requirement.

Pitch control is achieved through symmetric longitudinal cyclic θ'_{1s} , symmetric collective θ_0 , and elevator δ_e . For nacelle angles greater than $\delta_{nac} = 0$ deg, all three controls are used, with the demanded pitch moment allocation to rotor controls (θ'_{1s} and θ_0) decreasing and pitch moment allocation to aerosurface controls (δ_e) increasing with increasing airspeed. For $\delta_{nac} = 0$ deg (airplane mode), demanded pitch moment

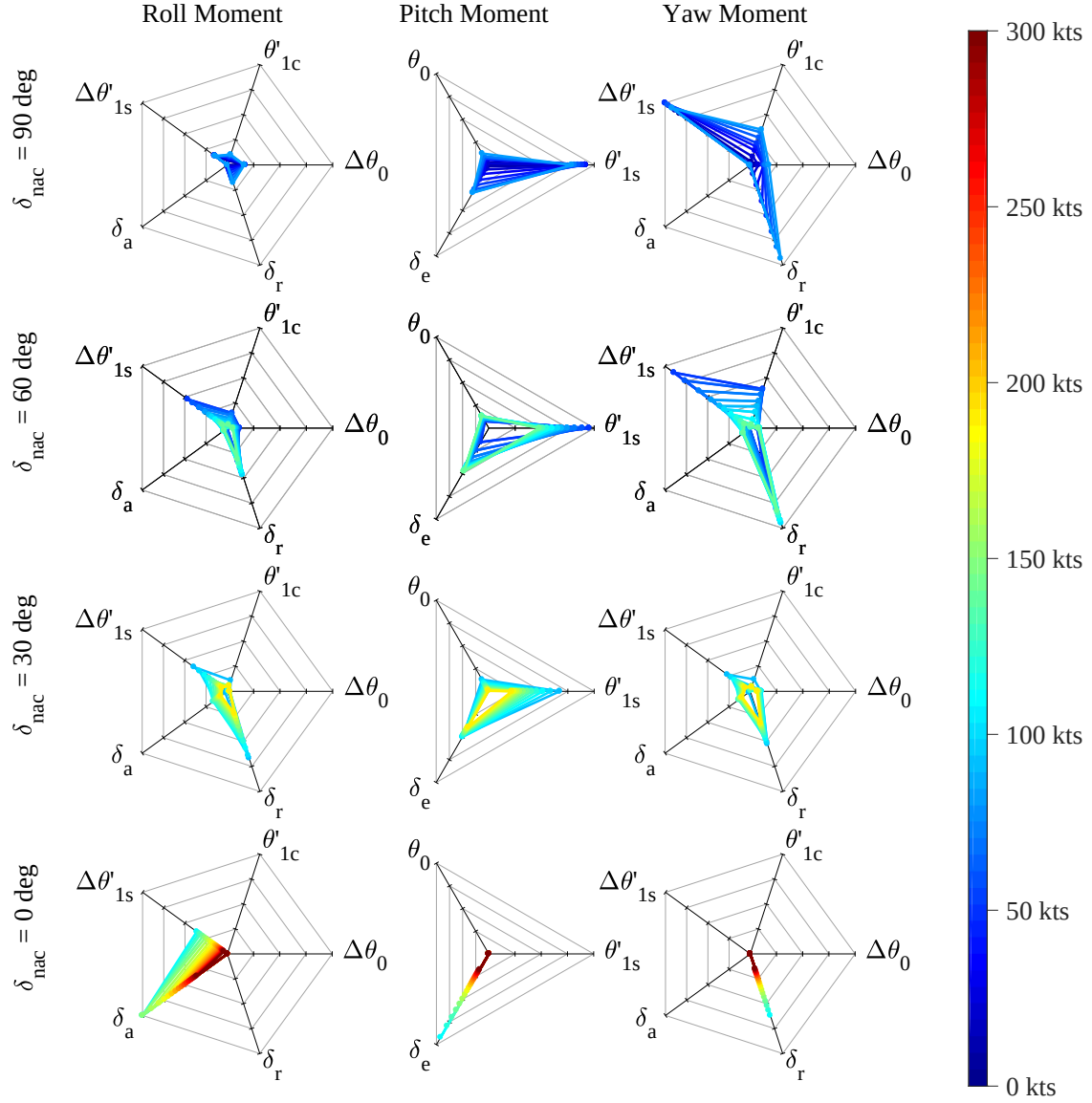


Figure 2.28. Roll, pitch, and yaw moment control allocation as a function of nacelle angle and true airspeed (tiltrotor).

is allocated to the elevator δ_e only, with decreasing amount as airspeed increases and the elevator becomes more effective.

Finally, with the nacelles at $\delta_{\text{nac}} = 90 \text{ deg}$, yaw control is achieved through differential longitudinal cyclic at hover $\Delta\theta'_{1s}$. As airspeed increases, rudder δ_r is used as well. Differential collective $\Delta\theta_0$ and symmetric lateral cyclic θ'_{1c} are used to remove the roll-due-to-yaw control coupling. With the nacelles at $\delta_{\text{nac}} = 60 \text{ deg}$, as airspeed increases, demanded yaw moment allocation decreases to rotor controls and

increases to the aerosurface control. For $\delta_{\text{nac}} = 30$ deg, rudder δ_r is the primary yaw control, with some demanded yaw moment being allocated to the rotor. In airplane mode ($\delta_{\text{nac}} = 0$ deg), demanded yaw moment is allocated to the rudder δ_r only, with decreasing amount as airspeed increases and the rudder becomes more effective.

2.3 Stitched Model

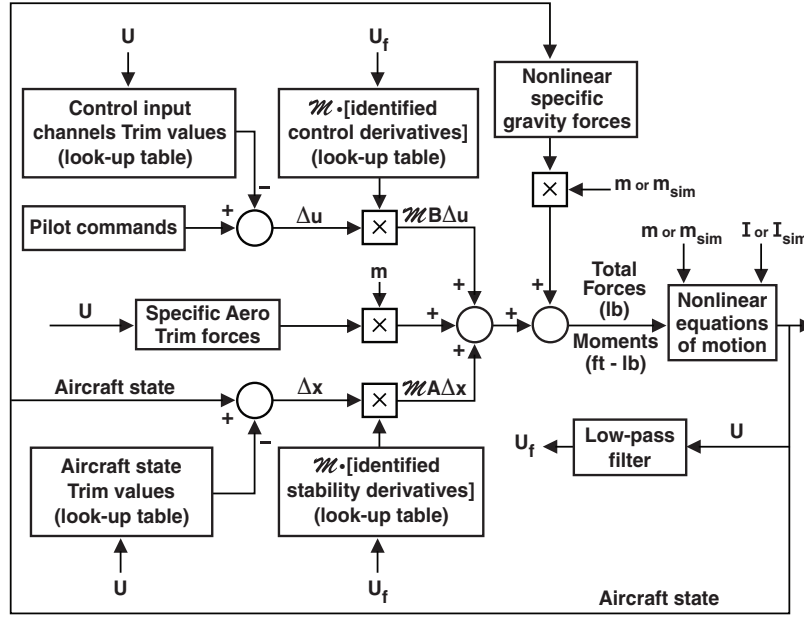
2.3.1 Overview

The linearized models and trim data extracted from HeliUM and presented above were used to develop a stitched simulation model [83, 94] of each aircraft, which can be run in real-time. At its core, the stitched model is comprised of a quasi-Linear-Parameter-Varying (qLPV) model, with distinctive features specific to aircraft and rotorcraft applications.

A block diagram schematic of the stitched model is shown in Figure 2.29 (reproduced from Ref. [83]). Note that in this schematic the model is only stitched with x-axis body axis velocity U . To summarize, model stitching is accomplished by implementing lookup tables of the aircraft state trim values $\mathbf{X}_0$, control input trim values $\mathbf{U}_0$, and stability $\mathbf{A}$ and control $\mathbf{B}$ derivatives based on linear models and trim data extracted from HeliUM. Trim states and controls are used to determine perturbation states $[\Delta \mathbf{x} = (\mathbf{X} - \mathbf{X}_0(U))]$ and controls $[\Delta \mathbf{u} = (\mathbf{U} - \mathbf{U}_0(U))]$, which in turn are multiplied by the stability and control derivatives and mass matrix ($\mathbf{MA}\Delta \mathbf{u}$ and $\mathbf{MB}\Delta \mathbf{x}$) to determine perturbation aerodynamic and control forces and moments. Note that the stability and control derivatives are determined based on a low-pass filtered x-axis body axis velocity U_f , to ensure that the derivative values remain approximately constant for short-term motion and retain the accurate dynamic response at the discrete point models. Trim values are determined based on instantaneous U , to preserve correct speed derivatives (e.g., X_u , M_u , etc.). The aerodynamic trim forces and moments are then summed to the perturbation values to yield the total aerodynamic forces and moments acting on the aircraft in body axes. The linearized Coriolis terms (e.g., W_0q , etc., due to formulating equations of motion in body axes) and linearized gravity terms normally included in the $\mathbf{A}$ and $\mathbf{B}$ state-space matrices are removed and added downstream in their nonlinear form.

For the coaxial-pusher, the stitched model is stitched in total x-axis body axis velocity U and scheduled with altitude h . For the tiltrotor the stitched model is stitched in total x-axis body axis velocity U and symmetric nacelle angle δ_{nac} and scheduled with altitude h .

The stitched models can be trimmed, simulated, and linearized at any flight condition within the flight envelope of the aircraft. They can also be used to extrapolate the models to different weight, inertia, and CG values. In addition, they contain three-point landing gear models to allow simulation of landing on moving platforms



Variable	Description
U	Total longitudinal body axis velocity
U_f	Filtered velocity
Δu	Control perturbations (e.g., δ_e)
Δx	State perturbations [e.g., $w \equiv (W - W_0)$]
$\mathcal{M}$	Mass and inertia matrix
A	Dimensional stability derivatives
B	Dimensional control derivatives
m	Aircraft mass
I	Aircraft inertia matrix

Figure 2.29. Model stitching block diagram schematic (reproduced from [83]).

(e.g., ship landing). The stitched models were used for control law development and piloted simulations (as discussed in the remainder of this report). The following subsections provide primary on-axis frequency responses of the linearized stitched model at several flight conditions. To generate the frequency responses, the control allocation described in Sec. 2.2 was included with the stitched models.

2.3.2 Coaxial-Pusher

Figures 2.30 through 2.34 show the primary on-axis frequency responses of the coaxial-pusher stitched model. Figure 2.30 shows the roll rate p to roll moment demand δ_{lat} (input to control allocation) frequency responses between hover and 250 kts. In hover, the roll rate response exhibits the unstable lateral phugoid at $\omega = 0.85$ rad/sec. The roll mode real pole (which along with the unstable lateral phugoid makes up the hovering cubic) is at $1/T_r = -L_p = 5$ rad/sec. The regressive flap mode can be seen as the peak at $\omega = 22$ rad/sec (corresponding to the eigenvalue labeled “Regressive Flap + p” in Figure 2.11), followed by the progressive flap mode at $\omega = 50$ rad/sec.

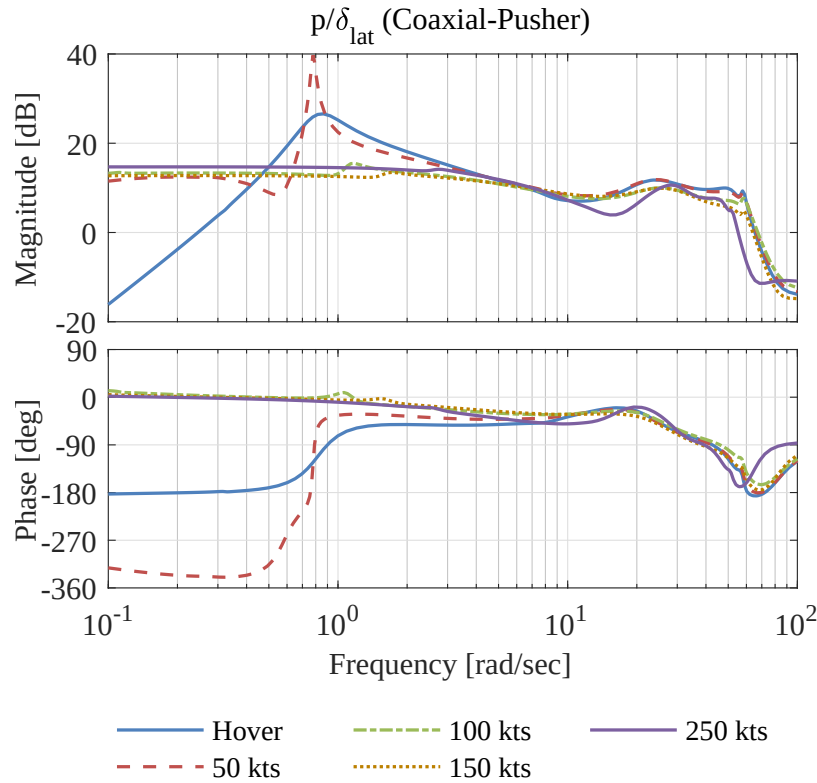


Figure 2.30. Roll rate frequency response (coaxial-pusher, sea level).

At $V_{\text{TAS}} = 50$ kts, the ω_ϕ complex pair of zeros [95] are present in the roll rate response at around the same frequency as the Dutch roll mode ($\omega_{\text{dr}} = 0.7$ rad/sec), flattening out the low-frequency asymptote. Furthermore, the Dutch roll mode damping is nearly $\zeta_{\text{dr}} = 0$ (corresponding to the eigenvalue labeled “Dutch Roll” in Figure 2.12). The rotor modes are similar to the hover case.

As airspeed increases, several trends can be seen: 1) The Dutch roll mode frequency increases and the ω_ϕ zeros more closely cancel the Dutch roll contribution to the roll response, 2) The roll mode increases in frequency to about $1/T_r = -L_p = 10$ rad/sec

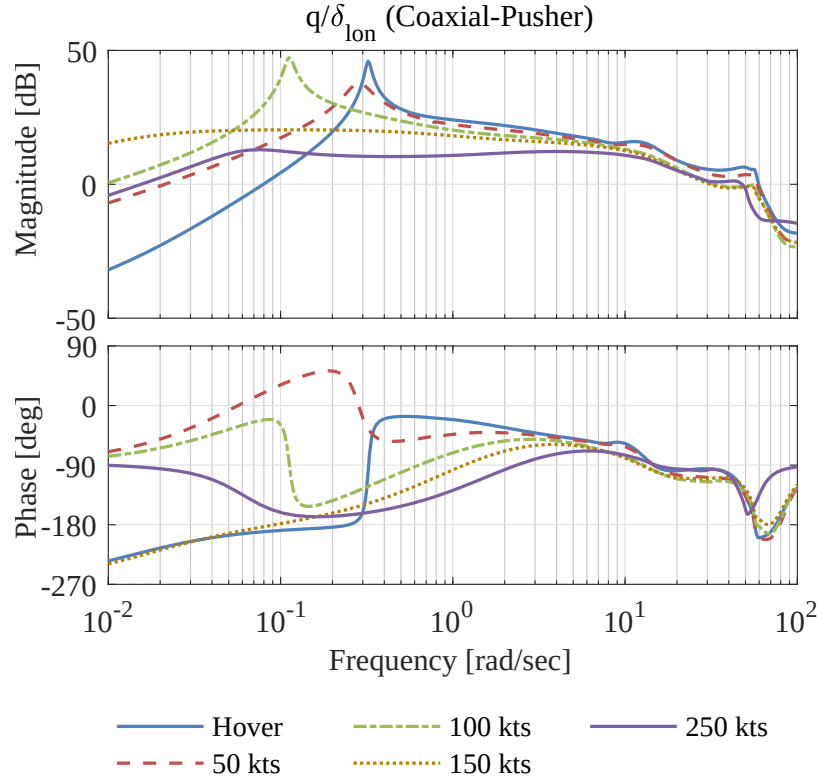


Figure 2.31. Pitch rate frequency response (coaxial-pusher, sea level).

at $V_{TAS} = 250$ kts, 3) The regressive flap mode increases in frequency to about $\omega = 28$ rad/sec at $V_{TAS} = 250$ kts, and 4) The progressive flap mode decreases in frequency to about $\omega = 48$ rad/sec at $V_{TAS} = 250$ kts. This is consistent with the variation of the rotor modes with airspeed seen in Figure 2.11.

Figure 2.31 shows the pitch rate q to pitch moment demand δ_{lon} frequency responses between hover and 250 kts. In hover, the pitch phugoid mode is at a frequency of $\omega_{ph} = 0.3$ rad/sec and is unstable. It is followed by a real pole at $\omega = -M_q = 2$ rad/sec. The regressive flap mode can be seen as the peak at $\omega = 12$ rad/sec (corresponding to the eigenvalue labeled “Regressive Flap + q ” in Figure 2.11), followed by the progressive flap mode at $\omega = 50$ rad/sec.

As airspeed increases, the phugoid mode decreases in frequency and becomes stable. The short period mode is composed of two real poles, one stable and one unstable (corresponding to the eigenvalues labeled Pitch Subsidence and Heave Subsidence in Figure 2.12) with increasing frequency as airspeed increases. As with the roll response, the regressive flap mode increases in frequency with increasing airspeed while the progressive flap mod decreases in frequency.

Figure 2.32 shows the yaw rate r to yaw moment demand δ_{ped} frequency responses

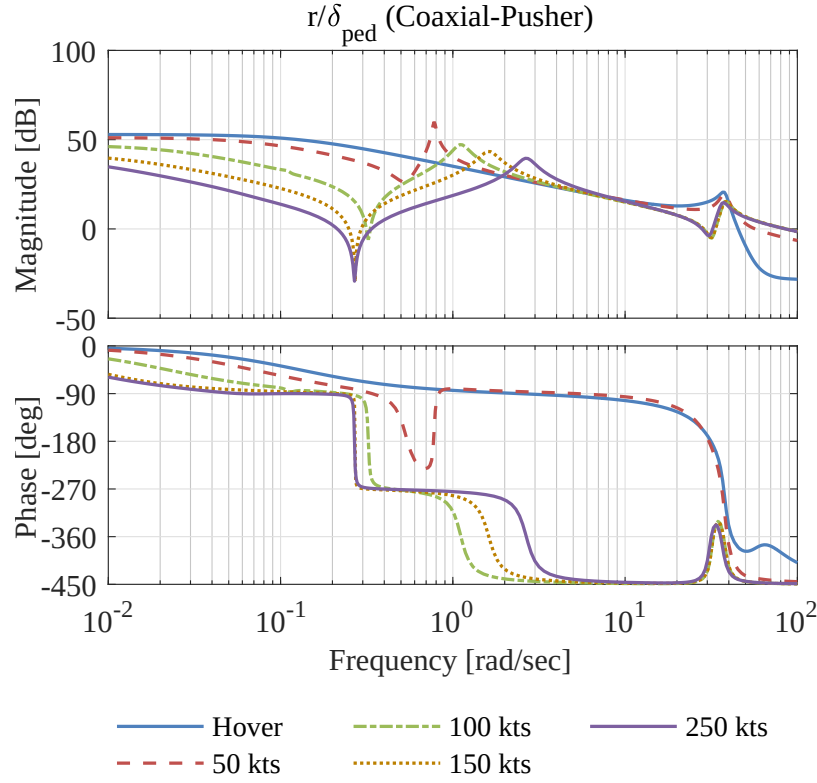


Figure 2.32. Yaw rate frequency response (coaxial-pusher, sea level).

between hover and 250 kts. In hover, below $\omega = 10$ rad/sec, the yaw rate response is first-order with a break frequency of $\omega = -N_r = 0.1$ rad/sec. One of the two collective lag modes (antisymmetric between the two rotors) can be seen at $\omega = 37$ rad/sec, due to differential collective $\Delta\theta_0$ being used for yaw control in hover. As airspeed increases, the Dutch roll mode can be seen in the yaw rate response as the peak between $\omega = 0.7 - 2$ rad/sec.

Figure 2.33 shows the z-axis body velocity w to symmetric collective θ_0 frequency responses between hover and 250 kts. In hover, below $\omega = 10$ rad/sec, the response is first-order with a break frequency of $\omega = -Z_w = 0.3$ rad/sec. The second collective lag mode (symmetric between the two rotors) can be seen at $\omega = 34$ rad/sec. As airspeed increases, the phugoid mode can be seen as the low-frequency distortion in the w frequency response, as well as the unstable short-period pole seen as the increase in phase with increasing frequency.

Figure 2.34 shows the x-axis body velocity u to pusher propeller collective $\theta_{0_{PP}}$ frequency responses between hover and 250 kts. In hover, the unstable longitudinal phugoid can be seen at $\omega_{ph} = 0.3$ rad/sec. In addition, the control power of the pusher propeller collective is lowest in hover, as seen by the low magnitude of the

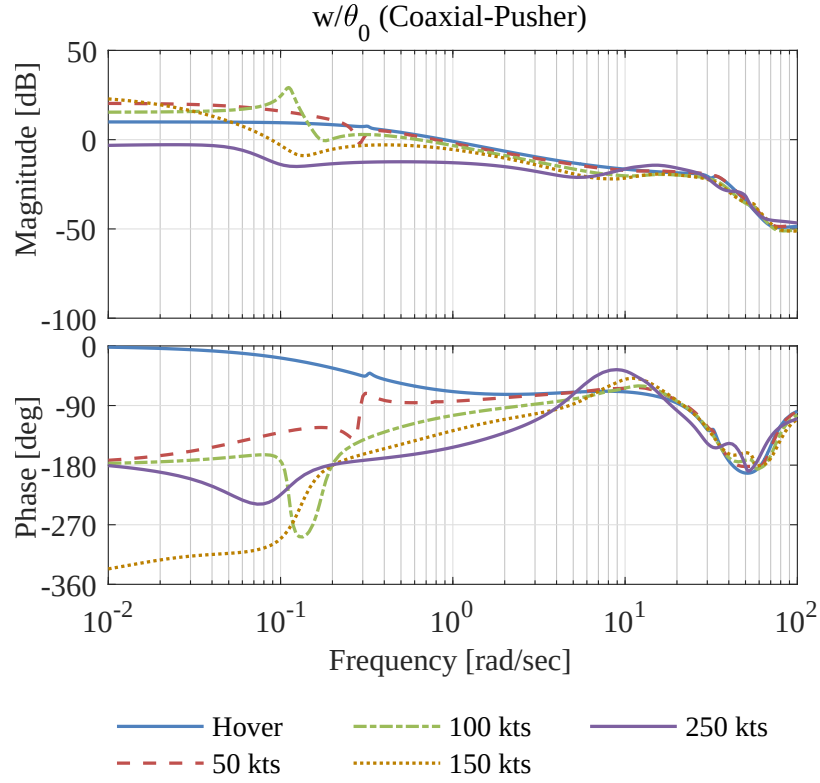


Figure 2.33. Vertical velocity frequency response (coaxial-pusher, sea level).

high-frequency asymptote as compared to the higher-airspeed responses. This also corresponds to the variation of the $X_{\theta_{0PP}}$ control derivative seen in Figure 2.8. As airspeed increases, the control power of the pusher propeller increases in magnitude and the phugoid mode decreases in frequency.

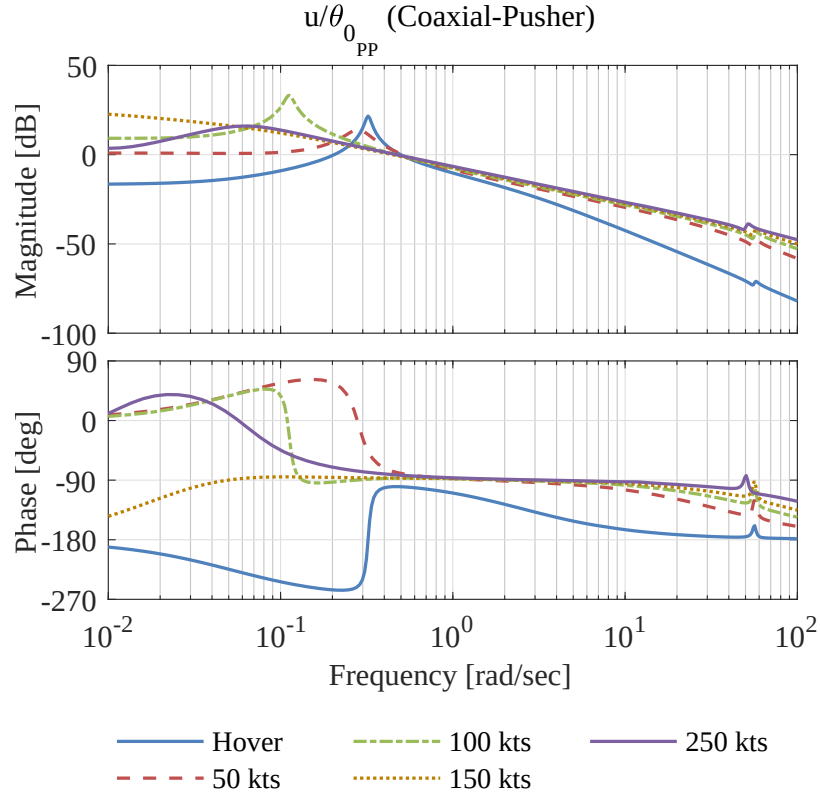


Figure 2.34. Longitudinal velocity frequency response (coaxial-pusher, sea level).

2.3.3 Tiltrotor

Figures 2.35 through 2.39 show the primary on-axis frequency responses of the tiltrotor stitched model. Figure 2.35 shows the roll rate p to roll moment demand δ_{lat} frequency responses between hover and 250 kts. In hover with nacelles at $\delta_{nac} = 90$ deg, the roll rate response exhibits the unstable lateral phugoid at $\omega = 0.5$ rad/sec, with the roll mode real pole at a similar frequency $1/T_r = -L_p = 0.5$ rad/sec. The tiltrotor roll mode is at a lower frequency than the coaxial-pusher model ($1/T_r = 5$ rad/sec) due to the significantly larger roll-axis inertia I_{xx} of the tiltrotor with its engines and rotors at the wingtips. The collective rotor Mode 1 can be seen as the peak at $\omega = 45$ rad/sec. At $V_{TAS} = 50$ kts with nacelles at $\delta_{nac} = 90$ deg, the lateral phugoid is still unstable, and above $\omega = 1$ rad/sec, the response is very similar to that in hover.

As airspeed increases and the nacelles are brought down, the roll-rate response looks like a typical fixed-wing response, with the ω_ϕ zeros and Dutch roll mode poles around $\omega = 1 - 3$ rad/sec. In addition, since differential collective is phased out for roll control at higher airspeeds and differential longitudinal cyclic is phased in (Figure 2.28), the collective rotor Mode 1 is not present at the higher speed roll rate responses. Instead, the progressive rotor Mode 1 can be seen at $\omega = 75$ rad/sec.

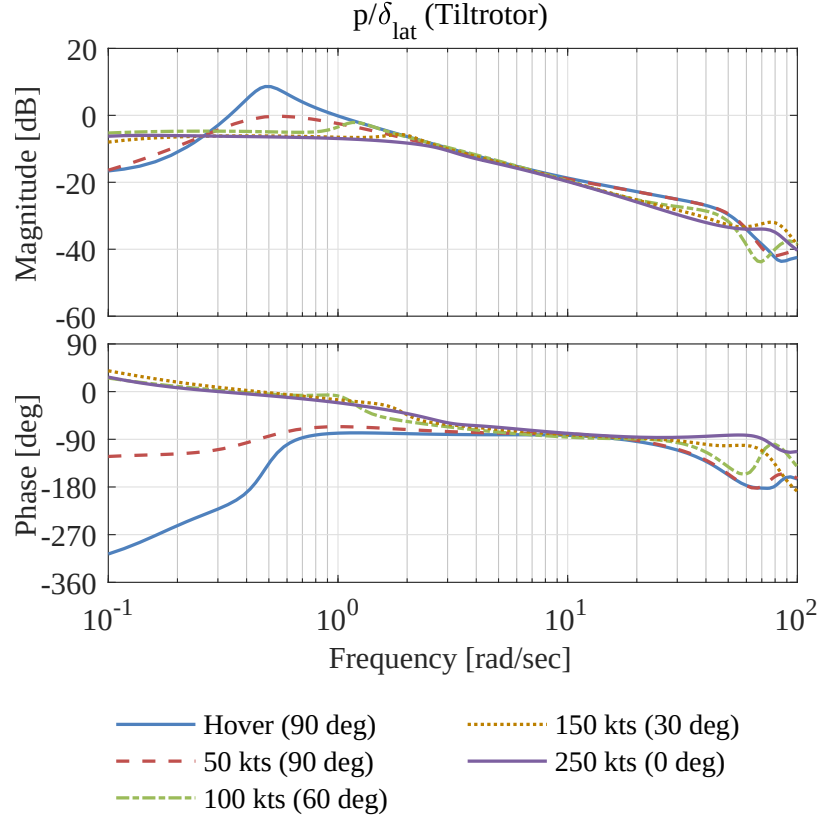


Figure 2.35. Roll rate frequency response (tiltrotor, sea level).

Figure 2.36 shows the pitch rate q to pitch moment demand δ_{lon} frequency responses between hover and 250 kts. In hover, the pitch phugoid mode is at a frequency of $\omega_{ph} = 0.6$ rad/sec and is unstable. It is followed by a real pole at $\omega = -M_q = 0.7$ rad/sec. The coupled rotor modes can be seen between $\omega = 10 - 100$ rad/sec. As airspeed increases to $V_{TAS} = 50$ kts, the phugoid mode decreases in frequency and becomes stable. Above $V_{TAS} = 50$ kts, the complex phugoid mode splits into two real poles. For nacelle angle $\delta_{nac} = 30$ deg, there is a low-frequency unstable phugoid pole. Unlike the coaxial-pusher, the short period mode of the tiltrotor remains stable at all airspeeds.

Figure 2.37 shows the yaw rate r to yaw moment demand δ_{ped} frequency responses between hover and 250 kts. In hover, below $\omega = 30$ rad/sec, the yaw rate response is first-order with a break frequency of $\omega = -N_r = 0.12$ rad/sec. A coupled rotor mode can be seen at $\omega = 80$ rad/sec. As airspeed increases, the Dutch roll mode can be seen in the yaw rate response as the peak between $\omega = 1 - 4$ rad/sec.

Figure 2.38 shows the z-axis body velocity w to symmetric collective θ_0 frequency responses between hover and 250 kts. In hover, below $\omega = 10$ rad/sec, the response

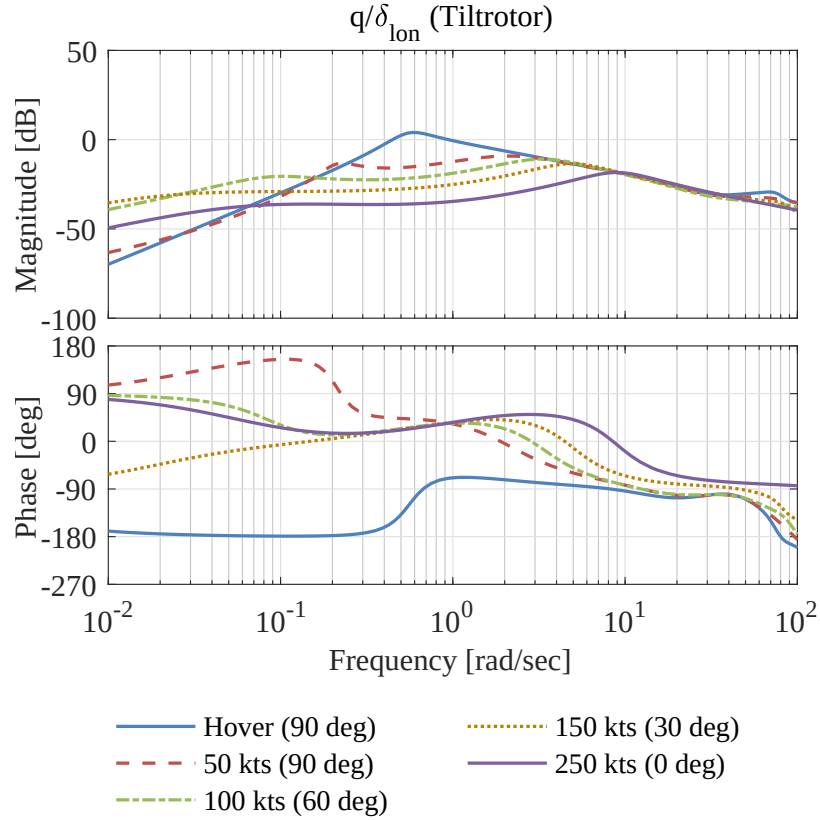


Figure 2.36. Pitch rate frequency response (tiltrotor, sea level).

is first-order with a break frequency of $\omega = -Z_w = 0.15$ rad/sec. The coupled rotor modes can be seen between $\omega = 10 - 100$ rad/sec. As airspeed increases, the phugoid mode can be seen as the low-frequency distortion in the z-axis body velocity w frequency response. Furthermore, as expected, collective becomes less effective at generating z-axis body velocity w as nacelle angle decreases, with the $\delta_{\text{nac}} = 0$ deg response have a magnitude around 60 dB lower than the $\delta_{\text{nac}} = 90$ deg response.

Figure 2.39 shows the x-axis body velocity u to symmetric collective θ_0 frequency responses between hover and 250 kts. In hover, the unstable longitudinal phugoid can be seen at $\omega_{\text{ph}} = 0.6$ rad/sec. For nacelle angles $\delta_{\text{nac}} = 90$ deg, increasing collective results in a short-term decrease in x-axis body velocity u (operating on the “back side” of the power curve), as seen by the phase between $\varphi = -360$ and $\varphi = -180$ deg at high frequency. Here, airspeed should be controlled with pitch attitude, like a typical helicopter. As the nacelle angle is decreased ($\delta_{\text{nac}} \leq 60$ deg), increasing collective results in a short-term increase in x-axis body velocity u (operating on the “front side” of the power curve), as seen by the phase between $\varphi = -180$ and $\varphi = 0$ deg at high frequency.

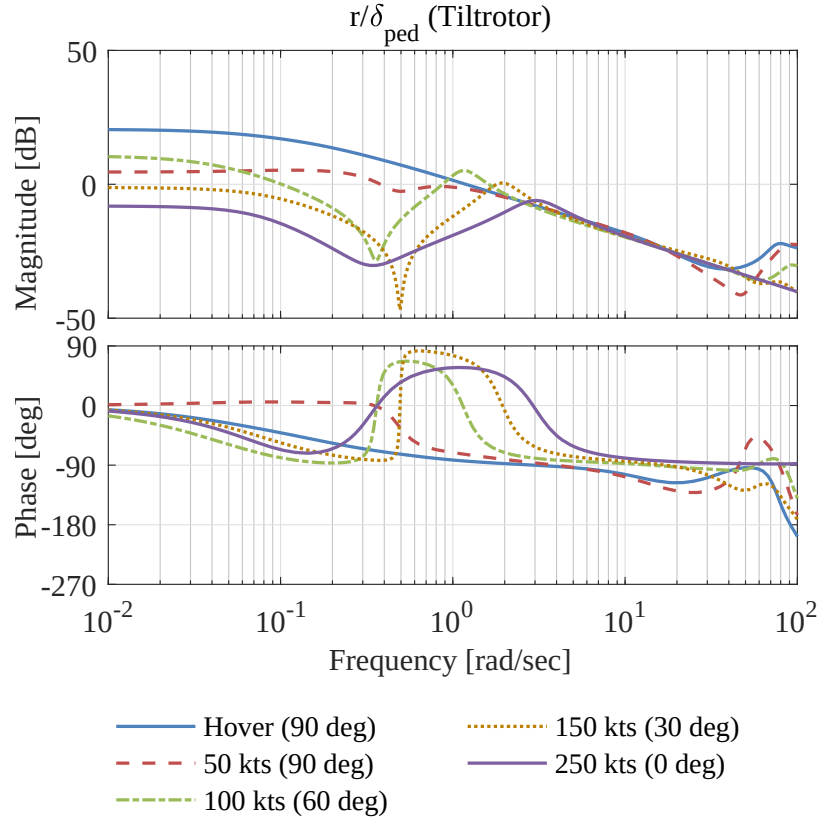


Figure 2.37. Yaw rate frequency response (tiltrotor, sea level).

2.4 Discussion

2.4.1 Expected Model Reliability and Accuracy

The generic coaxial-pusher and tiltrotor flight dynamics models presented here were developed to be used for control-system design, handling-qualities analysis, and piloted simulation research. Therefore, model accuracy in the mid-frequency range ($\omega = 1 - 10$ rad/sec) is most important. Any potential accuracy issues at low frequency will be masked by the control system when handling qualities are assessed for the closed-loop system. The modeling techniques used here benefit from extensive validation against flight data and other high-fidelity models (e.g., CAMRAD II) of multiple coaxial-pusher and tiltrotor aircraft (e.g., Refs. [68, 71]), which all show good matches in the mid-frequency range.

In addition, the flexible blade modeling in HeliUM provides accurate representation of the coupled flap/lag/torsion blade modes. The modes included in the linear models are expected to be accurate for use in control system design (e.g., notch filter

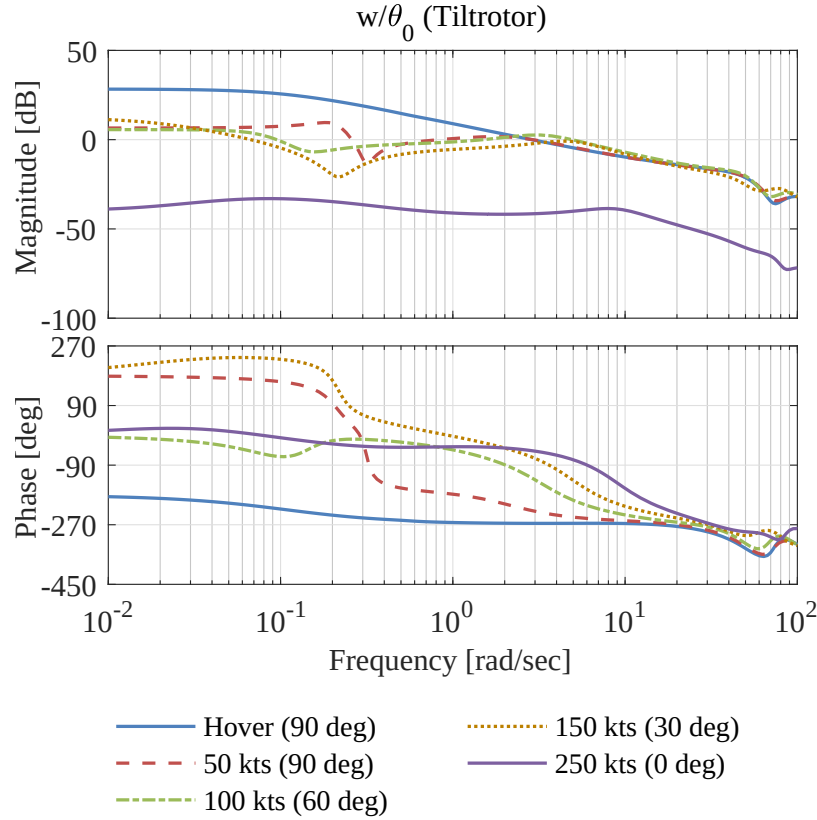


Figure 2.38. Vertical velocity frequency response (tiltrotor, sea level).

placement).

Finally, the stitched models developed from linear perturbation models and trim data are expected to be accurate and reliable for a wide range of motion (including large amplitude motion). Although the basis for the stitched models are the linearized models, only the linearized aerodynamic forces are retained in the stitched models. The gravity and kinematics are represented in their full nonlinear form. This type of modeling has been shown to be accurate for large-amplitude motion [96] (e.g., angular rate of about 60 deg/sec and normal accelerations of about 2 g).

2.4.2 Coaxial-Pusher

The coaxial-pusher configuration requires very large rotor cyclic control phasing to reduce off-axis flapping. To trim and linearize the model in HeliUM, a constant phase angle of $\Delta_{SP} = 70$ deg was used (similar to values used for XH-59A [4]). The optimal phase angle, however, is a function of airspeed. In the XH-59A, the phase angle Δ_{SP} was controlled in flight by the pilot through beeper switches [4], which led to high pilot workload. For the coaxial-pusher model described here, the pseudo inverse control

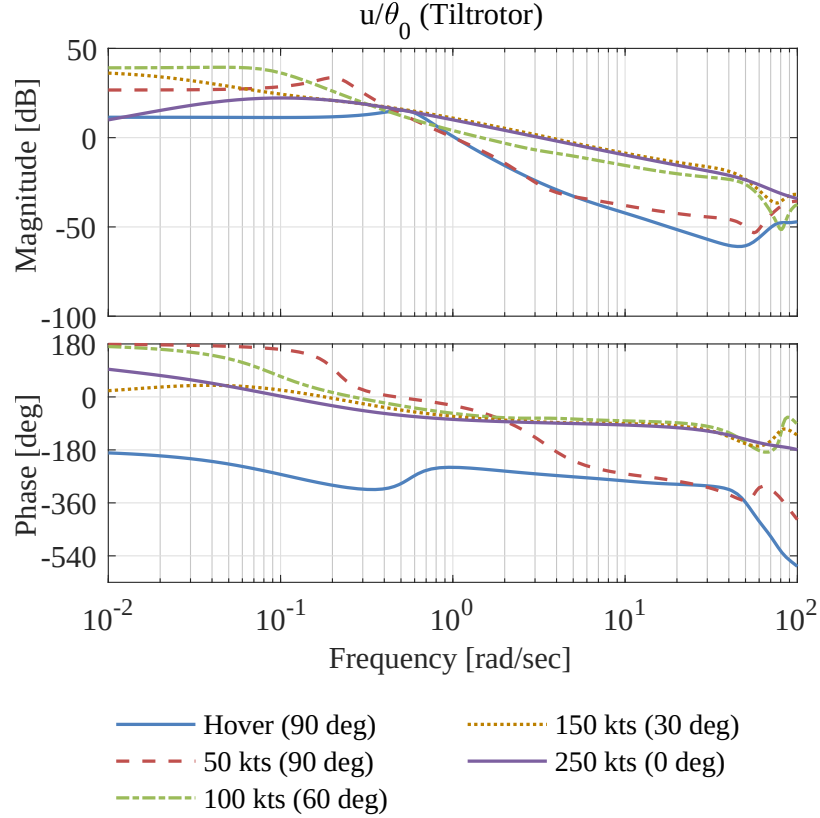


Figure 2.39. Longitudinal velocity frequency response (tiltrotor, sea level).

allocation gave the optimal phase angle at each airspeed, shown in Figure 2.27.

The coaxial-pusher has significant inter-axis coupling at hover/low-speed, as seen by its off-axis rigid body stability and control derivatives, despite having a symmetric configuration layout. Control coupling is accounted for by the pseudo inverse control allocation, however dynamic coupling may need to be accounted for in a control system.

The overall trim trends with airspeed for the coaxial-pusher model (shown in Sec. 2.1.2.2) match well with the X2 flight data presented in Ref. [71]. In addition, roll rate and pitch rate frequency responses of the X2 identified from flight data presented in Ref. [71] have similar characteristics to the coaxial-pusher frequency responses shown in Sec. 2.3.2. For example, the phase in the hover roll rate response p/δ_{lat} of the coaxial-pusher model (Figure 2.30) starts at $\varphi = -180$ deg at low frequency and then flattens out at $\varphi = -45$ deg at a frequency above the phugoid mode ($\omega \approx 1$ rad/sec). This is consistent with the flight-identified X2 frequency response shown in Figure 9 of Ref. [71].

The primary hover/low-speed yaw control (differential collective $\Delta\theta_0$) and primary

high-speed yaw control (rudder δ_{rud}) are both less effective in the mid-speed range ($V_{\text{TAS}} = 40 - 100$ kts). Therefore, two additional yaw controls are used to provide the necessary yaw authority. Differential longitudinal cyclic $\Delta\theta'_{1s}$ is used between $V_{\text{TAS}} = 40 - 100$ kts and pusher propeller monocyclic $\theta_{1\text{cPP}}$ is used between $V_{\text{TAS}} = 40 - 180$ kts.

The large peaks associated with the rotor modes in the roll rate response (Figure 2.30) between $\omega = 20 - 50$ rad/sec will impact the maximum achievable crossover frequency and gain margin of a control system in this axis. A lag or notch filter will likely be necessary to meet stability margin requirements [97].

2.4.3 Tiltrotor

Like the coaxial-pusher, the tiltrotor configuration is symmetric. In the case of the tiltrotor, this symmetry translates to decoupled lateral/directional and longitudinal/heave responses at all airspeeds, as similarly seen for the XV-15 tiltrotor [83].

The tiltrotor configuration does have large adverse roll due to rudder deflections at high speed, as a consequence of its high V-tail. Adverse roll is seen in the XV-15 as well, with its high H-tail configuration, however less than the tiltrotor presented here. For the XV-15 at $V = 170$ kts [83]:

$$\left. \frac{L_{\delta_r}}{N_{\delta_r}} \right|_{\text{XV-15}} = -0.32 \quad (2.21)$$

while, for the tiltrotor model presented here, that value is more than double:

$$\left. \frac{L_{\delta_r}}{N_{\delta_r}} \right|_{\text{Tiltrotor}} = -0.68 \quad (2.22)$$

This adverse roll should be accounted for through additional control mixing, potential use of differential collective $\Delta\theta_0$ for yaw control in airplane mode, or feedback control.

The tiltrotor frequency responses shown in Sec. 2.3.3 have similar characteristics to the flight-identified XV-15 frequency responses shown in Ref. [83] in the low- to mid-frequency range. For example, the tiltrotor pitch rate frequency response in cruise ($V_{\text{TAS}} = 250$ kts, $\delta_{\text{nac}} = 0$ deg) shown in Figure 2.36 has a well-damped phugoid mode, which is also seen in the XV-15 pitch rate frequency response in cruise shown in Figure 11.8 of Ref. [83].

2.5 Summary

This chapter described flight dynamics models of generic lift offset coaxial-pusher and tiltrotor aircraft developed in the comprehensive rotorcraft simulation code HeliUM. To use the models for control law design and real-time piloted simulation, a set of linear

models and trim data were extracted from HeliUM spanning the flight envelope of each aircraft. The linear models and trim data were combined into stitched simulation models that provide real-time simulation capabilities.

Chapter 3 |

Inner-Loop Control Law Design

Full-flight envelope control systems were developed for both the generic lift offset coaxial and tiltrotor aircraft described in Chapter 2. Linear models extracted from HeliUM were used to develop control system gain schedules using the Control Designer's Unified Interface (CONDUIT[®]) and design methods of Ref. 50. The control system for each aircraft was optimized to meet a wide range of stability, handling-qualities, and performance specifications throughout the flight envelope

This chapter provides a description of the flight control systems that were designed and optimized for both coaxial-pusher and tiltrotor aircraft. First, a description of the control law architecture used will be provided, including discussion of all major elements of the block diagram. Next, the flight-control and handling-qualities specifications used to drive the control system optimization will be given. Finally, after a brief discussion of the optimization strategy used, the results of the control system optimization will be provided for both aircraft.

3.1 Architecture

An explicit model following (EMF) control system architecture [50] was used for both aircraft, shown in block diagram form in Figure 3.1. The control system consists of five main elements: 1) Control Allocation, 2) Command Model, 3) Inverse Model, 4) Command Delays, and 5) Feedback. The following sections will describe each element in more detail.

3.1.1 Control Allocation

Since both aircraft have redundant bare-airframe controls, a weighted pseudo-inverse control allocation scheme is used to allocate the demanded roll, pitch, and yaw moments to each aircraft's (redundant) control actuator commands. The details of the control allocation are provided in Sec. 2.2.

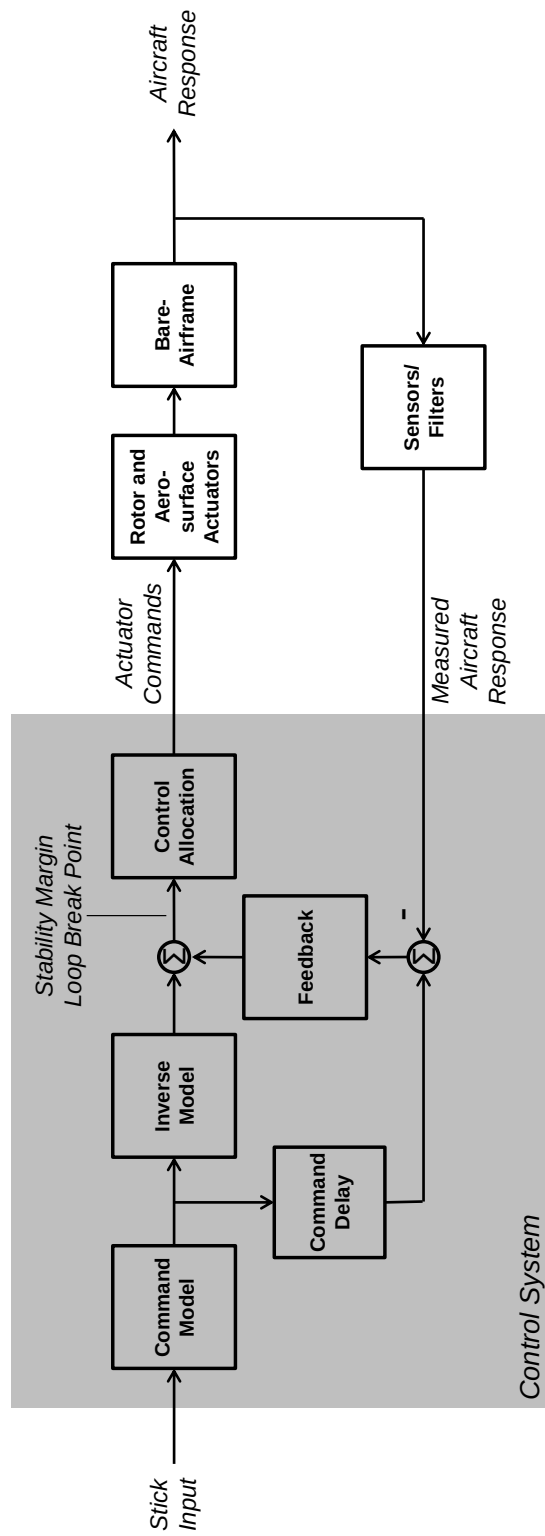


Figure 3.1. Top level block diagram representation of control system.

An advantage of a pseudo-inverse control allocation scheme is that it normalizes the bare-airframe control sensitivity. This removes the variations in overall feedback loop-gain as a function of dynamic pressure, and nacelle tilt angle in the case of the tiltrotor, from the feedback gains, as it is accounted for by the control allocation matrix.

3.1.2 Command Model

The command model in each axis sets the aircraft response characteristics to pilot inputs (i.e., response type, magnitude, and bandwidth). Table 3.1 lists the response type in each axis as a function of airspeed.

In the roll axis, a Rate Command/Attitude Hold (RCAH) response type is used throughout the flight envelope, with a first-order command model given by:

$$\frac{p_{\text{cm}}}{\delta_{\text{lat}_s}} = \frac{K_{\text{lat}_{\text{cm}}}}{\tau_{\text{lat}_{\text{cm}}} s + 1} \quad (3.1)$$

Note that rate command was chosen for the hover/low-speed to provide Level 1 handling qualities in a Good Visual Environment (GVE), however it is expected that an Attitude Command/Attitude Hold (ACAH) response type will be needed for Level 1 handling qualities in a Degraded Visual Environment (DVE) [98]. The EMF architecture used here provides the flexibility to easily change response types from rate command to attitude command for any future use of these control systems in DVE.

In the pitch axis, an RCAH response type is used between hover and $V_{\text{CAS}} = 200$ kts, with a first-order command model similar to the roll command model shown in Eq. 3.1. Above $V_{\text{CAS}} = 200$ kts, the response type changes to stability axes normal acceleration n_z command with angle of attack α hold, with a second-order command model given by:

$$\frac{n_{z_{\text{cm}}}}{\delta_{\text{lon}_s}} = \frac{K_{\text{lon}_{\text{cm}}} \omega_{\text{lon}_{\text{cm}}}^2}{s^2 + 2\zeta_{\text{lon}_{\text{cm}}} \omega_{\text{lon}_{\text{cm}}} s + \omega_{\text{lon}_{\text{cm}}}^2} \quad (3.2)$$

This is a typical fixed-wing response type (e.g., Ref. 99) which is useful for its simple ability to limit load factor and handle trim across a wide speed/loading range (trim $n_z = 1$ g regardless of speed/loading). As shown in Chapter 2, the bare-airframe dynamics of both the coaxial-pusher and tiltrotor aircraft have fixed-wing-like characteristics at high airspeed, and so fixed-wing response types at high speeds are a natural choice.

Between hover and $V_{\text{CAS}} = 100$ kts, pedals command yaw rate r , with a first-order command model similar to the roll command model shown in Eq. 3.1. In hover/low-speed, the hold mode on the pedals is direction (or heading) hold. The hold mode transitions to turn coordination above $V_{\text{CAS}} = 40$ kts. Above $V_{\text{CAS}} = 100$ kts, the response type commanded by the pedals changes to sideslip β command, with a

Table 3.1. Control System Response Types

Speed Range [KCAS]	Lateral Cyclic	Longitudinal Cyclic	Pedals	Thrust Control Lever
0–40	Rate Command/ Attitude Hold	Rate Command/ Attitude Hold	Rate Command/ Direction Hold	Rate Command/ Height Hold
40–100			Rate Command/ Turn Coordination	Direct Stick to Head
100–200			Sideslip Command/ Turn Coordination	
200–300		Normal Acceleration Command/An- gle of Attack Hold		

second-order command model similar to the normal acceleration command model in Eq. 3.2. This is another response type typical of fixed-wing aircraft (e.g., Ref. 100), which has the benefits of easily limiting sideslip at high speed and providing automatic turn coordination.

Finally, the collective/thrust control lever (TCL) commands vertical speed with height (or altitude) hold between hover and $V_{CAS} = 40$ kts. Above $V_{CAS} = 40$ kts, the TCL is open-loop and provides a direct stick-to-head response.

3.1.3 Inverse Model

The inverse model is used to generate feed forward inputs that command the aircraft to approximately follow the command model responses, and is composed of lower-

order inverses of the primary bare-airframe response in each axis. For each axis, the bare-airframe response of the commanded variable (listed in Table 3.1) is used. The lower-order inverse models are fit to the bare-airframe with the control allocation scheme included, such that a single-input/single-output (SISO) lower-order system is used for each axis.

For the tiltrotor roll rate p and yaw rate r responses, zeroth-order over first-order transfer function fits are used for all airspeeds, e.g., for roll:

$$\frac{p}{\delta_{\text{lat}}} = \frac{L_{\delta_{\text{lat}}}}{s - L_p} \quad (3.3)$$

For the coaxial-pusher, zeroth-order over first-order transfer function is used for the yaw rate r response and roll rate p response at airspeeds above $V_{\text{CAS}} = 100$ kts. However, at airspeeds below $V_{\text{CAS}} = 100$ kts, a zeroth-order over first-order transfer function does not fit the bare-airframe roll rate response well. Figure 3.2 shows the coaxial-pusher bare-airframe roll rate p response to demanded rolling moment δ_{lat} in hover. The phase of the bare-airframe response flattens out at $\varphi = -45$ deg between $\omega = 1 - 10$ rad/sec. This behavior is not captured by a zeroth-order over first-order fit as shown by the dashed red line in Figure 3.2. Therefore, a first-order over first-order fit:

$$\frac{p}{\delta_{\text{lat}}} = \frac{L_{\delta_{\text{lat}}}(s + 1/T_\phi)}{s - L_p} \quad (3.4)$$

is used for the coaxial-pusher roll axis from hover to $V_{\text{CAS}} = 100$ kts (dashed green line in Figure 3.2).

For both aircraft, from hover to $V_{\text{CAS}} = 60$ kts, a zeroth-order over first-order transfer function fit is used for the pitch rate q response. Above $V_{\text{CAS}} = 60$ kts, the bare-airframe pitch rate response of both aircraft is better represented by a classical fixed-wing first-order over second-order transfer function:

$$\frac{q}{\delta_{\text{lon}}} = \frac{M_{\delta_{\text{lon}}}(s + 1/T_{\theta_2})}{s^2 + 2\zeta_{\text{sp}}\omega_{\text{sp}}s + \omega_{\text{sp}}^2} \quad (3.5)$$

For sideslip and normal accelerations responses, a zeroth-order over second-order transfer function fit is used.

3.1.4 Command Delays

Command delays τ_{cmd} are used to synchronizes the commanded and actual states in time, before determining the error. The addition of command delays is typical in EMF control laws [50], and is done to account for higher-order dynamics and delays from the actuators, sensors, filters, and flight control computer processing time that are not accounted for by the inverse model.

Accounting for this additional delay before comparing the commanded states with

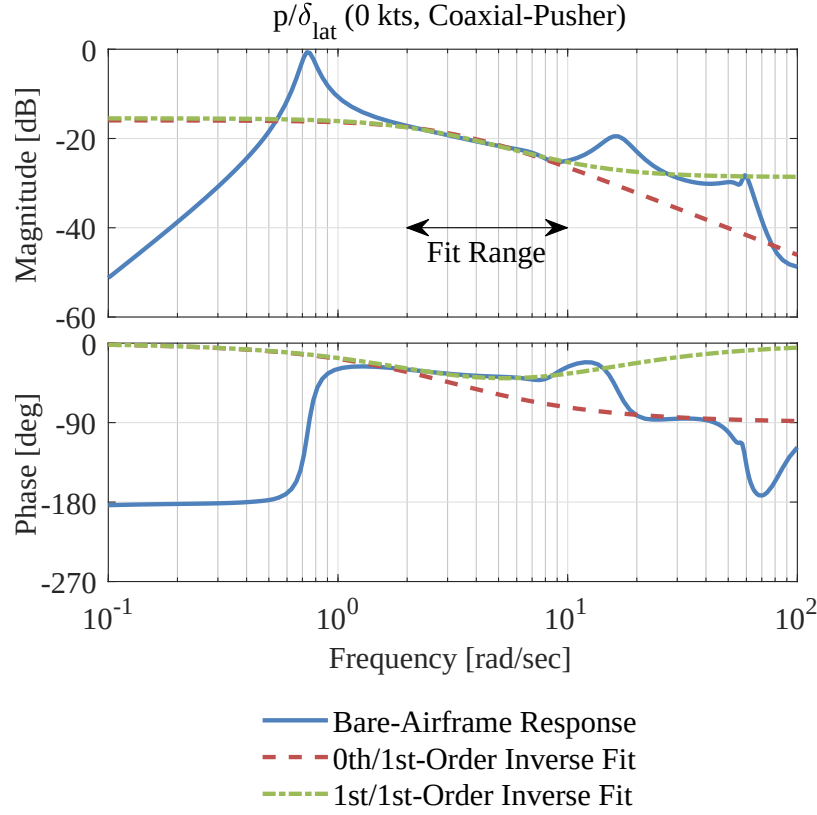


Figure 3.2. Lateral axis inverse model fit (hover, coaxial-pusher).

the actual states is useful to not overdrive the actuators. The addition of the command delays also reduces the amount of overshoot in the closed-loop end-to-end response with no added phase loss [99].

Table 3.2 shows the command delay τ_{cmd} values in each axis for both aircraft. As expected, similar command delay values were determined for both aircraft. The tiltrotor does have slightly lower command delay values in the lateral and directional axes, due to the use of differential collective to generate roll and yaw moments.

3.1.5 Feedback

Feedback is used to stabilize the aircraft, provide damping, account for inaccuracies in the lower-order inverse models, minimize the error between the commanded response and actual aircraft response due to cross-coupling and other disturbances, and provide hold capabilities and disturbance rejection. Rate, attitude, and attitude integral feedback gains are used.

Table 3.3 lists the feedback gains used in each axis as a function of airspeed for both the coaxial-pusher and tiltrotor. A common set of feedback gains is used for

Table 3.2. Command Delays

Axis	Coaxial-Pusher τ_{cmd} [sec]	Tiltrotor τ_{cmd} [sec]
Lateral	0.12 – 0.14	0.1 – 0.11
Longitudinal	0.13 – 0.14	0.1 – 0.13
Directional	0.12 – 0.16	0.07 – 0.15
Vertical	0.04 – 0.06	0.04 – 0.09

both aircraft with two exceptions:

1. In the mid- to high-speed range ($V_{\text{CAS}} > 50$ kts), the tiltrotor uses roll rate (K_p) and roll attitude (K_ϕ) feedback only in the lateral axis as is typical of fixed-wing lateral-axis control laws [100]. However, the coaxial-pusher control systems additionally includes roll attitude integral feedback (K_{ϕ_I}). This is because large, rapid changes in thrust (and therefore torque) on the pusher propeller result in a sustained roll disturbance. Roll attitude integral feedback was extended to the entire airspeed range for the coaxial-pusher to counteract this disturbance and minimize pusher propeller thrust-to-roll coupling during sustained acceleration or deceleration.
2. The tiltrotor includes a lateral acceleration integral gain ($K_{n_{y_I}}$) at high-speed ($V_{\text{CAS}} > 100$ kts), which is not used in the coaxial-pusher control systems. This gain was added to the tiltrotor control system to meet the turn coordination specification.

The feedback gains were optimized to meet a comprehensive set of stability, handling qualities, and performance specifications using a multi-objective optimization approach in CONDUIT[®] [50]. Section 3.2 lists the specifications used for the optimization.

3.1.6 Actuators, Sensors, and Filters

Accounting for all sources of time delays in the system is important during control system design and simulation testing to ensure that system performance is not pushed beyond what is capable. For this purpose, actuator, sensor, time delay, and filter models are included in both the linear control system design models as well as the nonlinear simulation models.

Rotor and aerosurface actuators are included in the block diagrams of both aircraft, and are represented by second-order systems with position and rate limits. Table 3.4 lists the actuator bandwidths and limits for the coaxial-pusher and tiltrotor actuators.

For both aircraft, the flight control computer (FCC) is assumed to be operating at 100 Hz. The FCC processing delay and digital-to-analog sample and hold delay

Table 3.3. Control System Feedback Gains

Speed Range [KCAS]	Lateral	Longitudinal	Directional	Vertical
0–40	K_p, K_ϕ, K_{ϕ_I}	$K_q, K_\theta, K_{\theta_I}$	K_r, K_ψ, K_{ψ_I}	$K_{V_z}, K_{V_{z_I}}$
40–100	$K_p, K_\phi, (K_{\phi_I})^*$		K_r	None
100–200			$K_{\dot{\beta}}, K_\beta, K_{\beta_I},$ $(K_{n_{y_I}})^\dagger$	
200–300		$K_{\dot{\alpha}}, K_\alpha, K_{n_{z_I}}$		

* Coaxial-pusher only

† Tiltrotor only

Table 3.4. Actuator Model Parameters

	Bandwidth [Hz]	Position Limit [deg]	Rate Limit [deg/sec]
<i>Both Aircraft</i>			
Main Rotor	8	± 20	40
Elevator	8	± 20	80
Rudder	8	± 30	70
<i>Coaxial-Pusher Only</i>			
Pusher Prop. Collective	4	$-10 - 100$	5
Pusher Prop. Monocyclic	4	± 10	20
<i>Tiltrotor Only</i>			
Aileron	8	± 40	80
Collective Trim	8	$0 - 35$	5
Nacelle	1.5	$0 - 95$	8

are each accounted for by a half time-step ($\tau = 5$ msec) delay [101], upstream of the actuator models.

Sensors are modeled as second-order systems with bandwidths of 5 Hz on all measured quantities. In addition, each sensor signal has a $\tau = 20$ msec delay, to account for sampling skew and bus transport delays.

Second-order complementary filters are used on aerodynamic measurements airspeed V , angle of attack α , and angle of sideslip β . The filters combine direct air data measurements of these values at low frequency (which provide good long-term measurements of these values, but contain noise for rapid changes) with estimates of their derivatives from inertial measurements at high frequency (provide good measurements of rapid changes, but can have long-term drift). Aerodynamic measurements are only used at high-speed ($V_{CAS} \geq 100$ kts).

Notch filters are used in the control system to avoid destabilizing the coupled fuselage rotor modes. For the coaxial-pusher, notch filters are used in the lateral axis ($\omega_n = 22$ rad/sec), in the longitudinal axis ($\omega_n = 12.5$ rad/sec), and in the directional axis ($\omega_n = 37$ rad/sec). These frequencies correspond to the coupled regressive flap/roll mode, coupled regressive flap/pitch mode, and differential collective lag mode, respectively. The locations of the rotor modes for the coaxial-pusher are shown as a function of airspeed in Figure 2.11. Similarly for the tiltrotor, notch filters are used in the lateral and longitudinal axes.

3.2 Specifications

A common, comprehensive set of stability, handling-qualities, and performance specifications, shown in Table 3.5, was used to optimize both coaxial-pusher and tiltrotor control laws. The specifications were divided into two categories—Tier 1 and Tier 2 specifications [50]. Tier 1 specifications are key flight-control and handling-qualities requirements that drive the design optimization, and are guaranteed to be met for an optimized design. Tier 2 specifications are those which are evaluated only at the end of the optimization. These are typically alternate requirements that give insight into the design and generally overlap with Tier 1 specifications. Because they are not evaluated during the optimization, due to computational time consideration, they are not always met.

3.2.1 Tier 1 Specifications

For the coaxial-pusher and tiltrotor control laws, Tier 1 specifications were selected primarily from SAE AS94900 [97] (stability margins) and ADS-33E [98] (handling-qualities requirements). For ADS-33E specifications with multiple boundaries based on the agility category MTE, boundaries for Aggressive Agility/Target Acquisition & Tracking were used.

Key Tier 1 specifications include absolute eigenvalue stability (CONDUIT[®] specification name EigLcG1), stability margins (StbMgG1), and Nichols margins (NicMgG1). These specifications ensure that the design is stable with sufficient stability margins for each control loop broken at the input to the control allocation matrix (“Stability Margin Loop Break Point” in Figure 3.1).

Standard stability margin boundaries of gain margin $GM \geq 6$ dB and phase margin $PM \geq 45$ deg [97] are used throughout the flight envelope with two exceptions. First, for both aircraft in hover/low-speed, more strict stability margin requirements of $GM \geq 6.6$ dB and $PM \geq 50$ deg are enforced. This is to enable future development of nested outer velocity and position loops, which degrade inner-loop stability margins [50]. The second exception to the standard stability margin boundaries is for the coaxial-pusher longitudinal axis at airspeeds greater than $V_{CAS} = 260$ kts. Although $V_H = 240$ kts for this generic coaxial-pusher pusher model, the control laws were designed up to $V_{CAS} = 300$ kts. As described in Chapter 2, the short-period mode of the coaxial-pusher is composed of two real poles, one of which is unstable. This pitch divergence becomes more unstable as airspeed increases, and the standard stability margin requirements cannot be met. Therefore, above $V_{CAS} = 260$ kts, $GM \geq 4$ dB and $PM \geq 40$ deg are enforced for the coaxial-pusher.

The model following cost specification (ModFoG2) compares the closed-loop frequency response in each axis with the frequency response of the command model. A cost function J_{MF} is computed based on the weighted difference in the magnitude and phase of the responses, and a value of $J_{MF} \leq 50$ is enforced, ensuring good command model following in each axis [50].

Disturbance rejection bandwidth (DRB, DstBwG1) and peak (DRP, DstPkG1) specifications [102, 103] were enforced in each axis for the appropriate hold variable. For example, in the pitch axis DRB and DRP were enforced for pitch attitude θ below $V_{CAS} = 200$ kts, and for angle of attack α for $V_{CAS} \geq 200$ kts. To ensure control system robustness to model uncertainty and improve disturbance rejection performance beyond the strict minimum Level 1/Level 2 boundaries in Refs. 102 and 103, a 110% design margin (DM) [50] was used on the roll and pitch axis DRB specification. The design margin moves the effective Level 1/Level 2 boundary into the Level 1 region by a percentage of the width of the Level 2 region. In this case, the effective DRB Level 1/Level 2 boundaries for roll and pitch attitudes were therefore $DRB_\phi = 1.4$ rad/sec and $DRB_\theta = 0.78$ rad/sec, instead of the specification values of $DRB_\phi = 0.9$ rad/sec and $DRB_\theta = 0.5$ [102, 103]. The increased roll and pitch DRB boundaries are shown as dashed lines in Figures 3.14 and 3.26 labeled “DM.”

A minimum crossover frequency specification (CrsMnG2) was included for each axis. This specification ensures that the frequency response for each control loop broken at the input to the control allocation matrix (“Stability Margin Loop Break Point” in Figure 3.1) has a crossover frequency ω_c above a specified value. The minimum crossover requirements are $\omega_{c_\phi} = \omega_{c_\theta} = 4.0$ rad/sec for roll and pitch, $\omega_{c_\psi} = 3.5$ rad/sec for yaw, and $\omega_{c_{V_z}} = 2.0$ rad/sec for heave. Setting the minimum crossover frequency is

an alternate way to ensure the control system is robust to off-nominal configurations, rather than explicitly including many off-nominal models in the evaluation of every specification, which is computationally expensive for optimization. It is especially useful in model following control law architectures such as the one used here, since most of the handling-qualities specifications can be met with a properly tuned command model and with less use of feedback. The attitude, sideslip, and vertical velocity integral gains (K_{ϕ_I} , K_{θ_I} , K_{ψ_I} , K_{ϕ_I} , K_{β_I} , and $K_{V_{z_I}}$) were all constrained to their respective proportional gains using the minimum crossover frequency specification values in each axis. The ratio of integral (K_I) to proportional (K_P) gain was constrained to be:

$$\frac{K_I}{K_P} = \frac{\omega_c}{5} \quad (3.6)$$

which ensures that the integral gain is effective, without overly degrading phase margin [50].

The Eigenvalue damping specification (EigDpG1) was used which evaluates the damping ratio of all closed-loop eigenvalues within a specified frequency range and compares them to the minimum required value.

The Open Loop Onset Point (OLOP, OlpOpG1) specification [104] is included to evaluate the control law's susceptibility to Pilot Induced Oscillations (PIOs) and limit cycle oscillations that can result from actuator rate limiting. Linear analysis methods ignore the nonlinear effects of actuator position and rate limiting. However, the OLOP specification is based on frequency domain describing function concepts, and is useful to include in the design process to not push the design beyond the actuator limits of the aircraft. Two OLOP specifications are used per axis for piloted and disturbance inputs for this two degree-of-freedom EMF architecture [50]. Since both coaxial-pusher and tiltrotor aircraft have redundant actuators, the OLOP specification in each axis was evaluated for the most critical actuator at each airspeed (first actuator to reach a rate limit for stop-to-stop inputs).

Piloted bandwidth specifications from ADS-33E (BnwRoH1–FrqHeH1) were included in each axis with appropriate boundaries based on flight condition (hover/low-speed and forward flight). For the hover/low-speed yaw attitude bandwidth specification (BnwYaH1), the reduced boundaries from Ref. 103 were used.

For high-speed flight ($V_{CAS} > 100$ kts), the ADS-33E flight path response to pitch attitude specification was included (FlpPiF1). Enforcing this specification ensures that the flight path or vertical rate response does not lag the pitch attitude response by more than $\varphi = 45$ deg at all frequencies below $\omega = 0.4$ rad/sec.

In addition, maximum achievable rates specifications (MaxRoH3–MaxHeH3) were included [98]. Finally, a set of cross-coupling requirements from ADS-33E were included in the Tier 1 specifications (CouPRH2–CouCPF1).

3.2.2 Tier 2 Specifications

Tier 2 specifications are used as “check only” and are not enforced by the control system optimization. In this case, Tier 2 specification include ADS-33E attitude quickness requirements as well as requirements from the military fixed-wing handling qualities requirements MIL-STD-1797B [17].

The ADS-33E attitude quickness specifications (QikRoH1–QikYaH2) are used as a check only for several reasons. First, it is a time domain specification which requires the computationally expensive simulation of the design model at each iteration of the control system optimization. Second, the results of this specification vary greatly with input shape, size, and duration, and would result in non-smooth gradients of the specification value with feedback gain selection, which is not a desirable characteristic in parametric optimization [50]. Target Acquisition & Tracking attitude quickness boundaries were used as an objective requirement, while the less stringent boundaries for All Other MTEs were used as a threshold requirement. For the yaw axis, the reduced attitude quickness boundaries from Ref. 103 were used.

For the MIL-STD-1797B specifications, boundaries for Category A flight phase (nonterminal flight phases that require rapid maneuvering, precision tracking, or precise flight-path control) and Class I aircraft (high-maneuverability air vehicles) are used.

Specifications from MIL-STD-1797B primarily consist of lower-order equivalent systems (LOES) specifications. For these specifications which are typical of fixed-wing aircraft, a lower-order transfer function fit is made to the closed-loop aircraft response, and the parameters of this equivalent transfer function are then evaluated against the specification boundaries. In the lateral/directional axes, individual LOES fits of the roll rate p response to lateral stick δ_{lat_s} input and sideslip β response to pedal δ_{ped_s} input are used. The fits were made over the frequency range of $\omega = 0.1 - 10$ rad/sec, and the LOES transfer functions are [83]:

$$\frac{p}{\delta_{\text{lat}_s}} = \frac{L_{\delta_{\text{lat}_s}} e^{-\tau_\phi s}}{s + 1/T_r} \quad (3.7)$$

$$\frac{\beta}{\delta_{\text{ped}_s}} = \frac{Y_{\delta_{\text{ped}_s}} e^{-\tau_\beta s}}{s^2 + 2\zeta_{\text{dr}}\omega_{\text{dr}}s + \omega_{\text{dr}}^2} \quad (3.8)$$

Two individual fits are used, instead of a simultaneous fit, since the responses are decoupled by design of the control system. Specifications on equivalent roll mode time constant (T_r), Dutch roll damping and frequency (ζ_{dr} , ω_{dr}), and time delays (τ_ϕ , τ_β) are enforced (FrqRoD4–TdlYaD1).

In the longitudinal axis, simultaneous LOES fits of the pitch rate q and normal acceleration n_z responses to longitudinal stick input δ_{lon_s} are used. The fits were made over the frequency range of $\omega = 0.5 - 7$ rad/sec using the LOES transfer functions

are [83]:

$$\frac{q}{\delta_{\text{lon}_s}} = \frac{M_{\delta_{\text{lon}_s}}(s + 1/T_{\theta_2})e^{-\tau_{\theta}s}}{s^2 + 2\zeta_{\text{sp}}\omega_{\text{sp}}s + \omega_{\text{sp}}^2} \quad (3.9)$$

$$\frac{n_z}{\delta_{\text{lon}_s}} = \frac{Z_{\delta_{\text{lon}_s}}e^{-\tau_{n_z}s}}{s^2 + 2\zeta_{\text{sp}}\omega_{\text{sp}}s + \omega_{\text{sp}}^2} \quad (3.10)$$

Specifications on equivalent Control Anticipation Parameter (CAP), short period damping (ζ_{sp}), and time delay (τ_{θ}) are enforced (CapPiL1 and TdPiL1).

A specification on LOES fit cost J_{LOES} is included (CosLoG1) to ensure the parameters of the LOES fit accurately represent the closed-loop aircraft dynamics. If the LOES fit cost is too high ($J_{\text{LOES}} > 50$), then the LOES specifications should not be evaluated, since the parameters of the LOES do not accurately represent the actual closed-loop aircraft dynamics.

In addition to the LOES specifications from MIL-STD-1797B listed above, five more specifications that do not require an LOES fit are used. The first two are the pitch attitude (BnwPiL4) and flight path (BnwFpL1) bandwidth specifications, which are evaluated directly from the closed-loop frequency responses. The next three MIL-STD-1797B specification that do not require an LOES fit are the time domain specifications that enforce time to bank requirements (roll performance, RolPfD1), pitch attitude dropback (DrpPiL1), and maximum lateral accelerating during steady turns (turn coordination, TrnCrD2).

3.3 Optimization Strategy

The control law parameters are gain scheduled as a function of airspeed for the coaxial-pusher. Parameters were determined at 10 kt increments from hover to $V_{\text{CAS}} = 300$ kts, for a total of 31 design points. For the tiltrotor, control law parameters are gain scheduled as a function of airspeed and nacelle angle. Figure 3.3 shows the tiltrotor design points (71 total design points), which are also in increments of 10 kts and span the conversion corridor.

At each design point, the command model parameters (stick gains and break frequencies) were hand-tuned to meet the piloted bandwidth, attitude quickness, and maximum achievable rate requirements. The parameters of the inverse model were determined from lower-order fits to the bare-airframe responses. The feedback gains listed in Table 3.3 were optimized in CONDUIT[®] using a multi-objective optimization approach to meet all of the Tier 1 Hard, Soft, and Summed Objective constraints [50] listed in Table 3.5. At any given design point, between 6–11 feedback gains were optimized simultaneously to meet all of the requirements. The following sections show the control system optimization results for both configurations.

Table 3.5. Control System Optimization Specifications

CONDUIT [®]					
	Spec Name	Description (Motivation)	Axis [*]	Speed Range [kts]	Source
Tier 1	Hard Constraints (Stability Requirements)				
	EigLcG1	Eigenvalues in L.H.P. (Stability)	All	All	Generic
	StbMgG1	Gain Phase Margin, loop broken at input to mixer/control allocation (Stability)	All	All	AS94900
	NicMgG1	Nichols Margins, loop broken at input to mixer/control allocation (Robust Stability)	All	All	Ref. 105
	Soft Constraints (Handling Qualities Requirements)				
	ModFoG2	Command model following cost (HQ)	All	All	Generic
	DstBwG1	Disturbance rejection bandwidth (Loads, Ride Quality)	All	All	Refs. 102, 103 [†]
	DstPkG1	Disturbance rejection peak (Loads, Ride Quality)	All	All	Refs. 102, 103
	CrsMnG2	Minimum $\omega_c \geq 4.0$ rad/sec (Robustness)	P,R	All	Generic
		Minimum $\omega_c \geq 3.5$ rad/sec (Robustness)	Y	All	Generic
		Minimum $\omega_c \geq 2.0$ rad/sec (Robustness)	H	0-40	Generic
		Eigenvalue Damping (HQ, Loads)	All	All	Generic
	OlpOpG1	Open Loop Onset Point, pilot input (PIO)	All	All	Ref. 104
		Open Loop Onset Point, disturbance input (PIO)	All	All	Ref. 104
	BnwRoH1	Roll attitude bandwidth and phase delay, hover (HQ)	R	0-60	ADS-33E
	BnwRoF1	Roll attitude bandwidth and phase delay, forward flight (HQ)	R	60-300	ADS-33E
	BnwPiH1	Pitch attitude bandwidth and phase delay, hover (HQ)	P	0-60	ADS-33E
	BnwPiF1	Pitch attitude bandwidth and phase delay, forward flight (HQ)	P	60-300	ADS-33E
	BnwYaH1	Yaw attitude bandwidth and phase delay, hover (HQ)	Y	0-60	ADS-33E [‡]
	BnwYaF1	Yaw attitude bandwidth and phase delay, forward flight (HQ)	Y	60-300	ADS-33E
	FrqHeH1	Heave response equivalent time constant, delay (HQ)	H	0-40	ADS-33E
	FlpPiF1	Flight path response to pitch attitude (HQ)	H	100-300	ADS-33E
	MaxRoH3	Maximum achievable roll rate, hover (HQ)	R	0-60	ADS-33E
	MaxRoF4	Maximum achievable roll rate, forward flight (HQ)	R	60-300	ADS-33E
	MaxPiH3	Maximum achievable pitch rate, hover (HQ)	P	0-60	ADS-33E
	MaxPiF1	Maximum achievable load factor, forward flight (HQ)	P	60-300	ADS-33E
	MaxYaH3	Maximum achievable yaw rate, hover (HQ)	Y	0-60	ADS-33E
MaxHeH3	Maximum achievable vertical rate, hover (HQ)	H	0-40	ADS-33E	
CouPRH2	Pitch-roll coupling (HQ)	P,R	All	ADS-33E	
CouYaH1	Yaw-collective coupling, hover (HQ)	Y,H	0-60	ADS-33E	
CouCPF1	Pitch-heave coupling, forward flight (HQ)	P,H	40-300	ADS-33E	
Summed Objective (Performance Requirements)					
CrsLnG1	Crossover Frequency (Act. Activity)	All	All	Generic	
RmsAcG1	Actuator RMS (Act. Activity)	All	All	Generic	
Tier 2	Check Only				
	QikRoH1	Roll attitude quickness, hover (HQ)	R	0-60	ADS-33E
	QikRoF1	Roll attitude quickness, forward flight (HQ)	Y	60-300	ADS-33E
	QikPiH1	Pitch attitude quickness, hover (HQ)	P	0-60	ADS-33E
	QikYaH2	Yaw attitude quickness, hover (HQ)	Y	0-60	ADS-33E [‡]
	FrqRoD4	LOES Roll model time constant (HQ)	R,Y	60-300	MIL-STD-1797B
	FrqDrD3	LOES Dutch roll frequency (ω_{dr}) (HQ)	R,Y	60-300	MIL-STD-1797B
	DmpDrD2	LOES Dutch roll damping (ζ_{dr}) (HQ)	R,Y	60-300	MIL-STD-1797B
	TdlRoD1	LOES Roll axis time delay (HQ)	R,Y	60-300	MIL-STD-1797B
	TdlYaD1	LOES Yaw axis time delay (HQ)	R,Y	60-300	MIL-STD-1797B
	CapPiL1	LOES Control Anticipation Parameters (HQ)	P	60-300	MIL-STD-1797B
	TdlPiL1	LOES pitch axis time delay (HQ)	P	60-300	MIL-STD-1797B
	CosLoG1	Max LOES Cost ($J \leq 10$) (HQ)	P,R,Y	60-300	Generic
	BnwPiL4	Bandwidth, phase delay (HQ)	P	60-300	MIL-STD-1797B
	BnwFpL1	Transient flight-path response (HQ)	P	60-300	MIL-STD-1797B
	RoLPdD1	Time to achieve bank angle (HQ)	R	60-300	MIL-STD-1797B
	DrpPiL1	Pitch dropback (HQ)	P	60-300	MIL-STD-1797B
	TrnCrD2	Turn coordination max n_y (HQ)	R,Y	60-300	AS94900

* R = Roll, P = Pitch, Y = Yaw, H = Heave

[†] 110% design margin applied to the roll and pitch axes specifications

[‡] Reduced boundaries from Refs. 103, 106 used

3.4 Optimization Results

3.4.1 Coaxial-Pusher

3.4.1.1 Point Design

Figures 3.4 through 3.9 show several of the key frequency responses of the optimized control system design for the coaxial-pusher at one flight condition ($V_{CAS} = 180$ kts). Figure 3.4 shows the roll axis broken-loop frequency response (for the loop broken

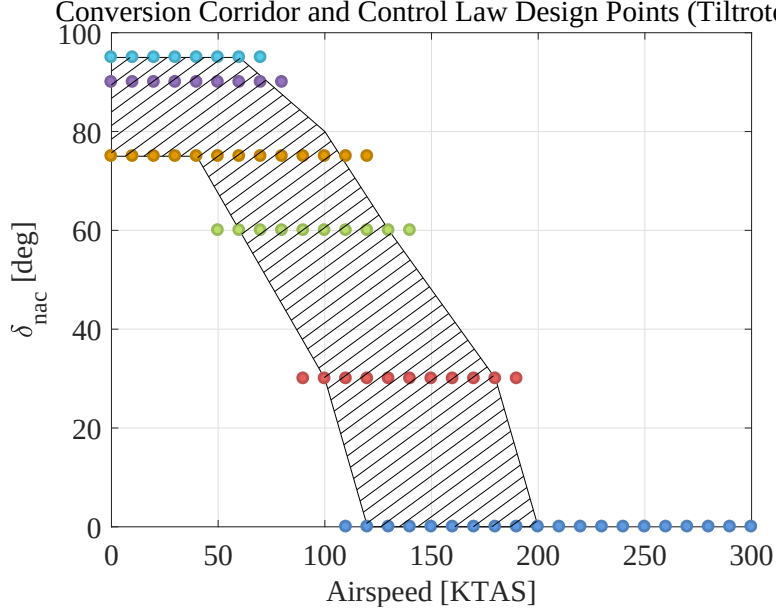


Figure 3.3. Conversion corridor (hatched region) and gain schedule design points (tiltrotor).

at the “Stability Margin Loop Break Point” in Figure 3.1). The crossover frequency of $\omega_{c_\phi} = 4.0$ rad/sec corresponds to the value enforced by the minimum crossover frequency specification in the roll axis, and the gain and phase margins are well within the Level 1 stability margin requirements [97].

Figure 3.5 shows the pitch axis broken-loop frequency response. As with the roll axis, the pitch axis crossover frequency ($\omega_{c_\theta} = 4.0$ rad/sec) corresponds to the value enforced by the pitch axis minimum crossover frequency specification. Since the pitch axis is unstable at high-speed (recall positive longitudinal static stability derivative M_w , Figure 2.6), there are two $\varphi = -180$ deg crossings. The critical (lower-magnitude) gain margin is at the low-frequency $\varphi = -180$ deg crossings ($\omega = 0.87$ rad/sec) and is negative (GM = -6.67 dB). This corresponds to a reduced gain margin, where a lower loop gain will not stabilize the unstable bare-airframe enough to meet the stability margin requirements. The higher-frequency gain margin is GM = 7.82 dB at $\omega = 14.22$ rad/sec and corresponds to an increased gain margin, limiting how high the loop gain can be increased before instability is reached.

Figure 3.6 shows the yaw axis broken-loop frequency responses, which has sufficient stability margins and a crossover frequency corresponding to the value enforced by the yaw axis minimum crossover frequency specification.

Figure 3.7 shows the closed-loop aircraft roll rate response to lateral stick input p/δ_{lat_s} as well as the delayed commanded roll rate response $p_{\text{cm}}/\delta_{\text{lat}_s}$. There is excellent command model tracking across a broad frequency range, with a model following cost of $J_{\text{MF}_p} = 4.73$ calculated between $\omega = 1 - 10$ rad/sec, well below the enforced limit of $J_{\text{MF}} \leq 50$.

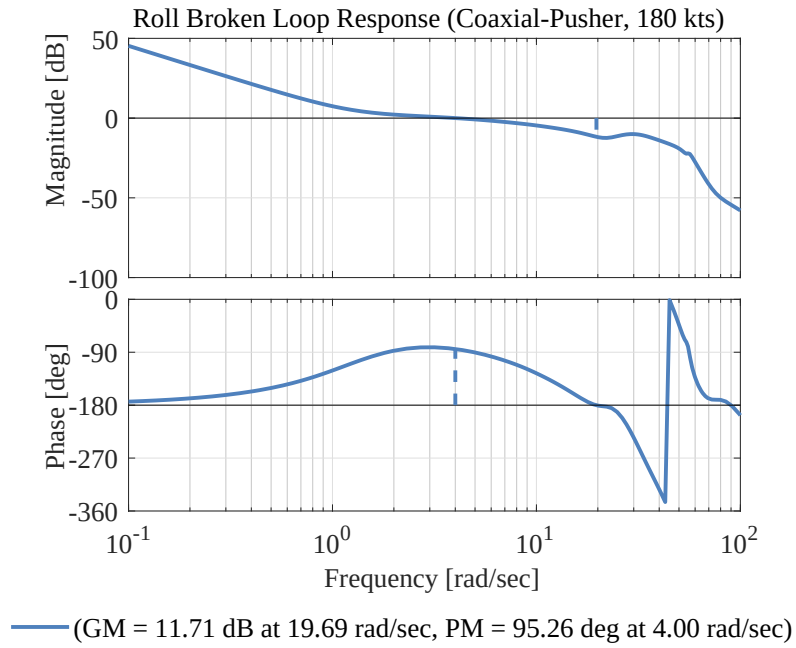


Figure 3.4. Roll axis broken-loop responses (coaxial-pusher, 180 kts).

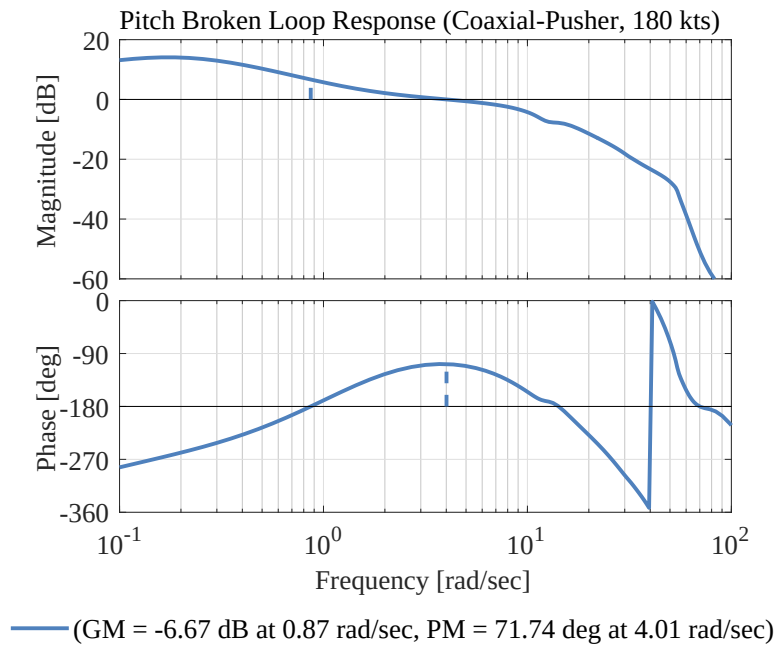


Figure 3.5. Pitch axis broken-loop responses (coaxial-pusher, 180 kts).

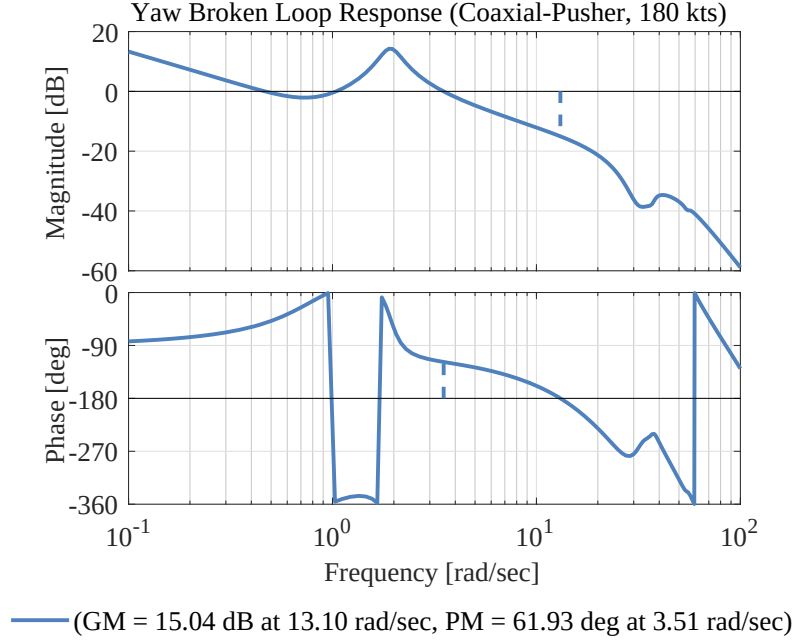


Figure 3.6. Yaw axis broken-loop responses (coaxial-pusher, 180 kts).

Figure 3.8 shows the closed-loop and commanded pitch rate response to longitudinal stick input and Figure 3.9 shows the closed-loop and commanded sideslip response to pedal input. As in the lateral axis, there is excellent model following performance in the longitudinal and directional axes, with model following costs of $J_{MF_q} = 1.51$ and $J_{MF_\beta} = 2.82$ calculated between $\omega = 1 - 10$ rad/sec.

Figure 3.10 shows a lateral stick 25% magnitude pulse response for the coaxial-pusher at $V = 180$ kts. There is excellent model following performance seen in both the roll rate p (top subplot) and bank angle ϕ (second subplot) responses, which track the first-order command model response (Eq. 3.1). The yaw rate response r (Figure 3.10, third subplot) is proverse and the sideslip response (Figure 3.10, fourth subplot) is small and goes to zero demonstrating the good turn coordination performance of the control laws.

Figure 3.11 shows a longitudinal stick 12.5% magnitude pulse response for the coaxial-pusher at $V = 180$ kts. As in the roll axis, there is excellent model following performance seen in both the pitch rate q (top subplot) and pitch angle θ (second subplot) responses. Since the pitch rate command model is first-order (similar to the roll rate command model shown in Eq. 3.1), there is nearly zero pitch attitude dropback. Furthermore, the angle of attack α (Figure 3.11, third subplot) and flight path γ (Figure 3.11, fourth subplot) responses are lower-order and well-damped, and match classical fixed-wing responses. For example, the flight path response γ can be seen to lag behind the pitch attitude response θ by a value of about 1 sec, which

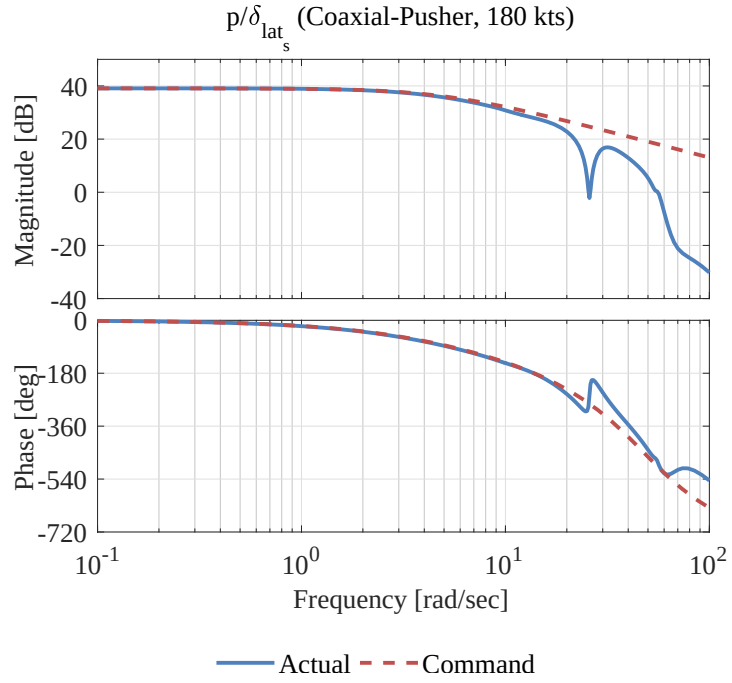


Figure 3.7. Closed-loop roll rate frequency responses (coaxial-pusher, 180 kts).

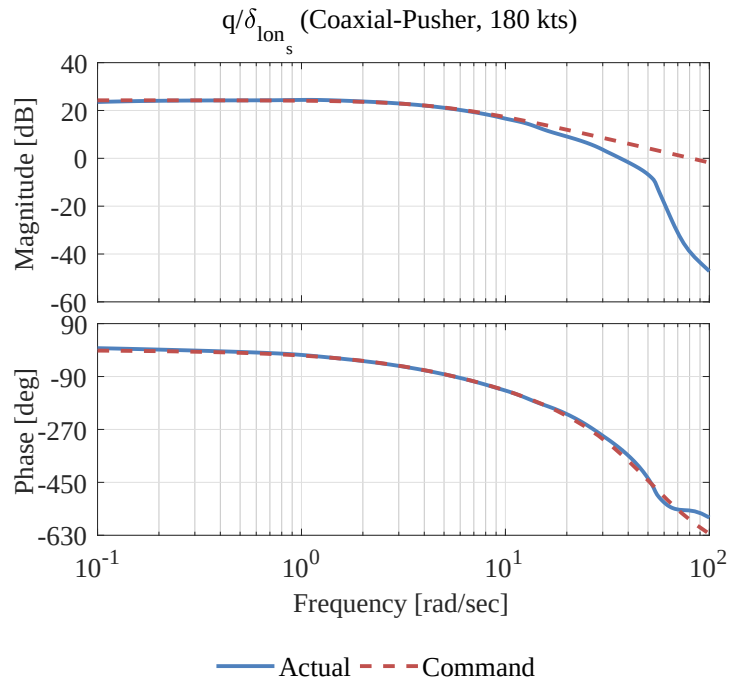


Figure 3.8. Closed-loop pitch rate frequency responses (coaxial-pusher, 180 kts).

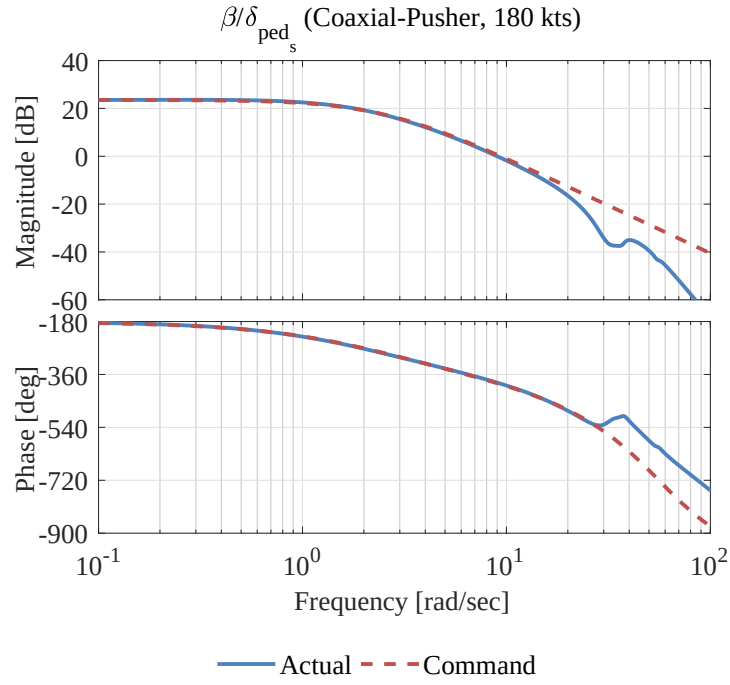


Figure 3.9. Closed-loop sideslip frequency responses (coaxial-pusher, 180 kts).

corresponds to the bare-airframe flight path-attitude lag value of $T_{\theta_2} = 0.89$ sec at this flight condition.

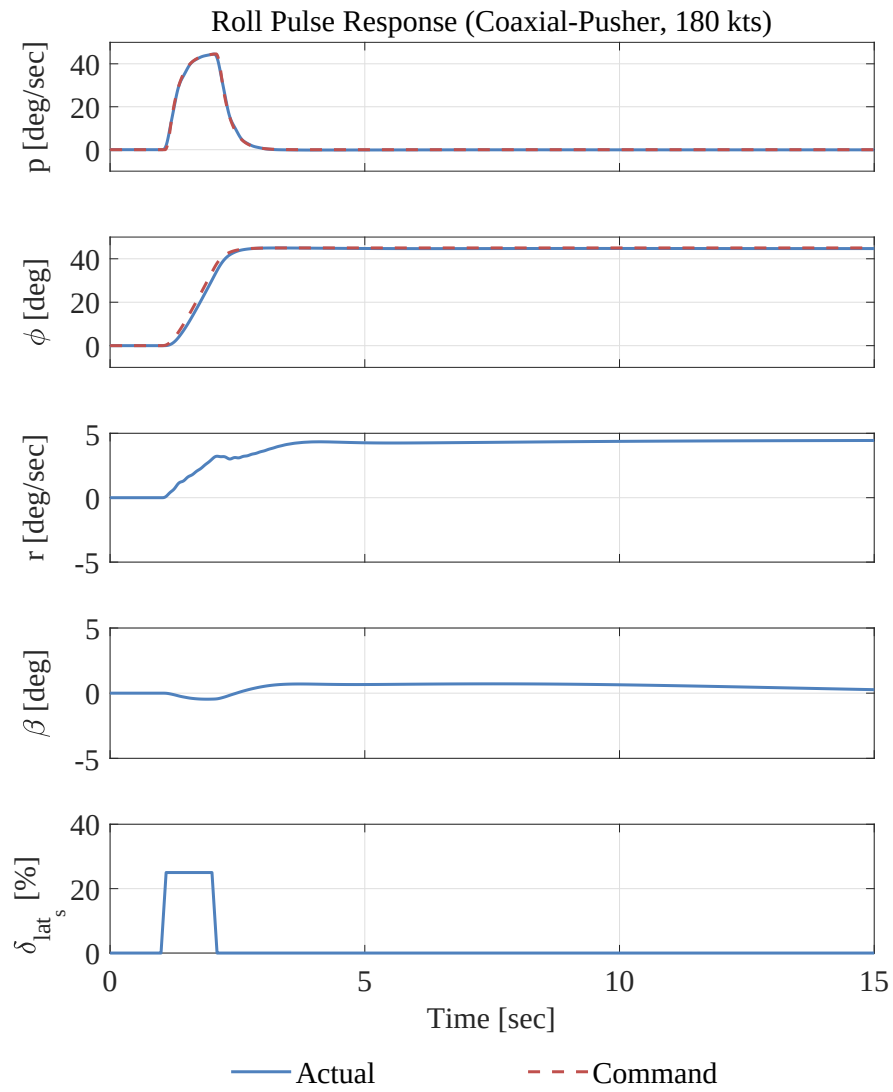


Figure 3.10. Lateral stick pulse response (Coaxial-Pusher, 180 kts).

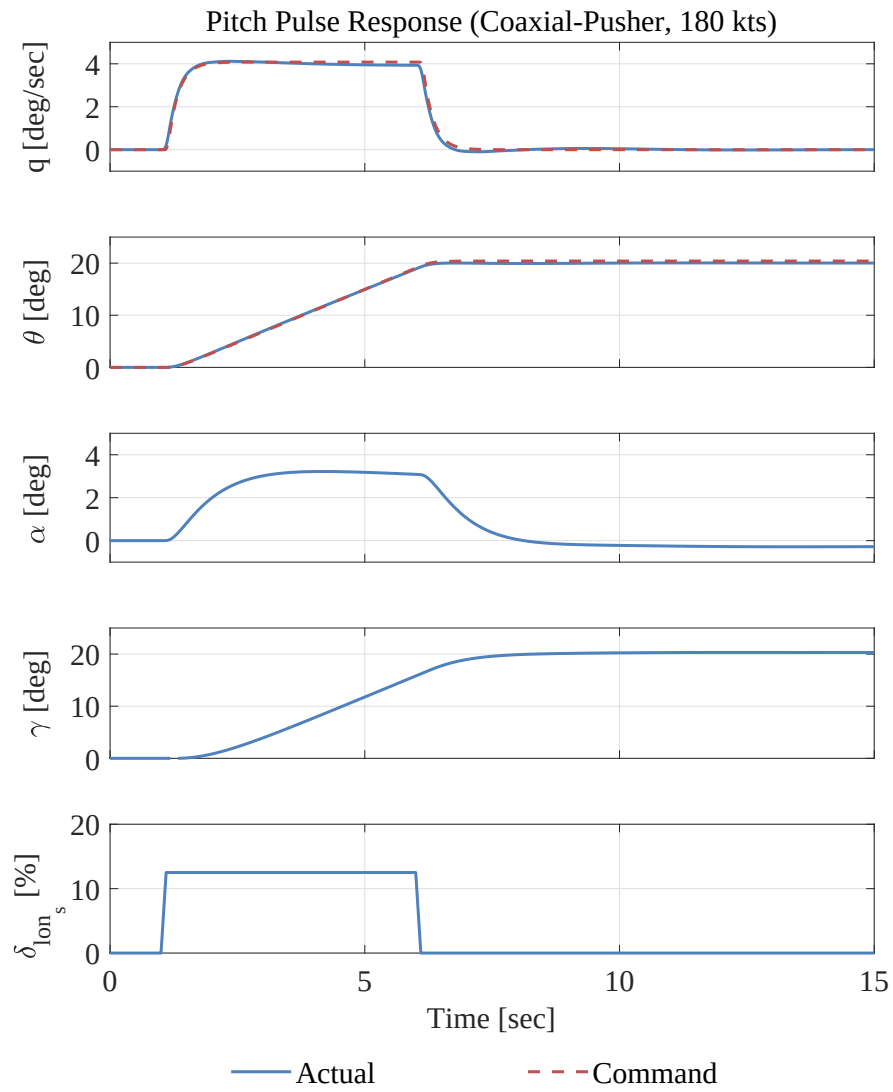


Figure 3.11. Longitudinal stick pulse response (Coaxial-Pusher, 180 kts).

3.4.1.2 Gain Schedule

Figure 3.12 shows the coaxial-pusher pitch rate (K_q) and angle of attack rate ($K_{\dot{\alpha}}$) feedback gains as a function of airspeed. Figure 3.13 shows the coaxial-pusher pitch attitude (K_θ) and angle of attack (K_α) feedback gains as a function of airspeed. Note that as described in Sec. 3.1.1, the pseudo-inverse control allocation scheme normalizes the variations in control sensitivity of the bare-airframe with airspeed. Therefore, those bare-airframe control sensitivity variations do not have to be accounted for by the variations in feedback gains shown in Figures 3.12 and 3.13.

Figures 3.12 and 3.13 demonstrate that CONDUIT[®] optimization resulted in smooth gain schedules. The only discontinuities are in changing response type from pitch rate command ($V_{CAS} < 200$ kts) to normal acceleration command ($V_{CAS} > 200$ kts), and from relaxing the stability margin requirements above $V_{CAS} = 260$ kts. The remaining feedback gains show similar trends.

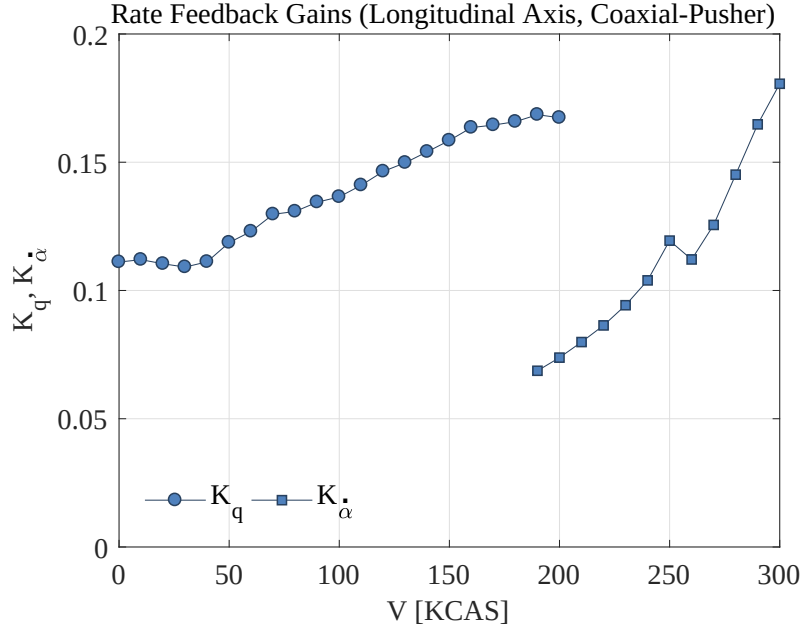


Figure 3.12. Pitch rate and angle of attack rate feedback gains (coaxial-pusher, optimized designs).

Figures 3.14 and 3.15 show several of the Tier 1 and Tier 2 specifications and the optimized design values for several airspeeds ranging from hover to $V_{CAS} = 240$ kts. Figure 3.14 shows the stability margin, minimum crossover frequency, DRB/DRP, and OLOP specifications in all four axes. These are all Tier 1 specifications, and so they are met for all designs through the optimization process. Several observations can be made:

- In the roll and yaw axes (Figure 3.14, first and third columns), stability margins are lowest in hover, and increase as airspeed increases. In contrast, in the pitch

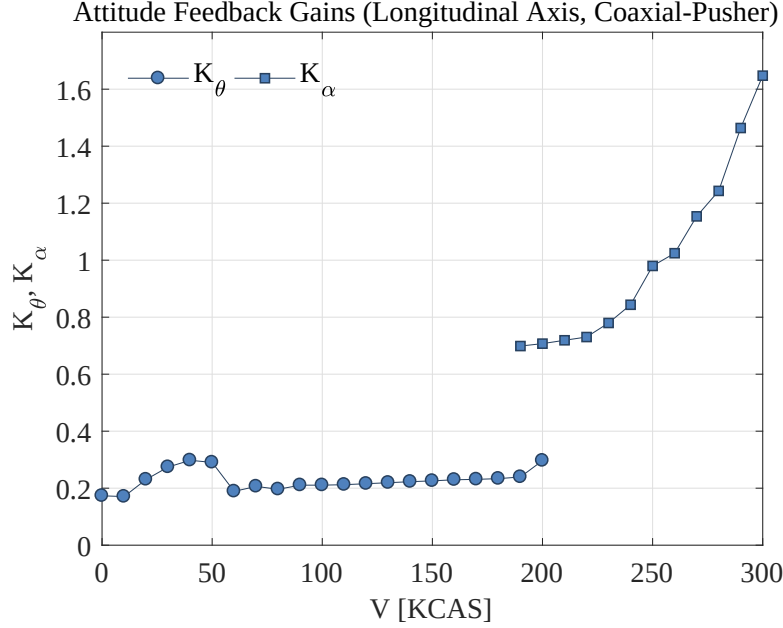


Figure 3.13. Pitch attitude and angle of attack feedback gains (coaxial-pusher, optimized designs).

axis (Figure 3.14, second column), the stability margins decrease as airspeed increases. At $V_{CAS} = 240$ kts (purple point), the pitch stability margins are in the corner of the stability margin requirement. This indicates that further reduction in phase margin cannot be counteracted by the addition of a lead-lag filter, since there is no gain margin to trade off for phase margin. And vice versa for further reduction in gain margin. Thus the relaxation of the longitudinal stability margin requirements above $V_{CAS} = 260$ kts.

- The resulting crossover frequencies ω_c in each axis are on the Level 1/Level 2 boundaries of the their respective minimum crossover frequency specifications. This is a result of the optimization approach used in CONDUIT[®], where the specifications are met with the minimum use of the actuators (i.e., the Pareto optimal solution) [50].
- The roll and pitch DRB requirements were optimized using a design margin of 110%, resulting in the values being above the minimum required boundaries (Figure 3.14, third row, first and second subplots). The effective Level 1/Level 2 boundary used in the control system optimization is shown as the dashed line labeled “DM.”
- The yaw axis OLOP specification is closest to the Level 1/Level 2 boundary in the low- to mid-airspeed range ($V_{CAS} = 60$ kts). This is because in this airspeed range the yaw control power of the coaxial-pusher is limited as differential torque on the rotors becomes less effective at generating yawing moment and the rudders

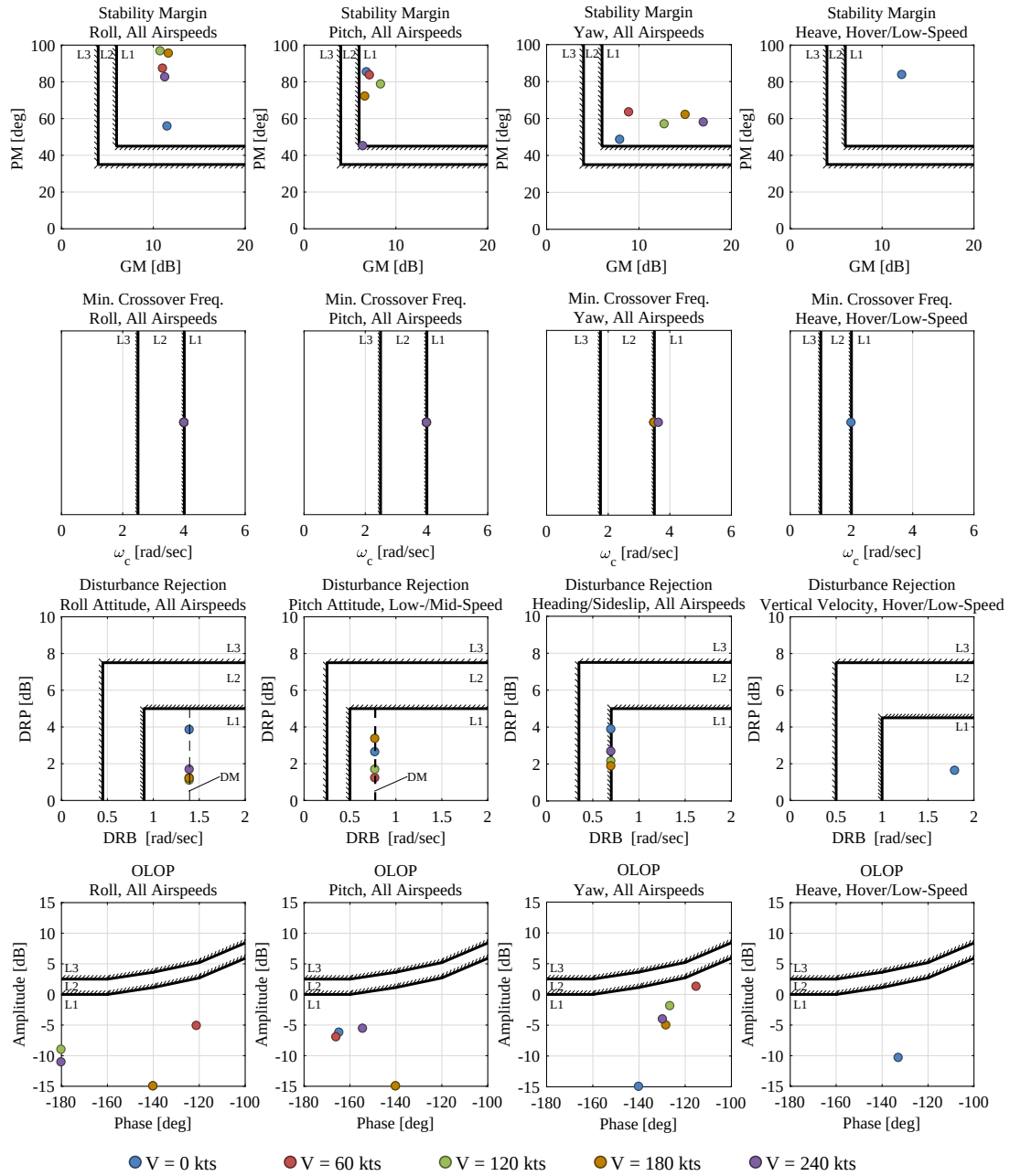


Figure 3.14. Example control system design specification and optimized design results (1/2, Coaxial-Pusher, hover–240 kts).

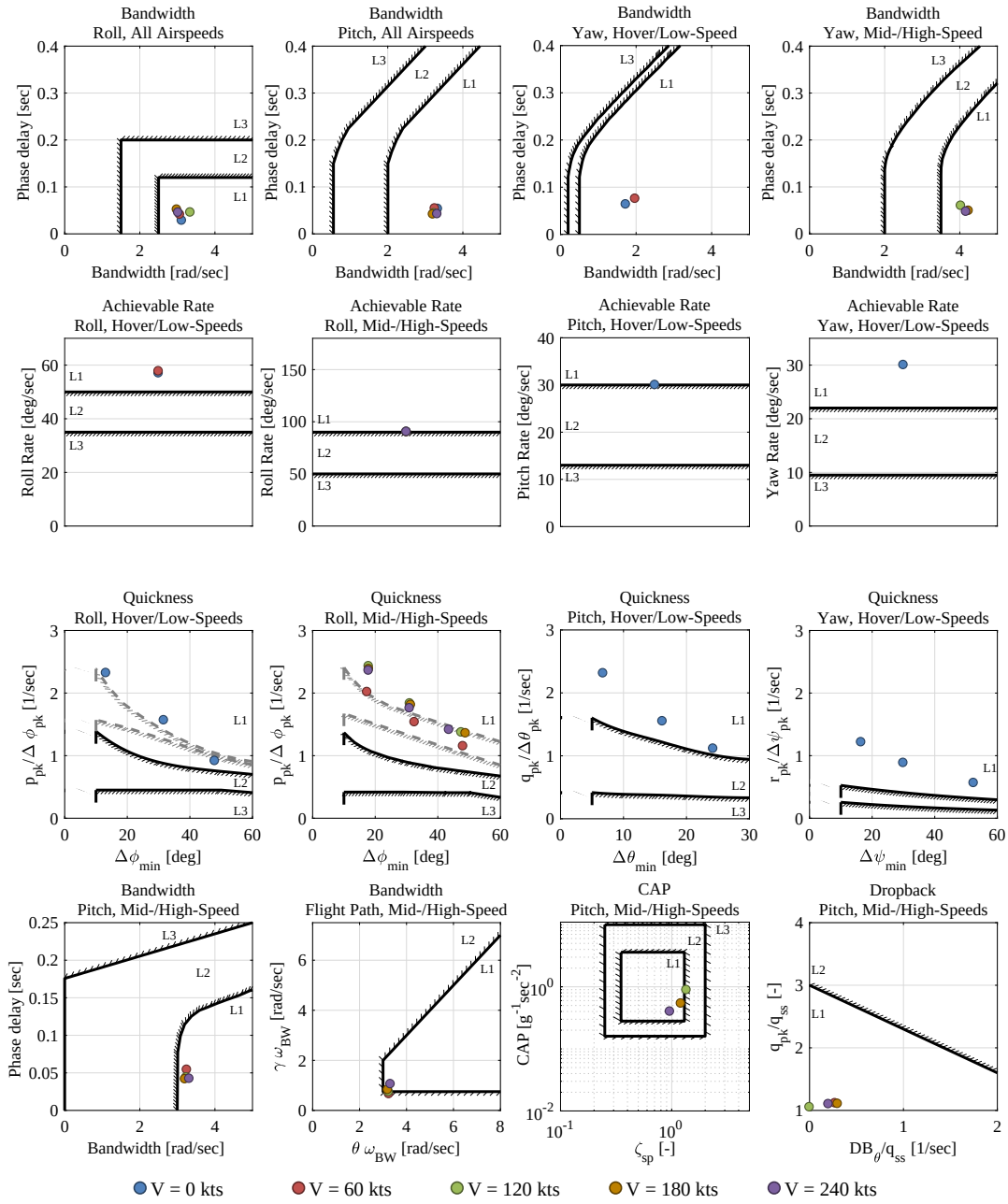


Figure 3.15. Example control system design specification and optimized design results (2/2, Coaxial-Pusher, hover–240 kts).

have not become fully effective yet. For this reason, pusher propeller monocyclic is used in this airspeed range as described in Chapter 2.

Figure 3.15 shows the piloted bandwidth/phase delay and maximum achievable rates for roll, pitch, and yaw (first two rows). These are Tier 1 requirements and are met for all designs through the optimization process and tuning of the command models. The last two rows of specification in Figure 3.15 are all Tier 2 requirements. Recall that these requirements are not enforced during the optimization, but are simply checked afterward. The third row of specifications in Figure 3.15 shows the attitude quickness specification for roll, pitch, and yaw. The roll hover/low-speed and mid-/high-speed specifications (Figure 3.15, third row, first two subplots) display two sets of boundaries. The gray dashed boundaries represent the Target Acquisition & Tracking attitude quickness requirement (used as an objective requirement), while the solid black boundaries represent All Other MTEs requirement (used as a threshold requirement). The low ($V_{CAS} < 60$ kts) and high-speed ($V_{CAS} > 120$ kts) designs meet the Target Acquisition & Tracking boundaries, while the mid-speed speed designs ($V_{CAS} \approx 60$ kts) meet the All Other MTEs boundaries.

Pitch and yaw attitude quickness specifications (Figure 3.15, third row, third and fourth subplots) are available for hover/low-speed only. In the pitch axis, the designs meet the Target Acquisition & Tracking boundaries, while in the yaw axis the designs meet the All Other MTEs boundaries.

The last row of specifications shown in Figure 3.15 consists of MIL-STD-1797B requirements: (from left to right) pitch attitude bandwidth, flight path bandwidth, CAP, and pitch attitude dropback. The pitch attitude bandwidth requirement from MIL-STD-1797B (Figure 3.15, fourth row, first subplot) is similar to the pitch attitude bandwidth requirement from ADS-33E (Figure 3.15, first row, second subplot), but requires a higher minimum bandwidth ($\omega_{BW} \geq 3.0$ rad/sec versus $\omega_{BW} \geq 2.0$ rad/sec) and has a lower maximum allowable value of phase delay. The MIL-STD-1797B requirement is geared towards faster-flying aircraft than the ADS-33E requirement, and might be more applicable to new advanced rotorcraft configurations with maximum speeds significantly faster than those of the legacy helicopters used to validate the ADS-33E specification boundaries.

The CAP requirement (Figure 3.15, fourth row, third subplot) is evaluated based on an LOES fit (Eqs. 3.9 and 3.10) of the closed-loop aircraft response. Note that the longitudinal LOES requirements were only evaluated for $V_{CAS} > 100$ kts, where the LOES fit cost was below $J_{LOES} = 50$. Although the CAP value is within the Level 1 requirement for all airspeeds, the equivalent short period damping ζ_{sp} is slightly too high for the mid-airspeed range, and is $\zeta_{sp} \geq 1.0$ for all airspeeds. A damping ratio $\zeta_{sp} > 1.0$ is expected for the RCAH designs ($V_{CAS} < 200$ kts) because the pitch command model, and therefore aircraft pitch rate response, is shaped like a first-order response. Therefore, the closed-loop transfer function in Eq. 3.9 must fit a first-order response, which happens when the denominator splits into two real poles, one at the command model break frequency and one canceling the bare-airframe $1/T_{\theta_2}$

zero. Although this may seem like an under-constrained fit (due to this pole-zero cancellation), the pole being canceled is constrained by the simultaneous fit of the normal acceleration response which does not have a zero in its transfer function fit (Eq. 3.10). The resulting large ζ_{sp} indicated that either this requirement is not applicable to this response type, or that a higher-order pitch command model, similar to a classical bare-airframe response (Eq. 3.9) may be needed to provide good handling qualities at higher airspeeds. Note that for normal acceleration n_z command portion of the ($V_{CAS} > 200$ kts), the normal acceleration command model is second order (matches the form of the transfer function in Eq. 3.10) and therefore the response is similar to a classical aircraft response. For this response type, the CAP requirement is applicable and is met for the coaxial-pusher.

Finally, the dropback requirement is shown in the last subplot in Figure 3.15 and is met for all design conditions.

3.4.2 Tiltrotor

3.4.2.1 Point Design

Figures 3.16 through 3.21 show several of the key frequency responses of the optimized control system design for the tiltrotor at one flight condition ($V_{CAS} = 180$ kts, $\delta_{nac} = 0$ deg). Figure 3.16 shows the roll axis broken-loop frequency response (for the loop broken at the “Stability Margin Loop Break Point” in Figure 3.1). The crossover frequency of $\omega_c = 4.1$ rad/sec corresponds to the value enforced by the minimum crossover frequency specification in the roll axis. Figures 3.17 and 3.18 show the pitch and yaw axis broken-loop frequency responses, respectively. The pitch and yaw axes both have crossover frequencies above their respective enforced minimum crossover frequency values. In addition, all loops have sufficient stability margins.

Figure 3.19 shows the closed-loop aircraft roll rate response to lateral stick input p/δ_{lat_s} as well as the delayed commanded roll rate response p_{cm}/δ_{lat_s} . There is excellent command model tracking across a broad frequency range, with a model following cost of $J_{MF_p} = 5.72$ calculated between $\omega = 1 - 10$ rad/sec, well below the enforced limit of $J_{MF} \leq 50$.

Figure 3.20 shows the closed-loop and commanded pitch rate response to longitudinal stick input and Figure 3.21 shows the closed-loop and commanded sideslip response to pedal input. As in the lateral axis, there is excellent model following performance in the longitudinal and directional axes, with model following costs of $J_{MF_q} = 0.92$ and $J_{MF_\beta} = 9.54$ calculated between $\omega = 1 - 10$ rad/sec.

Figure 3.22 shows a lateral stick 25% magnitude pulse response for the tiltrotor at $V_{CAS} = 180$ kts. There is excellent model following performance seen in both the roll rate p (top subplot) and bank angle ϕ (second subplot) responses, which track the first-order command model response (Eq. 3.1). The yaw rate response r (Figure 3.22, third subplot) is proverse and the sideslip response (Figure 3.22, fourth subplot) is

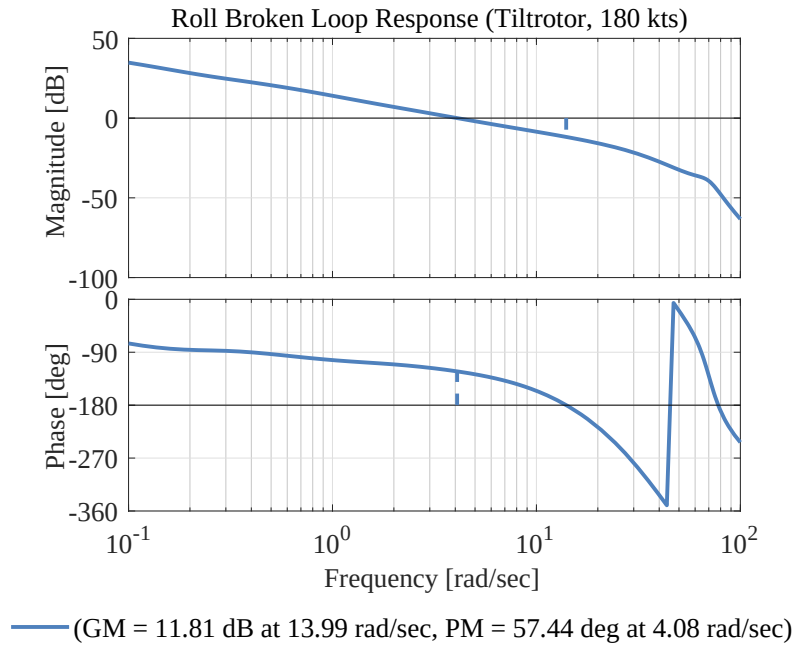


Figure 3.16. Roll axis broken-loop responses (tiltrotor, 180 kts).

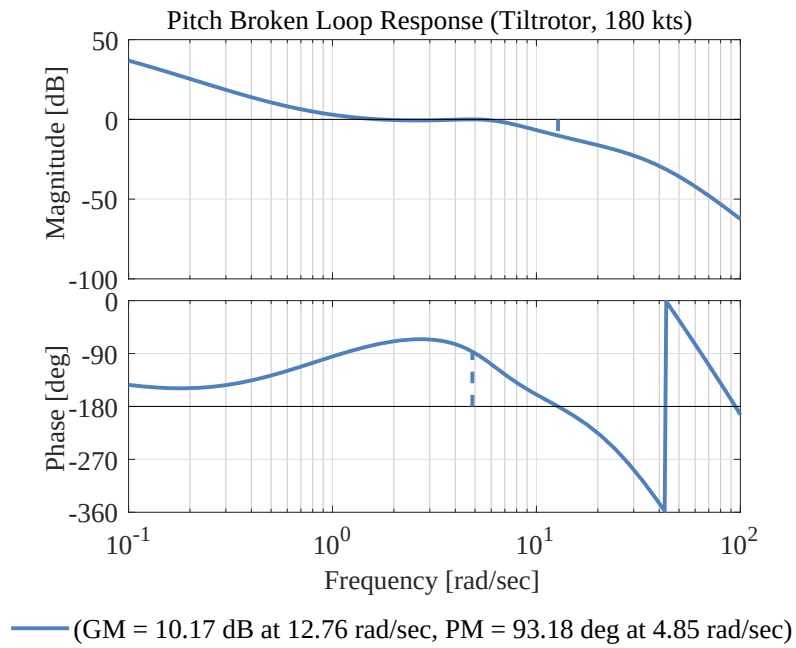


Figure 3.17. Pitch axis broken-loop responses (tiltrotor, 180 kts).

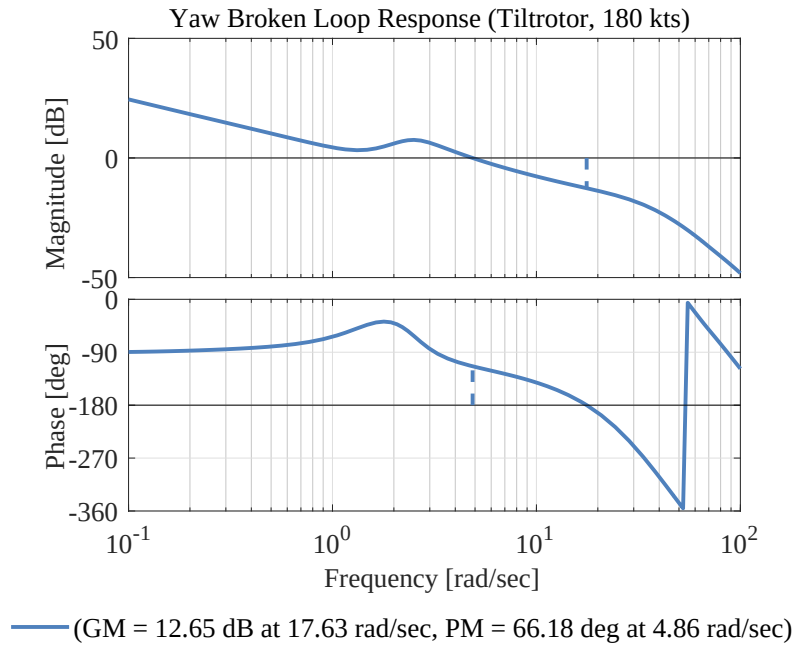


Figure 3.18. Yaw axis broken-loop responses (tiltrotor, 180 kts).

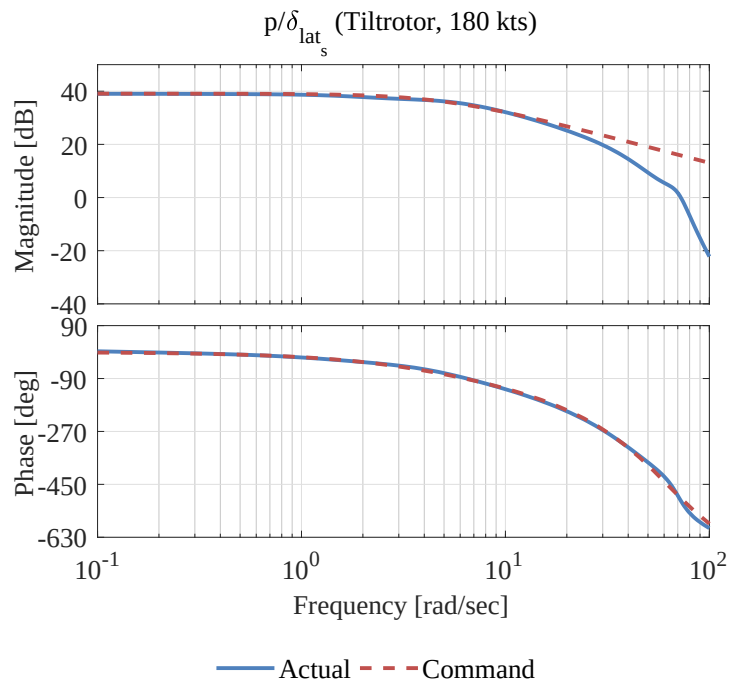


Figure 3.19. Closed-loop roll rate frequency responses (tiltrotor, 180 kts).

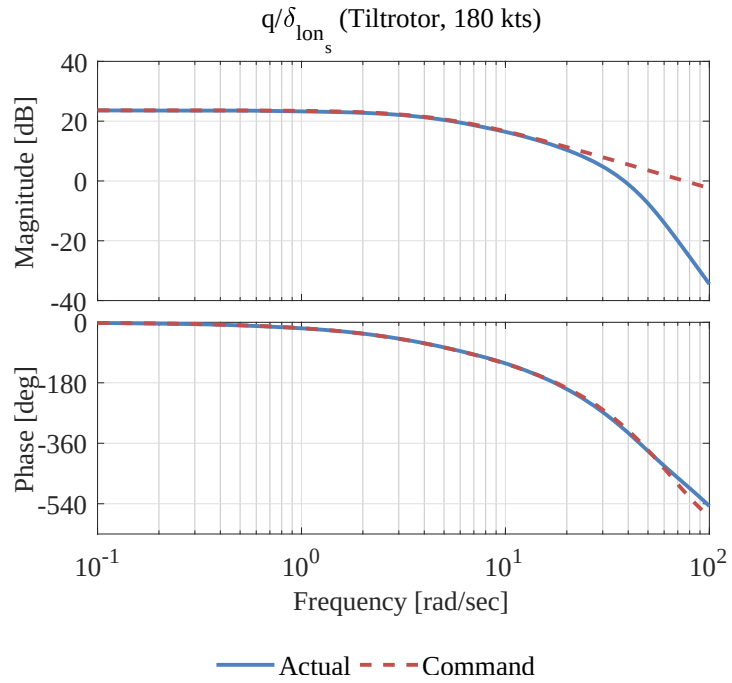


Figure 3.20. Closed-loop pitch rate frequency responses (tiltrotor, 180 kts).

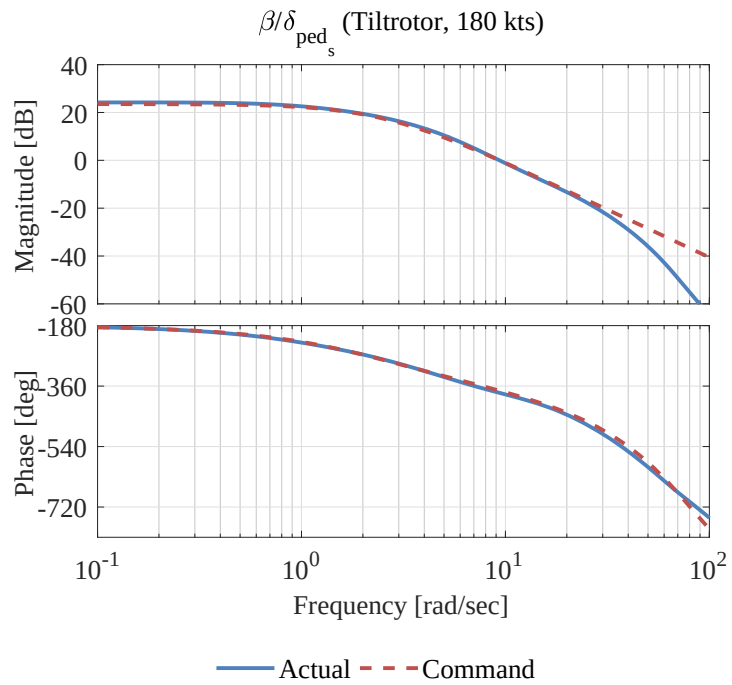


Figure 3.21. Closed-loop sideslip frequency responses (tiltrotor, 180 kts).

small and goes to zero demonstrating the good turn coordination performance of the control laws.

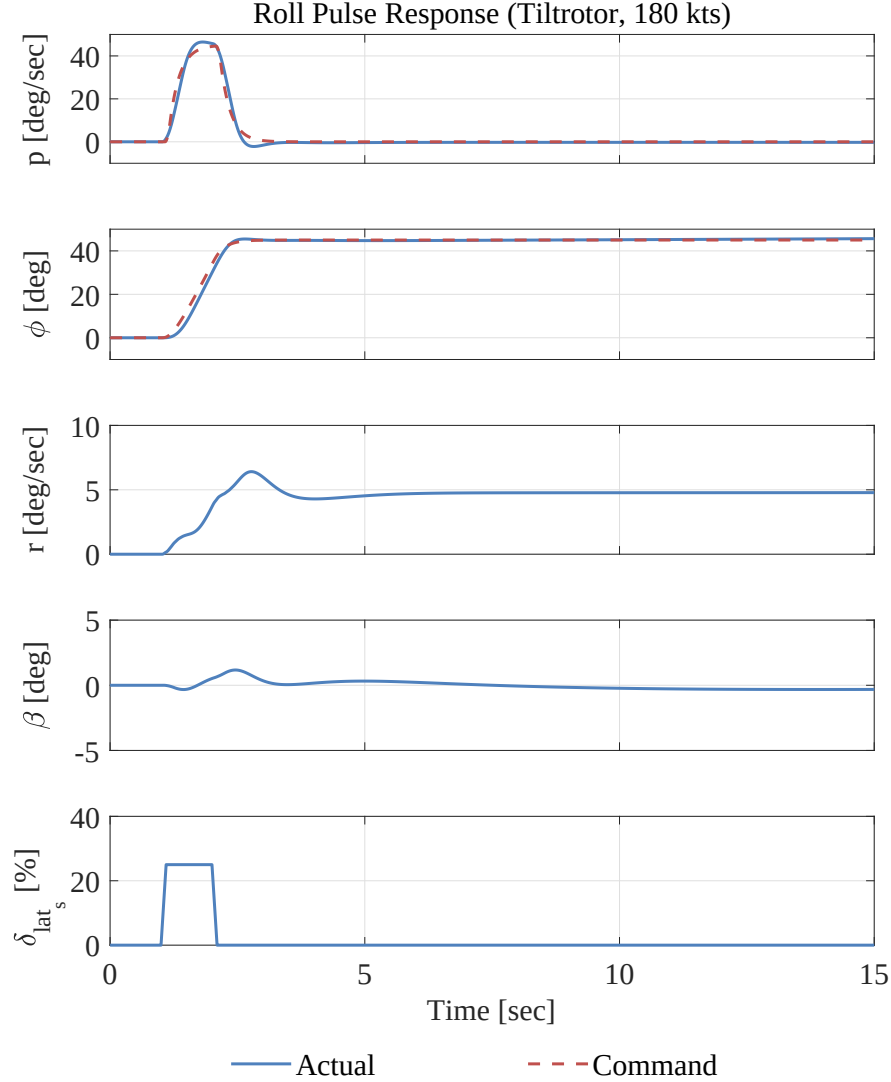


Figure 3.22. Lateral stick pulse response (tiltrotor, 180 kts).

Figure 3.23 shows a longitudinal stick 12.5% magnitude pulse response for the coaxial-pusher at $V_{CAS} = 180$ kts. As in the roll axis, there is excellent model following performance seen in both the pitch rate q (top subplot) and pitch angle θ (second subplot) responses. Since the pitch rate command model is first-order (similar to the roll rate command model shown in Eq. 3.1), there is nearly zero pitch attitude dropback. Furthermore, the angle of attack α (Figure 3.23, third subplot) and flight path γ (Figure 3.23, fourth subplot) responses are lower-order and well-damped, and match classical fixed-wing responses. For example, the flight path response γ can be

seen to lag behind the pitch attitude response θ by a value of about 1.25 sec, which corresponds to the bare-airframe flight path-attitude lag value of $T_{\theta_2} = 1.25$ sec.

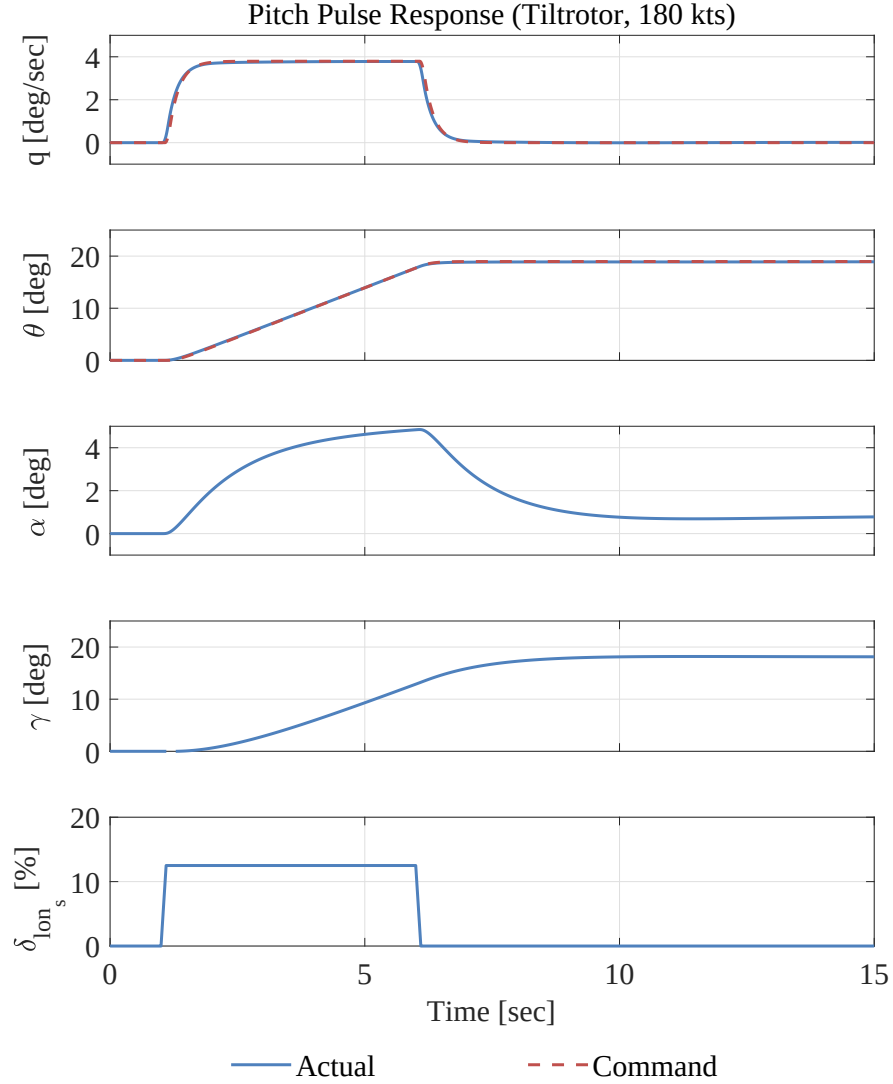


Figure 3.23. Longitudinal stick pulse response (tiltrotor, 180 kts).

3.4.2.2 Gain Schedule

Figure 3.24 shows the tiltrotor pitch rate (K_q) and angle of attack rate ($K_{\dot{\alpha}}$) feedback gains as a function of airspeed V and nacelle angle δ_{nac} (with $\delta_{\text{nac}} = 90$ deg corresponding to helicopter mode and $\delta_{\text{nac}} = 0$ deg corresponding to airplane mode). Figure 3.25 shows the tiltrotor pitch attitude (K_θ) and angle of attack (K_α) feedback gains as a function of airspeed V and nacelle angle δ_{nac} . Note that as described in Sec. 3.1.1,

the pseudo-inverse control allocation scheme normalizes the variations in control sensitivity of the bare-airframe with airspeed V and nacelle angle δ_{nac} . Therefore, those bare-airframe control sensitivity variations do not have to be accounted for by the variations in feedback gains shown in Figures 3.24 and 3.25.

Like for the coaxial-pusher, CONDUIT[®] optimization of the tiltrotor control laws resulted in smooth gain schedules. The remaining feedback gains show similar trends.

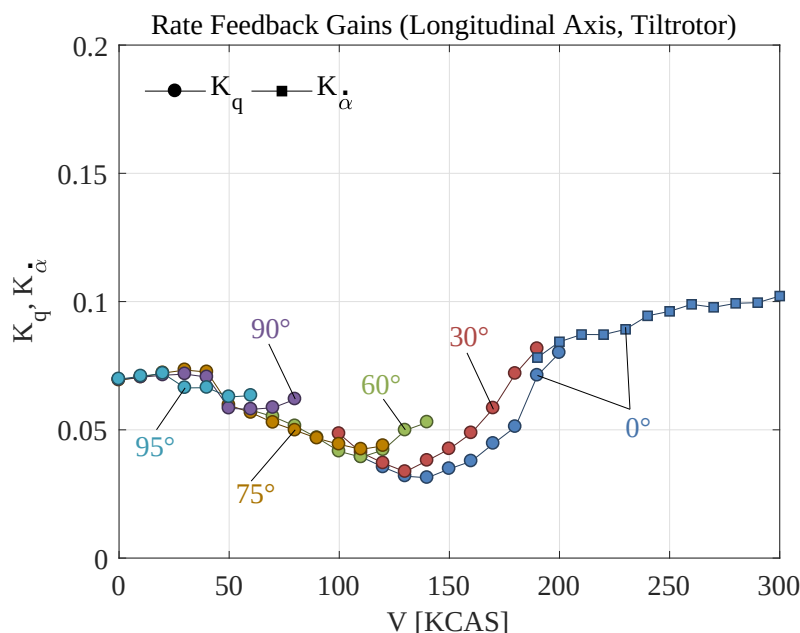


Figure 3.24. Pitch rate and angle of attack rate feedback gains (tiltrotor, optimized designs).

Figures 3.26 and 3.27 show several of the Tier 1 and Tier 2 specifications and their values for several airspeeds ranging from hover to $V_{\text{CAS}} = 240$ kts. At each airspeed, the results are shown for the nacelle range spanning the conversion corridor shown in Figure 3.3, with the results for the different nacelle angles at each airspeed connected with lines.

Figure 3.26 shows the stability margin, minimum crossover frequency, DRB/DRP, and OLOP specifications in all four axes. These are all Tier 1 specifications, and so are met for all designs through the optimization process. In the roll axis (Figure 3.26, first column), stability margin, minimum crossover frequency, and DRB/DRP are not strong functions of nacelle angle. However, The OLOP specification is a strong function of nacelle angle, with the values getting closer to the Level 1/Level 2 boundary for lower nacelle angles. This is because at lower nacelle angles, differential collective becomes less effective at generating roll moment, and differential longitudinal cyclic is phased in [74]. In contrast, the opposite is true for the yaw OLOP specification (Figure 3.26, last row, third subplot), where the values move away from the Level

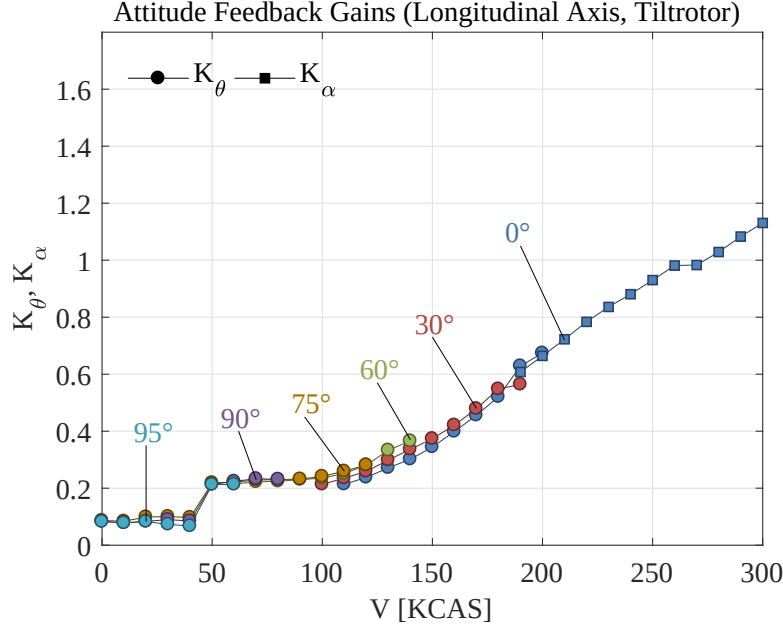


Figure 3.25. Pitch attitude and angle of attack feedback gains (tiltrotor, optimized designs).

1/Level 2 boundaries for lower nacelle angles as differential collective becomes more effective at generating yaw moment.

Figure 3.27 shows the piloted bandwidth/phase delay and maximum achievable rates for roll, pitch, and yaw (first two rows). These are Tier 1 requirements and are met for all designs through the optimization process and tuning of the command models.

The third row of specifications in Figure 3.27 shows the attitude quickness specification for roll, pitch, and yaw (Tier 2 requirements). The roll hover/low-speed and mid-/high-speed specifications (Figure 3.27, third row, first two subplots) display two sets of boundaries. The gray dashed boundaries represent the Target Acquisition & Tracking attitude quickness requirement (used as an objective requirement), while the solid black boundaries represent All Other MTEs requirement (used as a threshold requirement). At airspeed below $V_{CAS} = 200$ kts, the roll attitude quickness values meet the Target Acquisition & Tracking boundaries for larger nacelle angles, and meet the All Other MTEs boundaries for smaller nacelle angles. This is consistent with the decreased roll control power as the nacelles are lowered. In the high airspeed range (airplane mode), the ailerons are fully effective and roll control power is increased, and the $V_{CAS} = 240$ kts design meets the Target Acquisition & Tracking roll boundaries.

Pitch and yaw attitude quickness specifications (Figure 3.27, third row, third and fourth subplots) are available for hover/low-speed only. In the pitch axis, the designs meet the Target Acquisition & Tracking boundaries, while in the yaw axis the designs meet the All Other MTEs boundaries.

The last row of specifications shown in Figure 3.27 consists of MIL-STD-1797B requirements: (from left to right) pitch attitude bandwidth, flight path bandwidth, CAP, and pitch attitude dropback. All of the designs meet the Level 1 pitch attitude bandwidth requirement from MIL-STD-1797B (Figure 3.27, fourth row, first subplot). However, some of the mid-speed designs ($60 < V_{CAS} < 120$ kts) have flight path bandwidth values that are in the Level 2 region (Figure 3.27, fourth row, second subplot). These design points correspond to the designs that have an equivalent short period damping ζ_{sp} above the requirement in the CAP specification.

All designs meet the ADS-33E Level 1 flight path response to pitch attitude requirement (Tier 1 specification). Like the pitch attitude bandwidth requirement, the MIL-STD-1797B requirement on flight path bandwidth is more stringent than the ADS-33E requirement.

Finally, the dropback requirement is shown in the last subplot in Figure 3.27 and is met for all designs. As expected, there is near zero dropback for the pitch rate command/attitude hold (RCAH) designs ($V_{CAS} < 200$ kts). There is > 0 dropback for the normal acceleration designs ($V_{CAS} > 200$ kts), although it is well below the maximum allowable value.

3.5 Configuration Specific Considerations

3.5.1 Coaxial-Pusher

Several configuration-specific modifications were made to the coaxial-pusher control laws. The first modification was inclusion of feed-forward compensation for rotor tip clearance to ensure sufficient separation between the upper and lower rotors during maneuvering flight. As explained in Sec. 2.1.2.4, a rotor tip separation output was included in the coaxial-pusher model. This was done by using the flapping states of each rotor to determine the position of each rotor's tip path plane (TPP) and then determining the minimum separation of the two TPPs around the rotor azimuth.

It was noticed early in the coaxial-pusher control law design process, that without compensation for tip clearance, large maneuvers could cause the two rotors to hit (i.e., tip separation < 0) due to the gyroscopic forces on the two rotors. Figure 3.28 shows a time history of a roll piloted pulse response at an airspeed of $V_{CAS} = 240$ kts without compensation for rotor tip clearance (blue line). Note that for this roll maneuver, a maximum roll rate of $p \approx 50$ deg/sec is generated, which is about half of the required maximum roll rate specification in ADS-33E for Target Acquisition & Tracking ($p \geq 90$ deg/sec). Even for this reduced roll rate, without the tip clearance controller, the tip separation is reduced from its trim value of 4.1 ft to -1.1 ft (i.e., the tip path planes of the upper and lower rotors intersect).

To compensate for this, ideal crossfeeds between roll and pitch rate (p and q) and differential lateral and longitudinal cyclic commands ($\Delta\theta'_{1c}$ and $\Delta\theta'_{1s}$) were determined

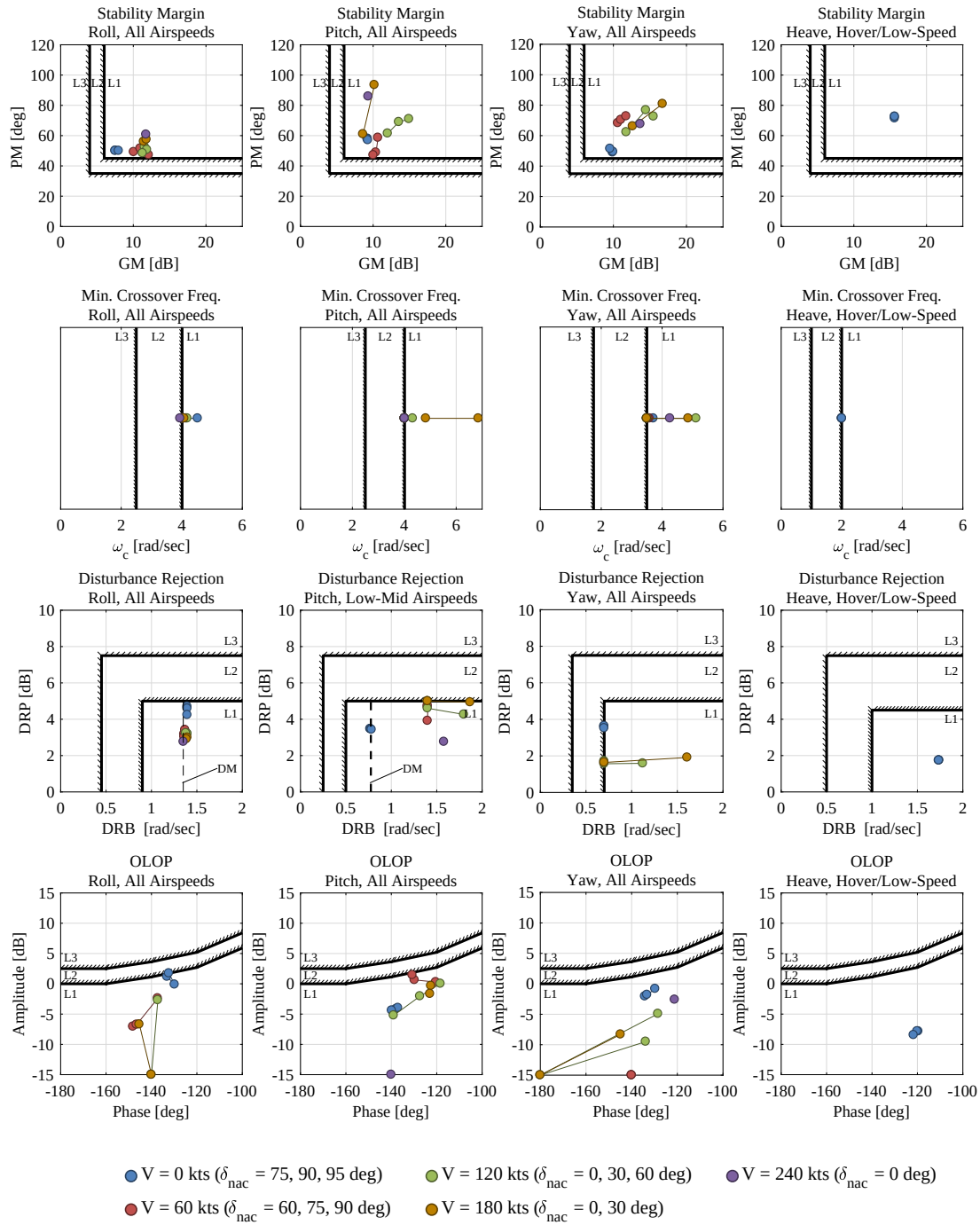


Figure 3.26. Example control system design specification and optimized design results (1/2, Tiltrotor, hover–240 kts).

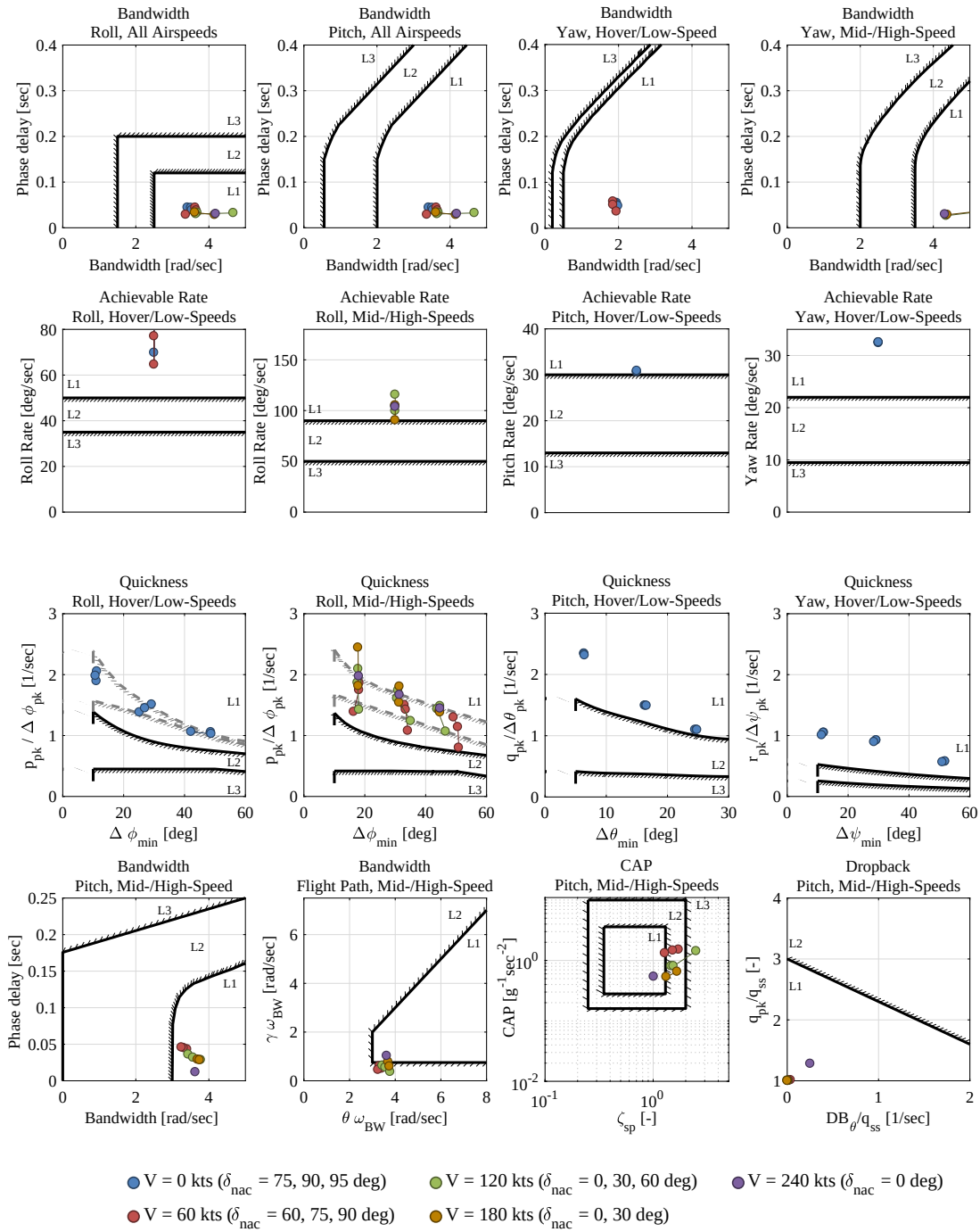


Figure 3.27. Example control system design specification and optimized design results (2/2, Tiltrotor, hover–240 kts).

from the differential flapping equations:

$$\begin{bmatrix} \Delta \ddot{\beta}_{1c} \\ \Delta \ddot{\beta}_{1s} \end{bmatrix} = \begin{bmatrix} Mf_p & Mf_q \\ Lf_p & Lf_q \end{bmatrix} \begin{bmatrix} p \\ q \end{bmatrix} + \dots + \begin{bmatrix} Mf_{\Delta\theta'_{1c}} & Mf_{\Delta\theta'_{1s}} \\ Lf_{\Delta\theta'_{1c}} & Lf_{\Delta\theta'_{1s}} \end{bmatrix} \begin{bmatrix} \Delta\theta'_{1c} \\ \Delta\theta'_{1s} \end{bmatrix} \quad (3.11)$$

To not impact the stability characteristics of the control system, the ideal crossfeeds to get $\Delta \ddot{\beta}_{1c} = \Delta \ddot{\beta}_{1s} = 0$ are determined using the *commanded* roll and pitch rates (p_{cm} , q_{cm}) as:

$$\begin{bmatrix} \Delta\theta'_{1c} \\ \Delta\theta'_{1s} \end{bmatrix} = - \begin{bmatrix} Mf_{\Delta\theta'_{1c}} & Mf_{\Delta\theta'_{1s}} \\ Lf_{\Delta\theta'_{1c}} & Lf_{\Delta\theta'_{1s}} \end{bmatrix}^{-1} \begin{bmatrix} Mf_p & Mf_q \\ Lf_p & Lf_q \end{bmatrix} \begin{bmatrix} p_{cm} \\ q_{cm} \end{bmatrix} \quad (3.12)$$

Thus, the tip clearance controller is implemented as a feed-forward crossfeed. The underlined terms in Eq. 3.12 are dominant, and so a simplified on-axis tip clearance feed-forward crossfeed is used:

$$\begin{aligned} \Delta\theta'_{1c} &= -\frac{Lf_q}{Lf_{\Delta\theta'_{1c}}} q_{cm} \\ \Delta\theta'_{1s} &= -\frac{Mf_p}{Mf_{\Delta\theta'_{1s}}} p_{cm} \end{aligned} \quad (3.13)$$

The differential cyclic commands are converted to upper and lower rotor actuator commands which are then summed with the individual actuator commands output by the control allocation section of the control system.

Figure 3.28 shows a time history of the same roll maneuver previously discussed with compensation for rotor tip clearance (dashed red line). With the tip clearance controller, the tip separation during the roll maneuver is only reduced from its trim value of 4.1 ft to 3.5 ft, a 90% improvement over the no tip clearance controller case (solid blue line) with negligible effects on the roll rate response. Figure 3.28 also shows the off-axis pitch rate q response during this maneuver which is small and not affected by the tip clearance controller. Similar results are seen in pitch, although since the maximum achievable rate requirements for roll are nearly twice as high as for pitch, the roll axis is more critical.

Note that since the tip clearance controller is implemented in the feed-forward based on commanded roll and pitch rates, instead of in the feedback based on actual roll and pitch rates, large uncommanded aircraft rates may still be an issue. An investigation into tip separation in gusts and turbulence is still needed.

The second configuration-specific modification made to the coaxial-pusher control laws was the inclusion of the a collective trim map on the collective/thrust control lever (TCL). This was done to reduce pilot workload during high-speed flight and acceleration/deceleration maneuvers. The collective trim map is only active above $V_{CAS} = 40$ kts where the vertical axis is open-loop (direct stick to head, Table 3.1). Figure 3.29 shows the collective trim map used in the coaxial-pusher control laws.

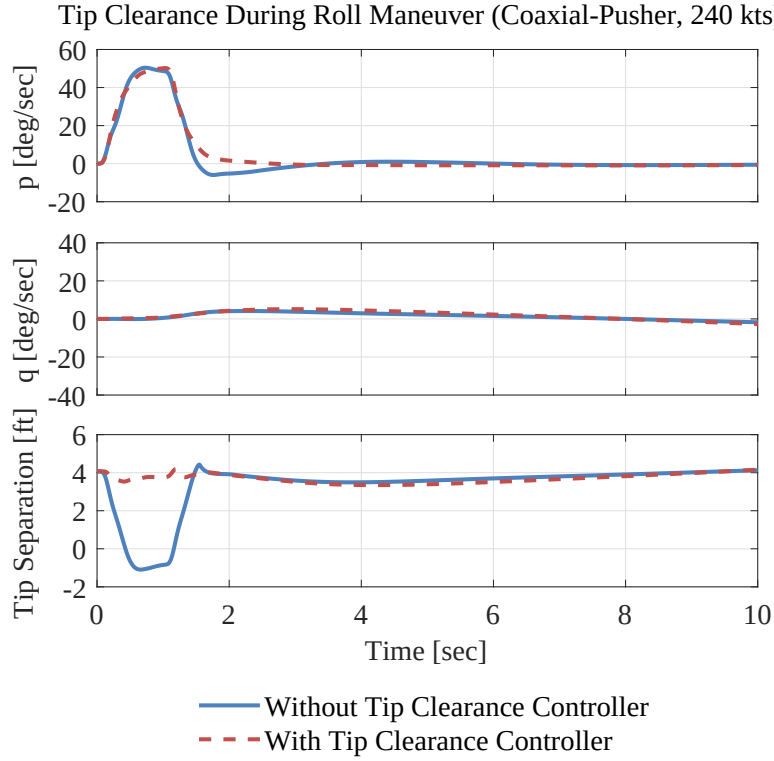


Figure 3.28. Tip clearance during roll maneuver (coaxial-pusher, 240 kts).

These values correspond to the collective θ_0 values required to trim the aircraft at each airspeed with ship level pitch attitude ($\theta = 0$ deg).

The third configuration-specific modification made to the coaxial-pusher control laws was a set of pilot controls for the pusher propeller. The pusher propeller is controlled via a thumb wheel on the TCL. The wheel commands a pusher propeller collective rate $\dot{\theta}_{0PP}$ proportional to its deflection, with maximum deflection commanding $\dot{\theta}_{0PP} = 7$ deg/sec and forward wheel deflection commanding increased θ_{0PP} (faster airspeed). Once released, θ_{0PP} remains fixed.

In addition to this thumb wheel, two push buttons can also be used to change pusher propeller collective. The first push button is the Zero Thrust switch and commands the pusher propeller to its zero thrust setting, thus providing the pilot a way to perform a quick deceleration. The second push button is the Thrust to Airspeed Couple button. This button sets the pusher propeller collective to roughly match the value required to hold the current airspeed with ship level pitch attitude. Note that this is not an airspeed hold mode as there is no airspeed feedback, and is simply a pusher propeller versus airspeed trim map. This button is useful when accelerating/decelerating like a conventional helicopter (with pitch attitude) to a given airspeed and then wanting to re-reference the pitch attitude to $\theta = 0$ deg.

Finally, to manage power, the control laws prioritize the main rotors over the

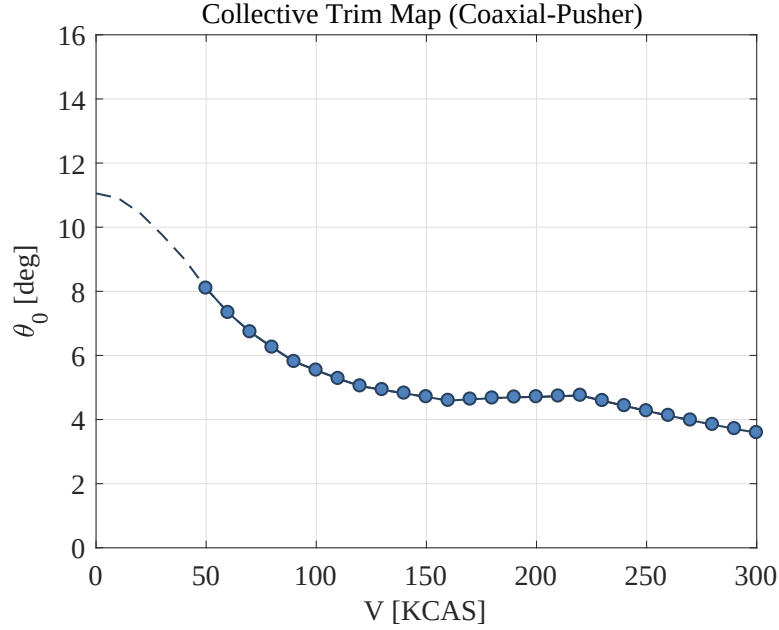


Figure 3.29. Collective trim map (coaxial-pusher).

pusher propeller. Therefore, if the pilot were to enter an aircraft state where power required P_{req} exceeded power available P_{avail} , the pusher propeller collective will be reduced until $P_{\text{req}} \leq P_{\text{avail}}$.

3.5.2 Tiltrotor

The tiltrotor-specific modifications include a collective bias map implemented as a function of nacelle angle. This is needed because the tiltrotor has a very large collective trim θ_0 range throughout its flight envelope [74], which is beyond the range of the rotor actuators (Table 3.4). Figure 3.30 shows the collective trim range for airspeeds spanning the conversion corridor at several nacelle angles, as well as the collective bias map implemented in the control laws which is used to drive the collective trim actuator on each rotor and offload the rotor actuators.

The second configuration-specific modification made to the tiltrotor control laws was a set of pilot controls to command nacelle angle. Nacelle angle δ_{nac} is controlled via a thumb wheel on the TCL. The wheel commands a nacelle angle rate $\dot{\delta}_{\text{nac}}$ proportional to its deflection, with forward deflection commanding forward nacelle (down towards airplane mode). The maximum rate is $\dot{\delta}_{\text{nac}} = 8$ deg/sec. In addition to the thumb wheel, two push buttons can also be used to cycle the nacelles between preset angles $\delta_{\text{nac}} = 0, 30, 60, 75, 90, 95$ deg.

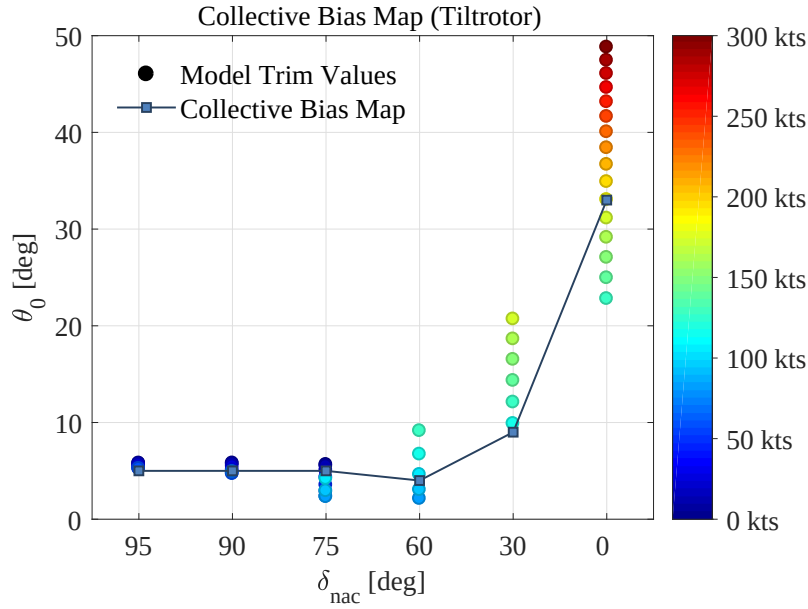


Figure 3.30. Collective bias map (tiltrotor).

3.6 Summary

This chapter discussed the development and optimization of full-flight envelope control systems for both coaxial-pusher and tiltrotor aircraft. A common explicit model following control system architecture was used for both aircraft, and was tuned to meet a common comprehensive set of stability, handling-qualities, and performance specifications. Optimization results were shown for both aircraft at a range of airspeeds spanning their respective flight envelopes and demonstrated the predicted Level 1 handling qualities of both configurations. Finally, several configuration specific aspects of the control laws were described, including a tip clearance feed-forward controller for the coaxial-pusher and a collective trim map for both aircraft.

Chapter 4 |

Inner-Loop Control Law Handling Qualities Simulation

Once the flight control systems described in Chapter 3 were developed, they were integrated with their respective bare-airframe models into full flight-envelope closed-loop models that are capable of real-time simulation. A piloted simulation experiment was then conducted in the NASA Ames Vertical Motion Simulator (VMS) facility using five Army experimental test pilots (XPs). Since the focus of the experiment was high-speed and transition handling qualities, seven newly developed National Rotorcraft Technology Center (NRTC) high-speed handling-qualities demonstration maneuvers were used: Pitch and Bank Attitude Capture and Hold [44], Pitch and Roll Sum-of-Sines Tracking [45], Break Turn [42], and High-Speed Acceleration and Deceleration [43].

This chapter provides a description of the simulation experimental setup, including information about the VMS and the handling qualities tasks used. Results are provided first for the coaxial-pusher and then for the tiltrotor. The chapter concluded with a discussion of the results.

4.1 Experimental Setup

4.1.1 Simulation Facilities

The handling qualities experiment was conducted in the Vertical Motion Simulator (VMS) at NASA Ames Research Center, shown in Figure 4.1(a). The VMS provides six degree-of-freedom motion with 60 feet of vertical and 40 feet of lateral travel.

Figure 4.1(b) shows the inside of the Transport Cab (T-Cab) used for this experiment. The cab provides a 180-degree field of view to the pilot (right seat), as well as a chin window. The inceptor configuration consisted of a side-stick attached to the right-hand side by the pilot seat, standard pedals, and a thrust controller lever (TCL) using pull-for-power logic. The TCL had a thumb wheel to control the coaxial-pusher propeller and the tiltrotor nacelle angle, in addition to two



(a) Cab external view



(b) Cab internal view

Figure 4.1. NASA Ames Vertical Motion Simulator.

special function push buttons for each platform (as described in Sec. 3.5). A heads-up display, depicted on the external world view, kept the pilot focused outside rather than on the instrumentation.

4.1.2 Nonlinear Simulation Model Validation

Before beginning handling qualities evaluations, implementation of the models in the simulator was validated. This was done by conducting both closed-loop and

broken-loop automated frequency sweeps of the nonlinear simulation models. The frequency sweep simulation data were analyzed using CIPHER[®] [83] to extract the appropriate frequency responses and compare to those of the linear point model used in the control law development. Three responses were analyzed in each axis to validate the implementation of the stitched model and control laws: the closed-loop piloted response, the closed-loop disturbance response, and broken-loop response.

Figures 4.2 and 4.3 show lateral axis closed-loop and broken-loop frequency responses for both the nonlinear simulation model and linear analysis models of the coaxial-pusher at $V_{CAS} = 180$ kts. The figures show an excellent agreement between the nonlinear simulation and linear models, validating the implementation of the stitched model, gain schedule, and control laws in the simulation model.

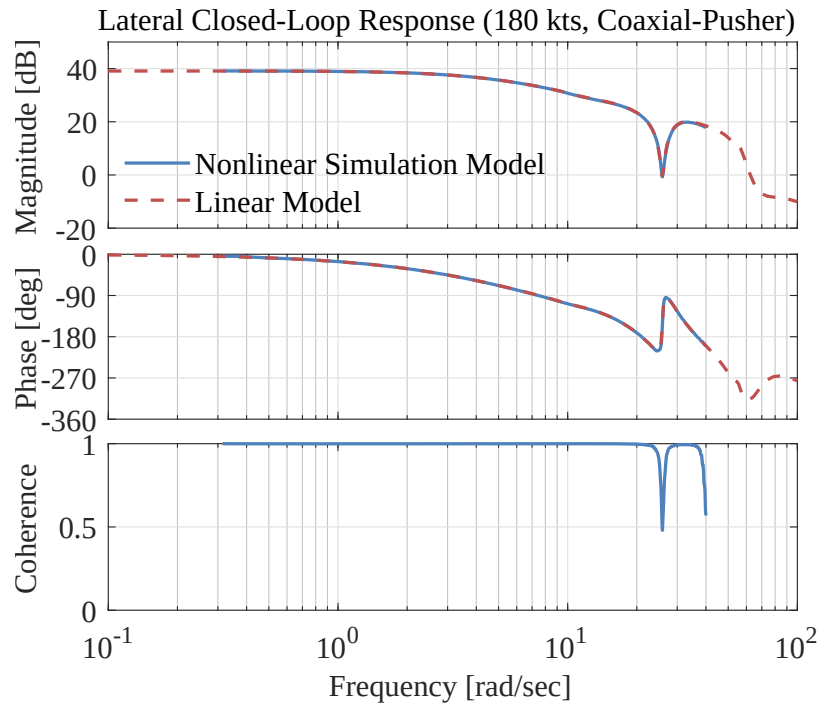


Figure 4.2. Lateral closed-loop frequency response comparison (coaxial-pusher, 180 kts).

Table 4.1 shows the pertinent control system metrics for the lateral axis of the coaxial-pusher at three different airspeeds spanning the flight envelope. Again, the excellent agreement between the simulation and analysis models indicates a proper implementation of the simulation model in the simulator.

Figure 4.4 shows a closed-loop lateral pulse response time history comparisons between the nonlinear simulation and linear analysis models of the coaxial-pusher at $V_{CAS} = 180$ kts. The time histories match very well, showing further validation of the nonlinear simulation model used for handling qualities assessment.

Similar comparisons were done for the other axes and at additional airspeeds for the coaxial-pusher and tiltrotor simulation models. All validations showed excellent

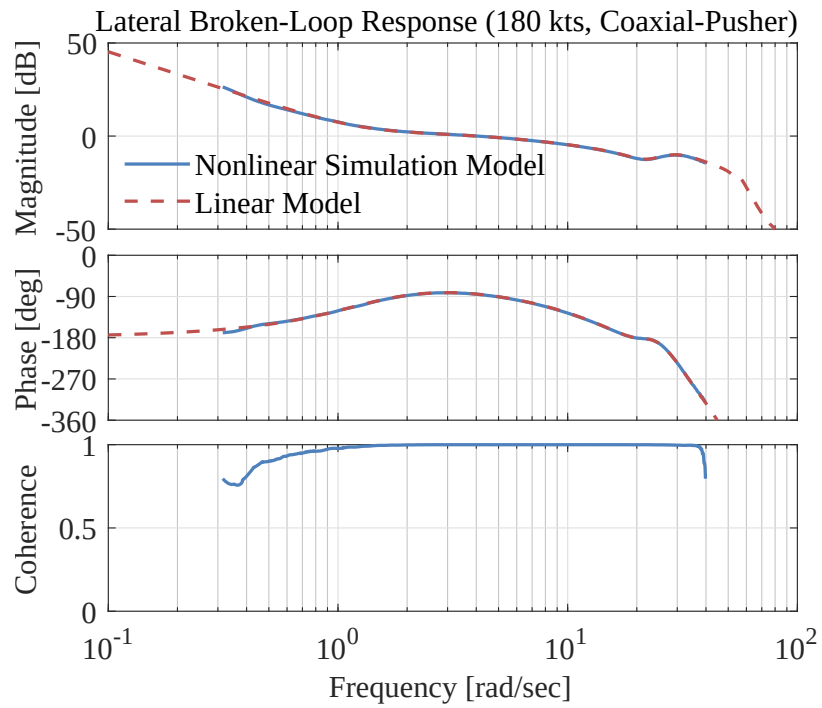


Figure 4.3. Lateral broken-loop frequency response comparison (coaxial-pusher, 180 kts).

agreement with the analysis models.

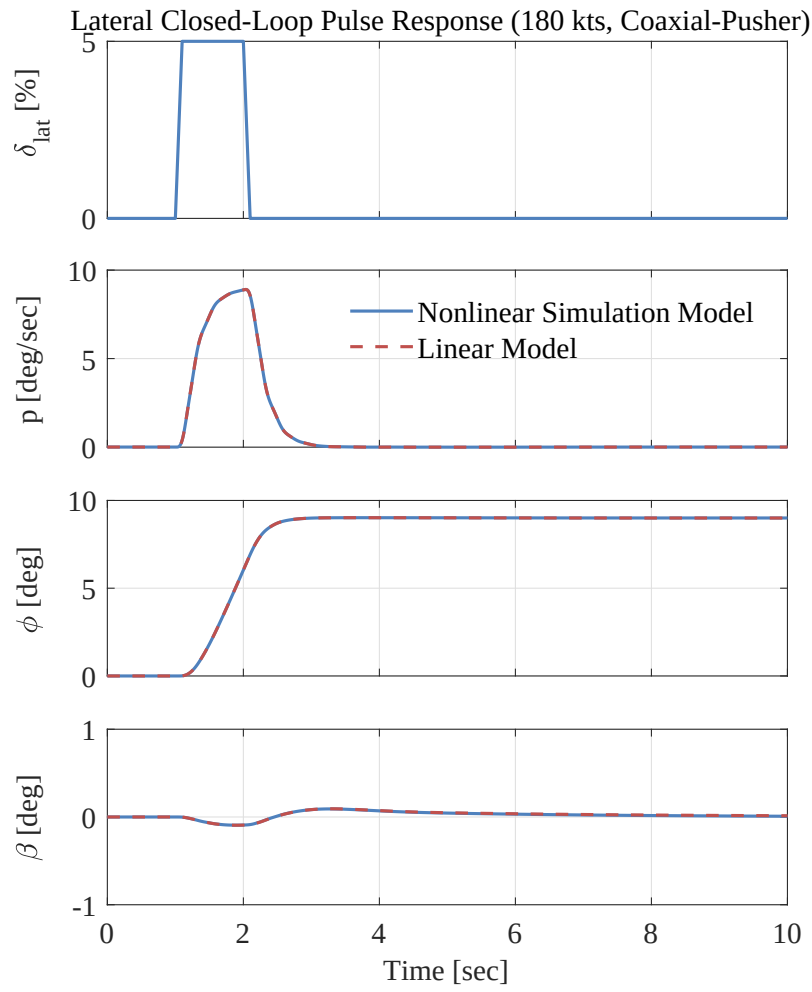


Figure 4.4. Lateral closed-loop pulse response comparison (coaxial-pusher, 180 kts).

Table 4.1. Simulation and Linear Point Model Comparison (Coaxial-Pusher)

	Hover				180 kts				240 kts			
	Sim Model	Linear Model	Sim Model	Linear Model	Sim Model	Linear Model	Sim Model	Linear Model	Sim Model	Linear Model	Sim Model	Linear Model
PM [deg] [*]	55.56	55.68	94.84	95.31	81.81	82.46	81.81	82.46	81.81	82.46	81.81	82.46
GM [dB] [*]	11.69	11.53	11.81	11.71	11.43	11.26	11.43	11.26	11.43	11.26	11.43	11.26
ω_c [rad/sec] [*]	4.00	4.01	4.00	4.01	4.05	4.01	4.05	4.01	4.05	4.01	4.05	4.01
ω_{BW_phase} [rad/sec] [†]	2.75	2.78	2.98	3.09	2.90	3.02	2.90	3.02	2.90	3.02	2.90	3.02
ω_{BW_gain} [rad/sec] [†]	4.29	4.30	4.96	5.06	4.90	5.03	4.90	5.03	4.90	5.03	4.90	5.03
τ_p [sec] [†]	0.05	0.05	0.05	0.05	0.04	0.05	0.04	0.05	0.04	0.05	0.04	0.05
DRB [rad/sec] [‡]	1.40	1.40	1.42	1.40	1.42	1.40	1.42	1.40	1.42	1.40	1.42	1.40
DRP [dB] [‡]	3.87	3.84	1.20	1.19	1.69	1.67	1.69	1.67	1.69	1.67	1.69	1.67

^{*} Loop broken at roll input to control allocation matrix

[†] Roll attitude bandwidth

[‡] Roll attitude disturbance response

4.1.3 Handling Qualities Task Definitions

Seven newly developed NRTC high-speed handling qualities demonstration maneuvers, or Mission Task Elements (MTEs), were used to assess the handling qualities of the coaxial-pusher and tiltrotor control laws in the transition and high-speed flight regime. The tasks used are Pitch and Bank Attitude Capture and Hold [44], Pitch and Roll Sum-of-Sines Tracking [45], Break Turn [42] and High-Speed Acceleration/Deceleration [43]. The tasks are briefly described here, but more details, including tables of desired and adequate performance bounds, are provided in Appendix A.

Pitch angle capture (PACH) and pitch tracking (Pitch SOS) tasks were flown at two airspeeds ($V_{CAS} = 180$ and 220 kts) to test both longitudinal responses types (RCAH and n_z command). Bank angle capture (BACH), roll tracking (Roll SOS), and the Break Turn maneuver were tested at $V_{CAS} = 180$ kts only. A speed range of $V_{CAS} = 50 - 220$ kts was used for the High-Speed Acceleration/Deceleration.

4.1.3.1 Pitch/Bank Angle Capture and Hold

The attitude capture and hold tasks [44] are precision, non-aggressive maneuvers. The tasks are flown using the display shown in Figure A.5, which is driven by a reference signal composed of a series of step changes in attitude (e.g., Figure A.6). From steady, wings level flight the aircraft is pitched or banked to capture and maintain the commanded angle within the specified tolerance for 5 sec. Pitch attitudes of ± 5 deg are used and roll attitudes of ± 30 deg are used.

The objective of the task is to evaluate the ability to pitch/bank the aircraft and capture a desired attitude. Additionally, the task identifies maneuverability limitations, inceptor characteristics, cross coupling, and any pilot induced oscillations (PIO) tendencies.

Table A.5 lists the desired and adequate performance criteria for the Bank/Pitch Angle Capture and Hold tasks.

4.1.3.2 Pitch/Roll Sum-of-Sines Tracking

The objectives of the Sum-of-Sines (SOS) Tracking MTE [45] are to evaluate the handling qualities in forward flight in a tight, closed-loop tracking task, evaluate the feel system, control sensitivity, and cross coupling, and identify any bobble or PIO tendencies.

The task is driven by an automated command signal based on a sum of sine waves of different frequencies and amplitudes (e.g., Figure A.4) and flown using the display shown in Figure A.5. A Fibonacci series-based SOS input is defined to emphasize a frequency range that encompasses key vehicle dynamics and typical closed-loop control [45]. In this case, the sine wave magnitudes were set via a second-order Butterworth filter with a bandwidth of $\omega_{BW} = 0.60$ rad/sec for the pitch axis and

$\omega_{BW} = 0.83$ rad/sec for the roll axis. The magnitudes of the sine waves were scaled to give a pitch target signal RMS of 2.5 deg and a roll target signal RMS of 10 deg.

Table A.4 lists the desired and adequate performance criteria for the Pitch/Roll Sum-of-Sines Tracking tasks.

4.1.3.3 Break Turn

The Break Turn MTE [42] is a non-precision, aggressive maneuver composed of a 90 deg heading change designed for evasive combat maneuvering. The pilot must perform an aggressive flight path change within the operation flight envelope (OFE) of the aircraft. The task is meant to investigate any potential handling qualities issues or cliffs or PIO tendencies resulting from aggressive roll-axis inputs. To increase pilot workload, the amount of altitude and airspeed loss allowed is specified.

The time requirement to complete the 90 deg heading change is based on an ideal time to complete the maneuver T_{ideal} and an allowable ΔT . Calculation of T_{ideal} is explained in Ref. 42 and in Appendix A, and is based on several parameters. For both aircraft used in this study, T_{ideal} was calculated using the MIL-STD-1797B Class I, Category A time to bank requirements of $\phi_{req} = 60$ deg in $t_{req} = 1.3$ sec, a task velocity of $V_{CAS} = 180$ kts, and an aircraft normal load factor limit $n_{zlim} = 2.5$ g. The resulting $T_{ideal} = 8.9$ sec, giving the following desired and adequate times:

$$T_{des} = T_{ideal} + 3.5 \text{ sec} = 12.4 \text{ sec} \quad (4.1)$$

$$T_{adq} = T_{ideal} + 7.0 \text{ sec} = 15.9 \text{ sec} \quad (4.2)$$

Table A.1 lists the desired and adequate performance criteria for the Break Turn task.

4.1.3.4 High-Speed Acceleration/Deceleration

The High-Speed Acceleration/Deceleration MTE [43] evaluates up-and-away handling qualities in transitional flight for aircraft that experience significant configuration changes with airspeed. The maneuver is composed of two phases: a maximum performance level-flight acceleration, and a maximum performance level-flight deceleration, over a speed range of $V_{CAS} = 50$ kts to $V_{CAS} = V_H - 10$ kts. In the case of the generic coaxial-pusher and tiltrotor aircraft tested here, the airspeed range used was $V_{CAS} = 50 - 220$ kts. Each of the phases has separate performance criteria and is rated separately.

The objectives of the MTE are to evaluate pitch and heave axis handling qualities for aggressive maneuvering throughout the speed range envelope; check for undesirable coupling between the longitudinal, lateral-directional, and unconventional axes; check for controller harmony between the heave axis, pitch axis, and unconventional/auxiliary control axes/inceptors; and check for handling qualities degradation during transitional flight regimes.

Like the Break Turn MTE described above, there is an ideal or minimum time to complete the maneuver (T_{minimum}) that must be determined prior to testing. In this case, T_{minimum} was determined during testing by having pilots fly the acceleration and deceleration portions of this task, using the tolerances given in Table A.3 as a guide only (but not adhering to them strictly). The following values of T_{minimum} were determined for the coaxial-pusher:

$$\text{Acceleration: } T_{\text{minimum}} = 36 \text{ sec} \quad (4.3)$$

$$\text{Deceleration: } T_{\text{minimum}} = 46 \text{ sec} \quad (4.4)$$

and for the tiltrotor:

$$\text{Acceleration: } T_{\text{minimum}} = 36 \text{ sec} \quad (4.5)$$

$$\text{Deceleration: } T_{\text{minimum}} = 34 \text{ sec} \quad (4.6)$$

Table A.3 lists the desired and adequate performance criteria for the High-Speed Acceleration/Deceleration task.

4.1.4 Pilot Questionnaire and Rating Scales

Appendix B shows the questionnaire and pilot rating scales that were provided to the pilots during their handling qualities evaluation. After several familiarization runs of each maneuver with each aircraft, pilots would conduct two to three runs for record. Subsequently, they would answer a questionnaire specifically tailored for this experiment (Figure B.1) and then use the Cooper-Harper Handling Qualities Rating Scale [30] (Figure B.2) to provide a Handling Qualities Rating (HQR). Pilots were also provided with a PIO rating scale [107] (Figure B.4) to use in case they encountered any PIO, however none were encountered during testing.

4.1.5 Pilot Demographics

Five U.S. Army experimental test pilots (XPs) participated in the handling qualities assessment. The ranks of the participants varied from Chief Warrant Officer 3 (CW3) to Lieutenant Colonel (O-5). All of the evaluation pilots had over 2,400 hours of total flight time with the maximum being 4,000 hours. Their median number of months deployed in combat was 21. Their flight experience was primarily in H-60, H-58, and H-47, but other aircraft have been evaluated. Their aircraft qualifications ranged mainly across 12 different aircraft types, although one had experience in 30 different aircraft.

4.2 Coaxial-Pusher Handling Qualities Results

Figure 4.5 shows the Handling Qualities Ratings (HQRs) collected for the coaxial-pusher aircraft for the high-speed MTEs. The error bars represent average, maximum, and minimum ratings collected. Average ratings for all MTEs except for the Break Turn are Level 1. The following sections will discuss the results for each MTE in more detail.

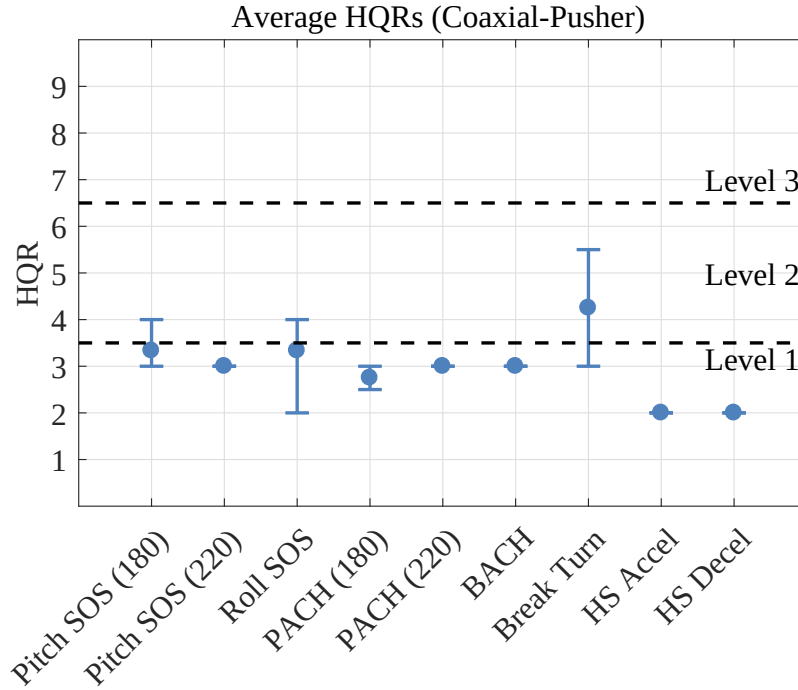


Figure 4.5. Handling qualities rating summary (coaxial-pusher).

4.2.1 Pitch Angle Capture and Hold

The Pitch Angle Capture and Hold (PACH) MTE was flown by two pilots (B and D), at two different airspeeds each ($V_{CAS} = 180$ kts and 220 kts). At $V_{CAS} = 180$ kts, the response type in the pitch axis is pitch rate q command, while at $V = 200$ kts, the response type is normal acceleration n_z command.

Pilots met desired performance for all runs. Figure 4.6 shows one example run for Pilot B at $V_{CAS} = 220$ kts. The figure shows the target pitch attitude with desired and adequate bounds, and actual aircraft pitch attitude, which remains well within the desired bounds for the entire run.

At $V_{CAS} = 180$ kts, pilots gave the task an average HQR 2.8 and noted that they were “able to meet desired [performance] easily” and be “quite precise,” and that the ride quality was “smooth.”

At $V_{CAS} = 220$ kts, pilots gave an average HQR 3 and had similar comments. One of the pilots noted that the normal acceleration command control laws were “not as tight” as the rate command control laws but that he was not working hard. The pilot also noted a slight tendency to overshoot, but that he was consistently meeting desired and that ride quality was smooth.

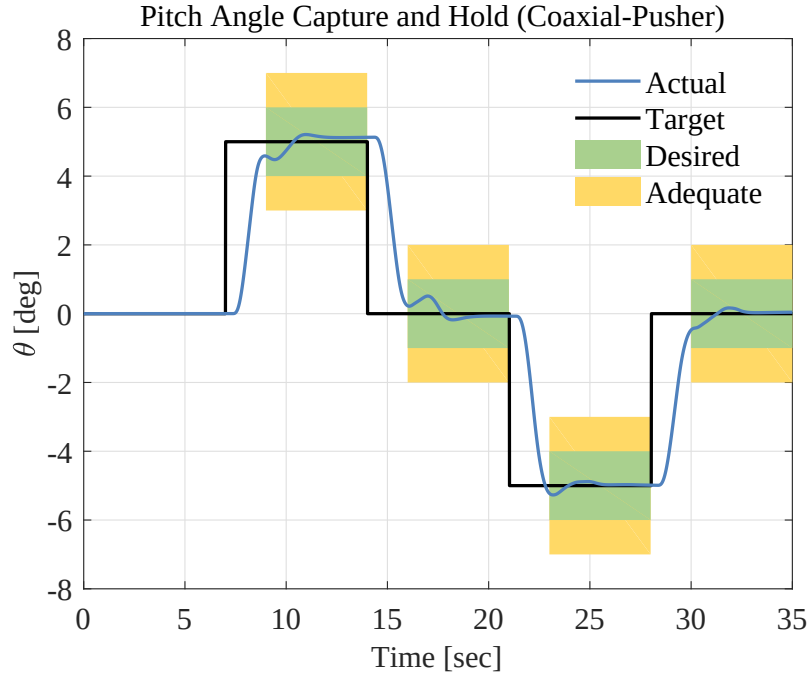


Figure 4.6. Pitch Angle Capture and Hold example time history (coaxial-pusher, 220 kts).

4.2.2 Bank Angle Capture and Hold

The Bank Angle Capture and Hold (BACH) MTE was flown by two pilots (B and D) at one airspeed ($V_{CAS} = 180$ kts), since there is no switch in roll response type with airspeed. In all cases, pilots were able to meet desired performance. Figure 4.7 shows an example run, with the bank angle within desired performance throughout the task.

The two pilots rated this MTE and average HQR 3. Pilot B commented that he was able to meet desired performance and that the “hold part was easy.” Pilot D commented that he was “consistently able to meet desired [performance]” and that the response was “tight.” Pilot D did note that he was inadvertently putting in off-axis inputs when applying large roll inputs, although this did not degrade the performance.

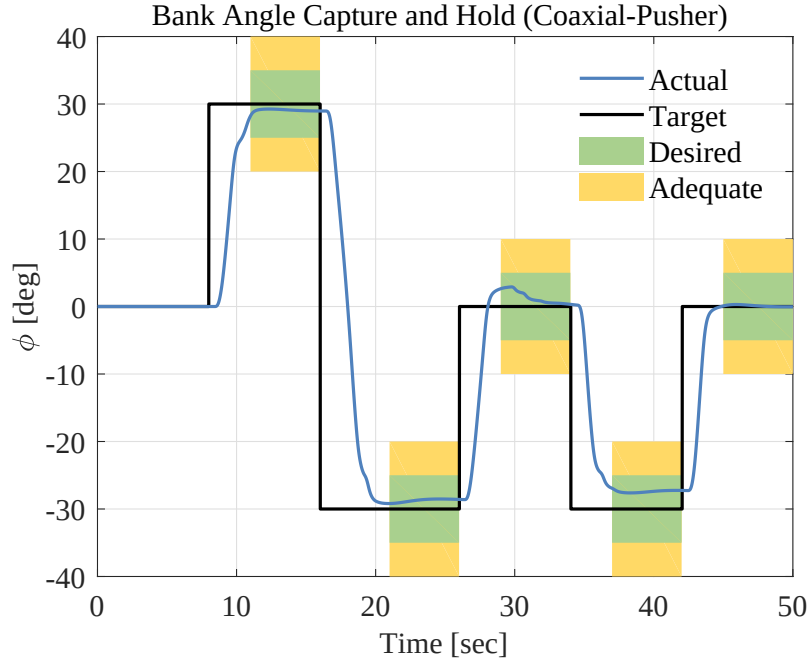


Figure 4.7. Bank Angle Capture and Hold example time history (coaxial-pusher, 180 kts).

4.2.3 Pitch Sum-of-Sines Tracking

The Pitch Sum-of-Sines (SOS) Tracking MTE was flown by four pilots. Two of the pilot flew the MTE at $V_{CAS} = 180$ kts (pitch rate q command), one pilot at $V_{CAS} = 220$ kts (normal acceleration n_z command), and one pilot at both airspeeds. Figure 4.8 shows the tracking performance for the task. All pilots were able to meet desired performance (i.e., percent within desired $\geq 50\%$) by a large margin.

Flying the task at $V_{CAS} = 180$ kts, pilots rated the task an average HQR 3.3 and commented that meeting desired was easy and that the attitude hold made maintenance easy. However, pilots also commented that the response was “heavily/too damped” which required “large inputs to overcome.” The comments about damping are consistent with the values of equivalent short period damping $\zeta_{sp} > 1$ for the $V_{CAS} = 180$ kts design in the Control Anticipation Parameter (CAP) specification in Figure 3.15 (last row, third subplot).

Flying the task at $V_{CAS} = 220$ kts, pilots rated the task an average HQR 3 and commented that they could consistently meet desired performance. Pilots did note that the response at $V_{CAS} = 220$ kts was more “jerky than RCAH” and that the biggest difference between the two airspeeds and response types was in ride quality, which was better at the lower airspeed. However, pilots still noted that there was minimal compensation.

The difference in ride quality between performing the Pitch Sum-of-Sines Tracking task at $V_{CAS} = 180$ kts versus $V_{CAS} = 220$ kts is expected, since the target signal RMS

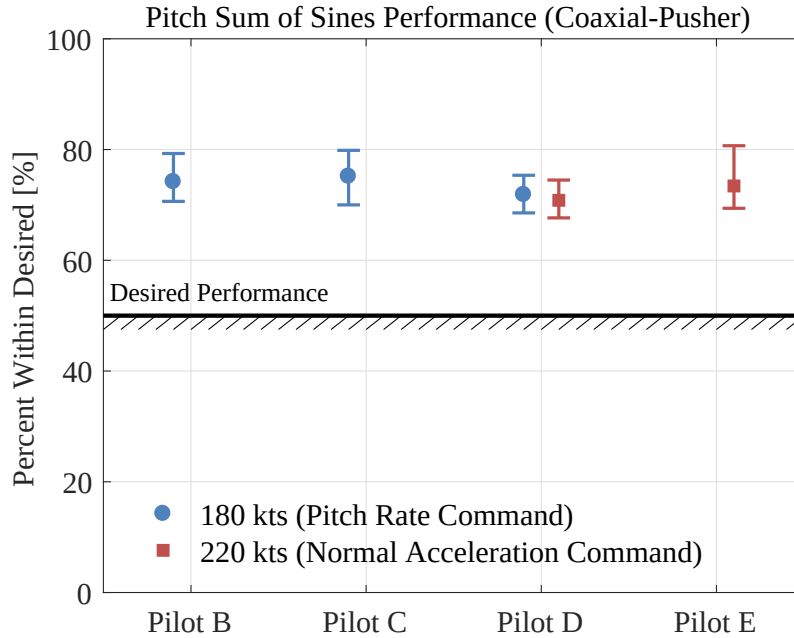


Figure 4.8. Pitch Sum-of-Sines Tracking performance (coaxial-pusher, 180 kts and 220 kts).

value is the same, requiring the pilot to generate the same pitch rates to meet desired performance at either airspeed. However, the resulting normal acceleration for the same pitch rate is 20% higher at $V_{CAS} = 220$ kts than at $V_{CAS} = 180$ kts. This 20% increase is consistent with the difference between peak-to-peak normal acceleration values seen in the data for this task at the two airspeeds.

4.2.4 Roll Sum-of-Sines Tracking

The Roll SOS Tracking MTE was flown by three pilots (B, C, and D) at $V_{CAS} = 180$ kts. Figure 4.9 shows the tracking performance for the task. All pilots were able to meet desired performance (i.e., percent within desired $\geq 50\%$), although Pilot B did have one run that was just outside of desired performance, but did meet adequate performance. Overall, pilots rated the task an average HQR 3.3 and commented that they could meet desired performance, but with large control inputs to get the necessary angles.

Pilot B did comment that he was “dealing with lots of pitch during roll,” however this was not noted or commented on by the other two pilots. Looking at the data from all the pilots’ runs, it was noted that the aircraft pitch rate q tracked the commanded pitch rate q_{cm} very closely. The mean RMS error between commanded and actual pitch rate during all runs was $RMS_{q_{err}} = 0.24$ deg/sec. However, it was noted that Pilot B’s longitudinal stick RMS was twice as large as the other two pilots’. This

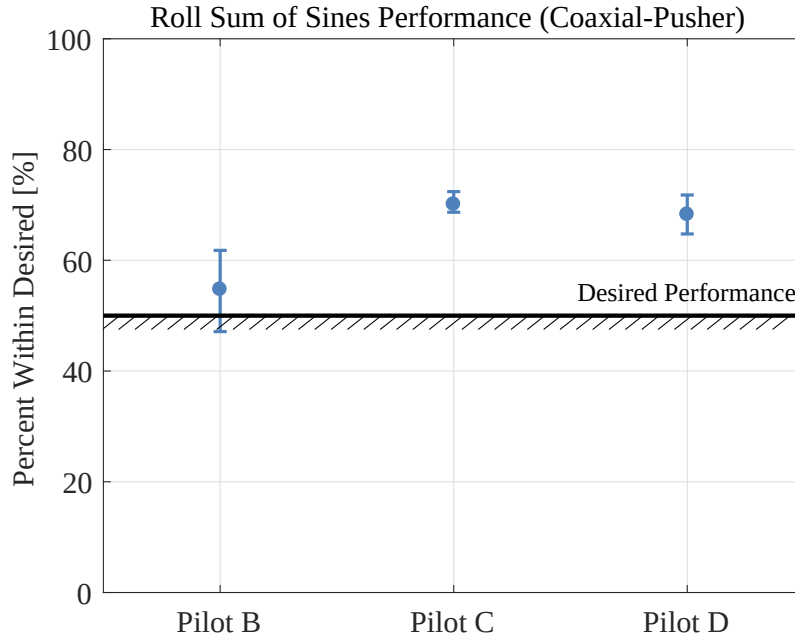


Figure 4.9. Roll Sum-of-Sines Tracking performance (coaxial-pusher, 180 kts).

suggest that perhaps the pitch-roll cross-coupling issue that Pilot B noted was actually due to cross contamination of his stick motion, similarly to what Pilot D noted for the BACH task.

4.2.5 Break Turn

The Break Turn MTE was flown by two pilots (A and E). Figure 4.10 shows the performance for all data runs for both pilots, plotted against the desired and adequate bounds listed in Table A.1. Figure 4.11 shows the pilot stick activity (cutoff frequency ω_{co} and RMS) for all axes during the Break Turn maneuver. The error bars represent the average, maximum, and minimum values for all of a given pilot's runs. For axes with RMS = 0, ω_{co} was not calculated. The cutoff frequency ω_{co} is defined as the half-power frequency of the pilot stick signal determined via spectral analysis, and is a measure of the pilot's operating frequency [83].

Pilot A was able to meet desired performance on all parameters except time to complete, which was over the desired time T_{des} by an average of 1.0 sec (Figure 4.10). Pilot A rated the maneuver an HQR 5.5 and commented that “targeting roll attitude [was] fairly easy” but that “pitch maintenance [was] difficult.” In fact, Pilot A noted that “pitch management during the turn and roll-out” was the critical subphase of this task. These comments correspond to Pilot A's higher ω_{co} in the longitudinal axis than lateral axis. However, Pilot A was able to maintain both altitude and airspeed within the desired bounds. Pilot A also commented that turn coordination worked

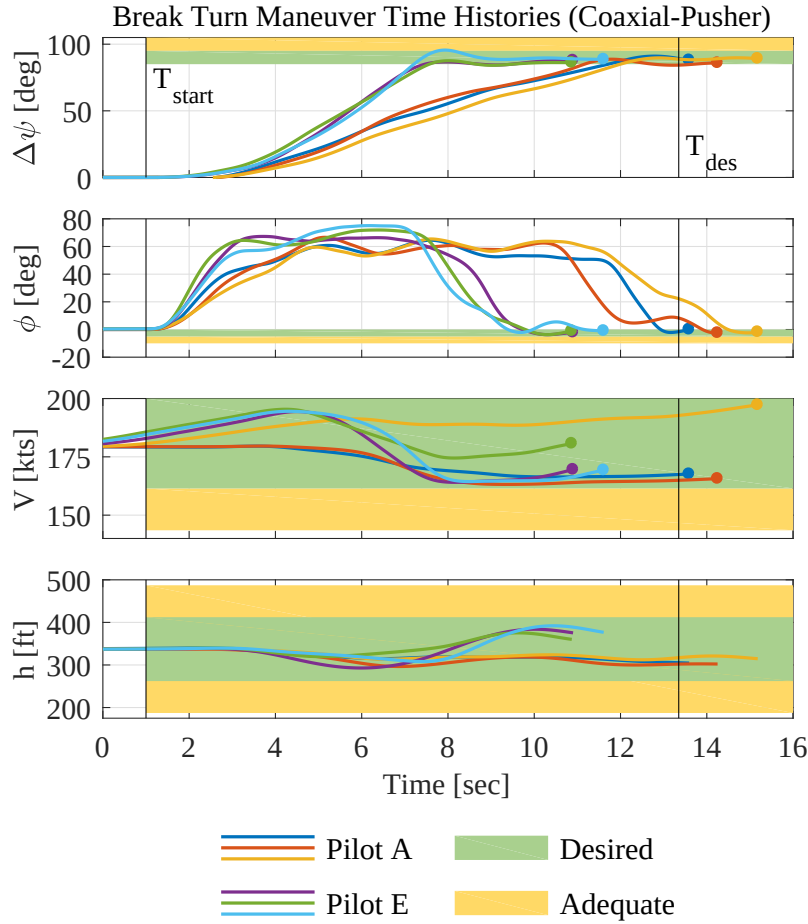


Figure 4.10. Break Turn task performance (coaxial-pusher, 180 kts).

well and that he was able to complete the maneuver with his feet on the floor (i.e., no pedal input), which can be seen as the $RMS = 0$ for pedals in Figure 4.11.

Pilot E rated this MTE an HQR 3 and commented that “aggressiveness was required” to complete the MTE in the desired time, and “precision was there” to stay within the desired bounds. Pilot E also noted that his feet were on the floor throughout the maneuver (pedal $RMS = 0$, Figure 4.11), and he relied on the control system to provide turn coordination. Pilot E did increase speed using the pusher propeller before entering the turn, as can be seen by the increasing airspeed in the third subplot in Figure 4.10.

Neither pilot used the TCL during this maneuver (TCL $RMS = 0$, Figure 4.11), preferring to adjust the airspeed by using the pusher propeller thrust control.

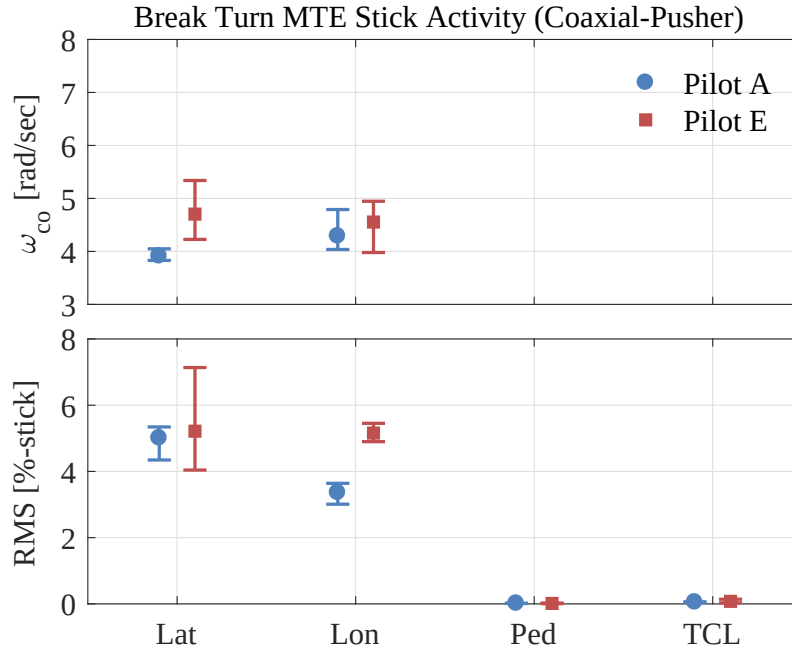


Figure 4.11. Break Turn stick activity (coaxial-pusher, 180 kts).

4.2.6 High-Speed Acceleration/Deceleration

Two pilots (A and E) flew the High-Speed Acceleration and Deceleration maneuvers with the coaxial-pusher. Figure 4.12 shows time histories for all record runs for Pilots A and E for the acceleration portion of the maneuver. Both pilots were able to meet desired performance on all parameters.

Pilot A rated the maneuver an HQR 2 and commented that the ability to meet desired was “all but effortless.” His technique was to accelerate by setting the pusher propeller commanded thrust to 100% and then use the Thrust to Airspeed Couple button at $V_{CAS} = 217$ kts to capture $V_{CAS} = 220$ kts. Pilot A also noted that the maneuver required no inputs on the lateral stick or pedals.

Pilot E rated the maneuver an HQR 2 and commented that he used the Thrust to Airspeed Couple button to manage speed, which was “very precise.” Pilot E noted some slight climbs, but that they were easily adjusted, and noted that the maneuver required very little compensation.

Figure 4.12 also shows the pitch attitude θ during the acceleration maneuver. Neither pilot had to adjust pitch attitude throughout the maneuver, since the coaxial-pusher can trim at $\theta = 0$ deg at all airspeeds. The collective trim map for the coaxial-pusher enabled the pilots to maintain altitude well within desired performance while performing a high-rate acceleration with inputs on the longitudinal stick and TCL of RMS < 0.5%, as shown in Figure 4.13. Note that pilot cutoff frequency ω_{co} was not calculated for cases of RMS = 0.

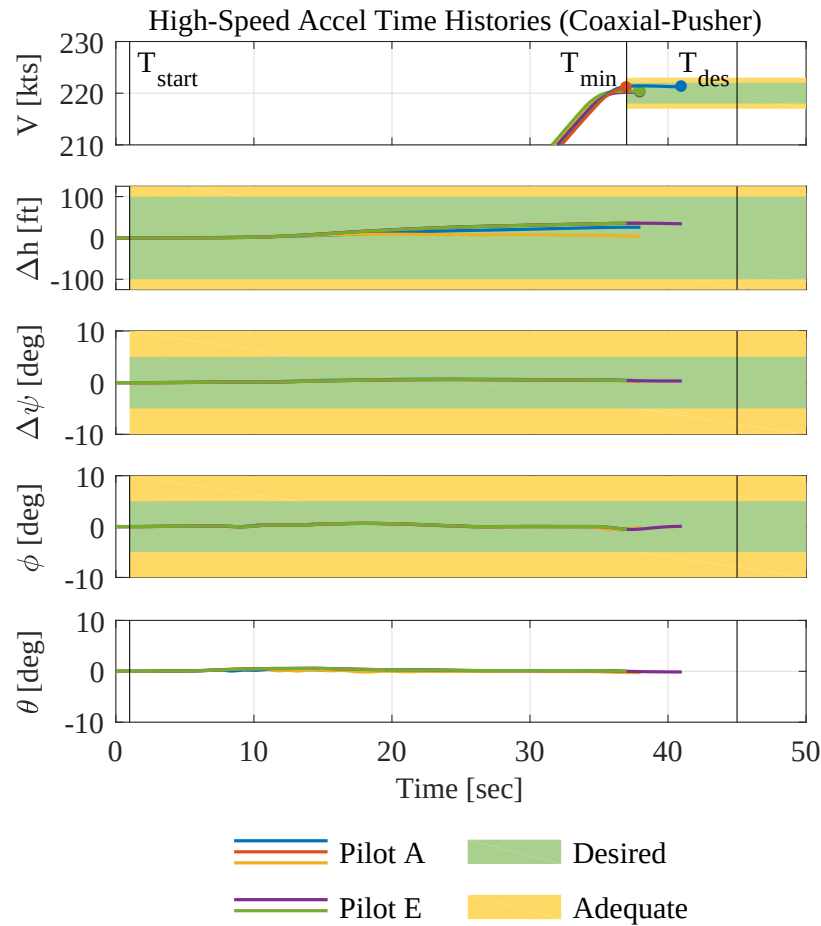


Figure 4.12. High-Speed Acceleration task performance (coaxial-pusher, 50–220 kts).

Figure 4.14 shows time histories for all record runs for Pilots A and E for the deceleration portion of the maneuver. Both pilots were able to meet desired performance on all parameters and rated the deceleration portion of the maneuver an HQR 2. Both pilots noted that they were able to meet desired performance using the pusher propeller thumbwheel and button controls only, with hands off all the other controls, which corresponds to the stick activity data for deceleration shown in Figure 4.13. In addition, as with the acceleration portion, pilots did not need to adjust the aircraft pitch attitude to control airspeed or altitude while transitioning through a wide range of airspeeds.

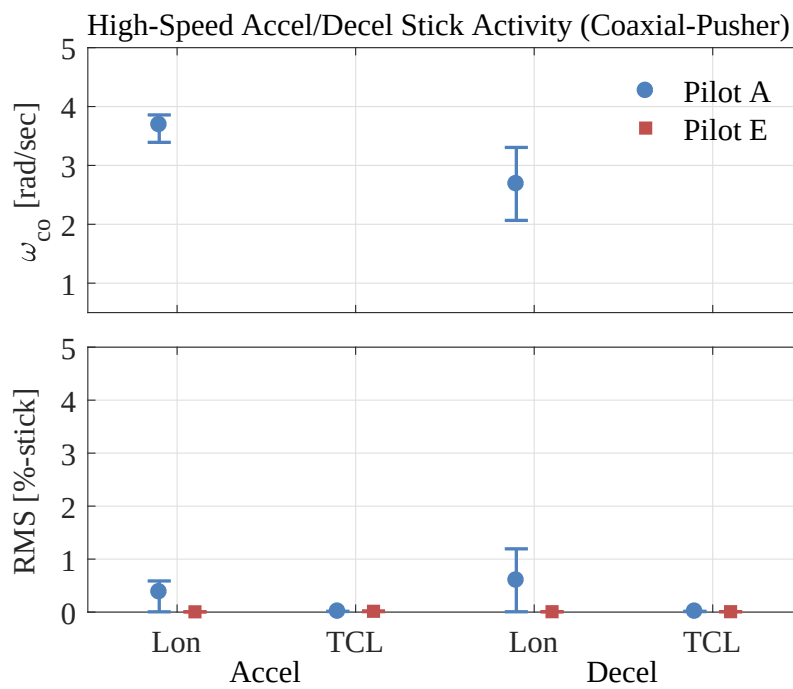


Figure 4.13. High-Speed Acceleration/Deceleration stick activity (coaxial-pusher).

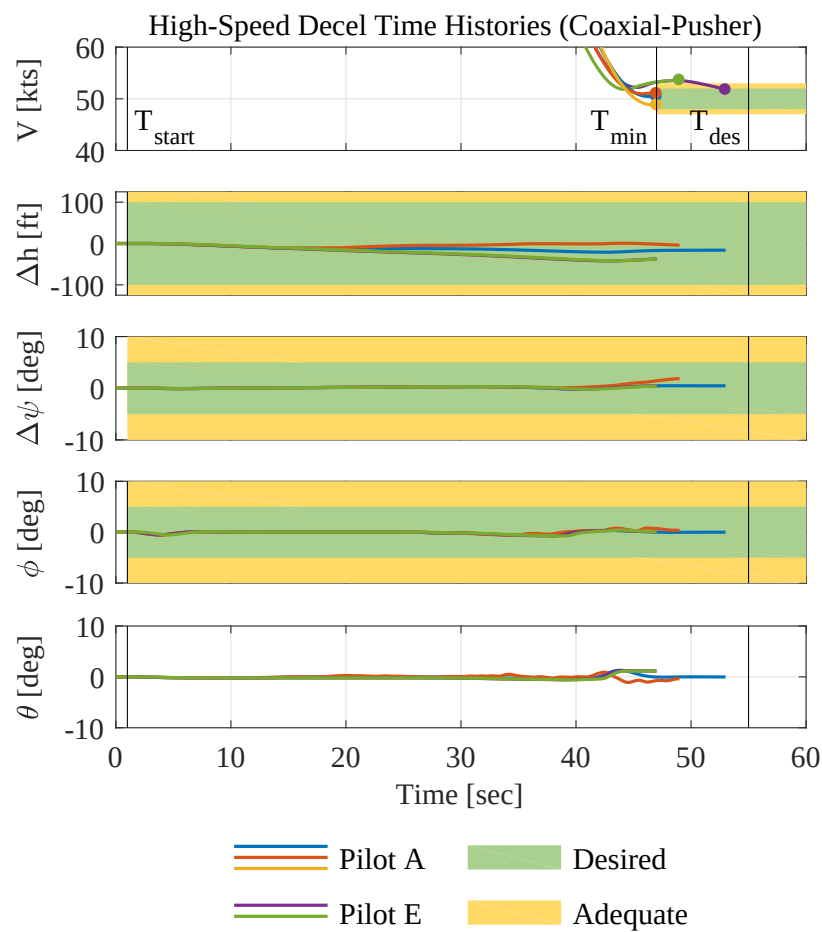


Figure 4.14. High-Speed Deceleration task performance (coaxial-pusher, 220–50 kts).

4.3 Tiltrotor Handling Qualities Results

Figure 4.15 shows the Handling Qualities Ratings (HQRs) collected for the tiltrotor aircraft for the high-speed MTEs. The error bars represent average, maximum, and minimum ratings collected. Average ratings for all MTEs except for the High-Speed Acceleration/Deceleration are Level 1. The following sections will discuss the results for each MTE in more detail.

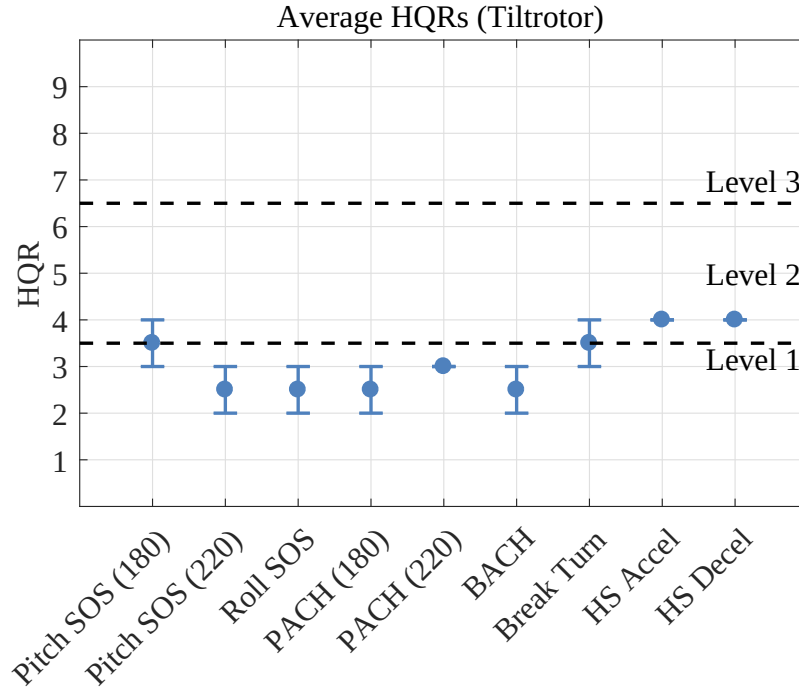


Figure 4.15. Handling qualities rating summary (tiltrotor).

4.3.1 Pitch Angle Capture and Hold

The Pitch Angle Capture and Hold MTE was flown by two pilots, at two different airspeeds each ($V_{CAS} = 180$ kts and 220 kts). At $V_{CAS} = 180$ kts, the response type in the pitch axis is pitch rate q command, while at $V_{CAS} = 220$ kts, the response type is normal acceleration n_z command.

Pilots met desired performance in all runs. Figure 4.16 shows one example run for Pilot B at $V_{CAS} = 180$ kts. The figure shows the target pitch attitude with desired and adequate bounds, and actual aircraft pitch attitude, which remains inside the desired bounds for the entire run.

At $V_{CAS} = 180$ kts, pilots gave the task an average HQR 2.5 and commented that it was “easy to meet desired [performance]” and “very easy to point the nose.” They

also noted that the response was “very precise” and they were able to use one single input.

At $V_{CAS} = 220$ kts, pilots gave the task an average HQR 3. Pilot B commented that he was able to meet desired performance, but that he was getting “slightly larger errors compared to 180 kts,” with some overshoots and “pitch bobbles.” Pilot A commented that he could also meet desired performance, but precision was reduced compared to at $V_{CAS} = 180$ kts, it was “harder to target a precise attitude,” and that he was getting “some overshoot in pitch.” These comments are all consistent with a normal acceleration command response type, which has a higher value of pitch attitude dropback as seen in Figure 3.27 (bottom row, last subplot). The value of pitch attitude dropback for this design is still within the desired region, which correlated to the Level 1 HQRs assigned by the pilots for this task.

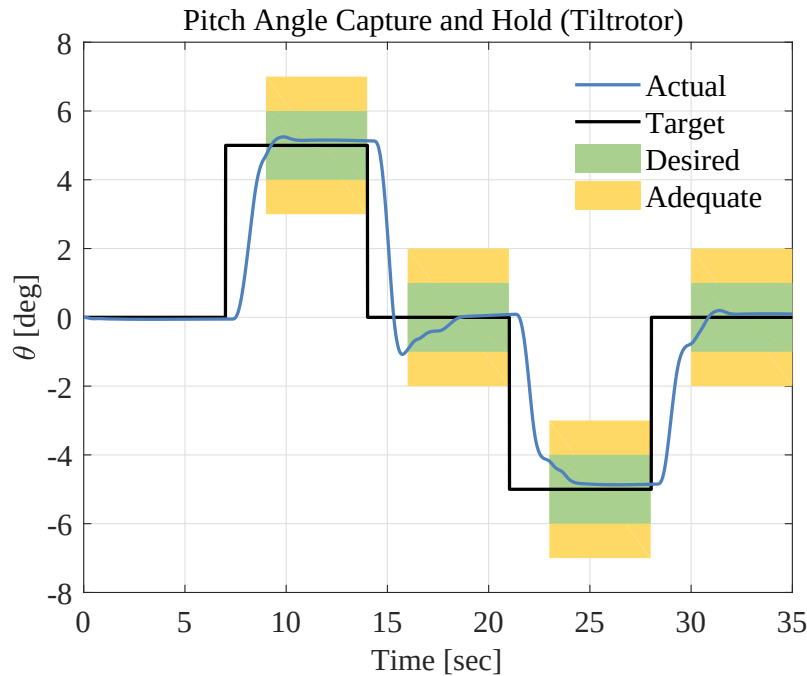


Figure 4.16. Pitch Angle Capture and Hold example time history (tiltrotor, 180 kts).

4.3.2 Bank Angle Capture and Hold

The Bank Angle Capture and Hold MTE was flown by two pilots (A and B), at $V_{CAS} = 180$ kts. Both pilots were able to meet desired performance for all of their runs. Figure 4.17 shows an example time history of the task flown by Pilot B.

Pilots rated this task an average HQR 2.5 and commented that it was “easy to meet desired.” Pilot A commented that he was using only lateral stick inputs without requiring to use the pedals or collective. He also noted that pitch never left the desired

circle on the bowtie display (Figure A.5 in Appendix A), which corresponds to pitch attitude θ staying within ± 1 deg of trim.

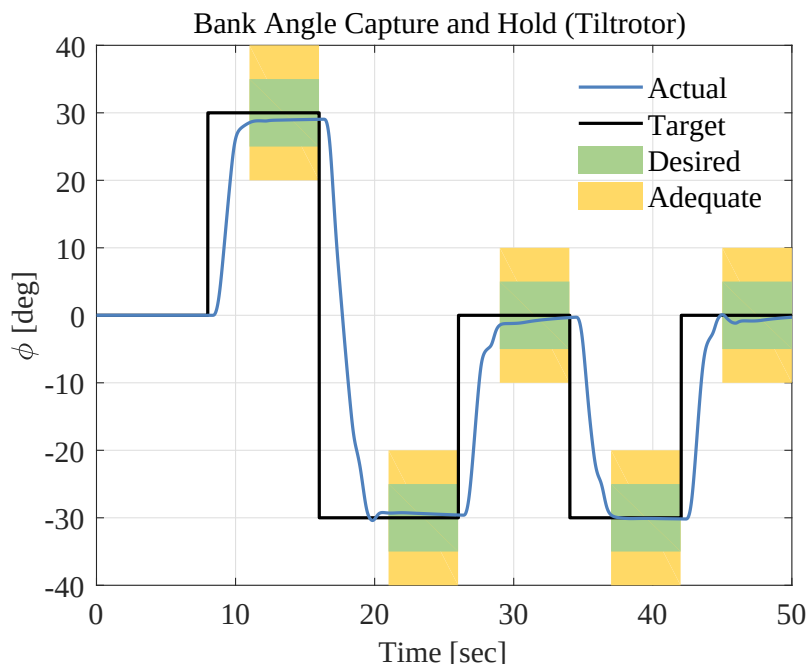


Figure 4.17. Bank Angle Capture and Hold example time history (tiltrotor, 180 kts).

4.3.3 Pitch Sum-of-Sines Tracking

The Pitch Sum-of-Sines Tracking MTE was flown by four pilots—two pilots at $V_{CAS} = 180$ kts (pitch rate command) and two pilots at $V_{CAS} = 220$ kts (n_z -command). Figure 4.18 shows the task performance for each pilot's runs. All pilots were able to meet desired performance by a wide margin.

Pilots B and C, who flew the task at $V_{CAS} = 180$ kts, rated the task an average HQR 3.5. They commented that they could “routinely meet desired” and “be very precise.” Pilot C did note that the response was damped and “not moving at the rated [he] would like.” This is consistent with the equivalent short period damping $\zeta_{sp} > 1.0$ seen in the CAP specification in Figure 3.27 (bottom row, third subplot).

Pilots A and E, who flew the task at $V_{CAS} = 220$ kts, rated the task an average HQR 2.5. They commented that they were “able to meet desired” and “be precise and aggressive,” and that this was a “single axis task” that required no “lateral, collective, or pedal” inputs. For the n_z -command response type, pilots noted that the response was “a little sluggish” but that they could “get the rates needed.”

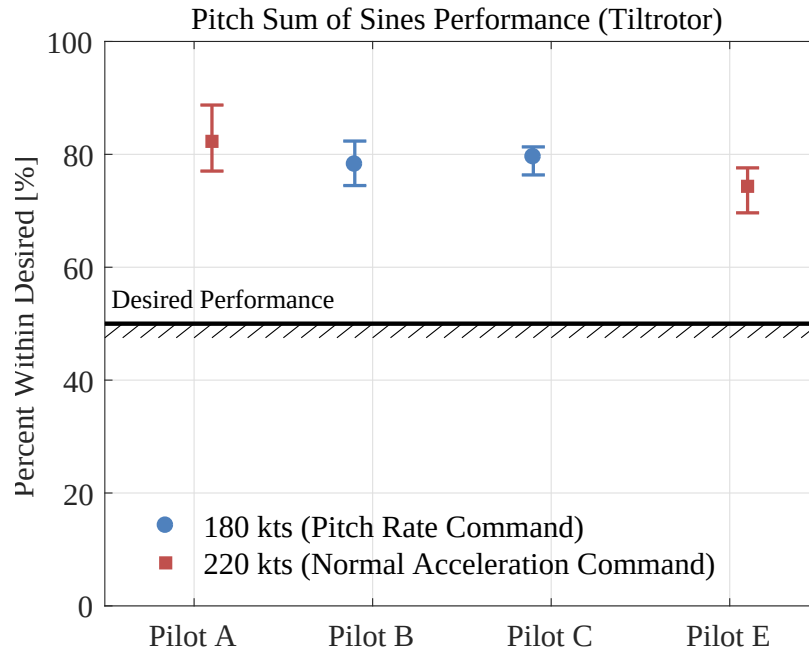


Figure 4.18. Pitch Sum-of-Sines Tracking performance (tiltrotor, 180 kts and 220 kts).

4.3.4 Roll Sum-of-Sines Tracking

The Roll Sum-of-Sines Tracking MTE was flown by two pilots (B and C) at $V_{CAS} = 180$ kts. Figure 4.19 shows the task performance for each pilot's runs. Both pilots were able to meet desired performance for all runs and rated this task an average HQR 2.5.

Both pilots noted that the pitch axis was a factor during this roll tracking task. However, inspection of the pitch rate data showed that it matched the commanded pitch rate with average RMS error less than 0.1 deg/sec. In addition, all designs meet the Level 1 ADS-33E pitch-due-to-roll coupling requirement for Target Acquisition & Tracking. This suggests that either the pilots were inadvertently commanding pitch during the maneuver or that pitch turn compensation was a factor for this task. It is notable that for the more discrete roll inputs used in the bank angle capture and hold tasks, pilots commented that pitch never left the desired circle.

4.3.5 Break Turn

The Break Turn MTE was flown by two pilots (A and E). Figure 4.20 shows the performance for all data runs for both pilots, plotted against the desired and adequate bounds listed in Table A.1. Both pilots were able to meet desired performance on all parameters.

Pilot A rated the maneuver and HQR 3 and commented that it was “easy to target bank angle” and that no pedal inputs were required to coordinate the turn. Pilot A

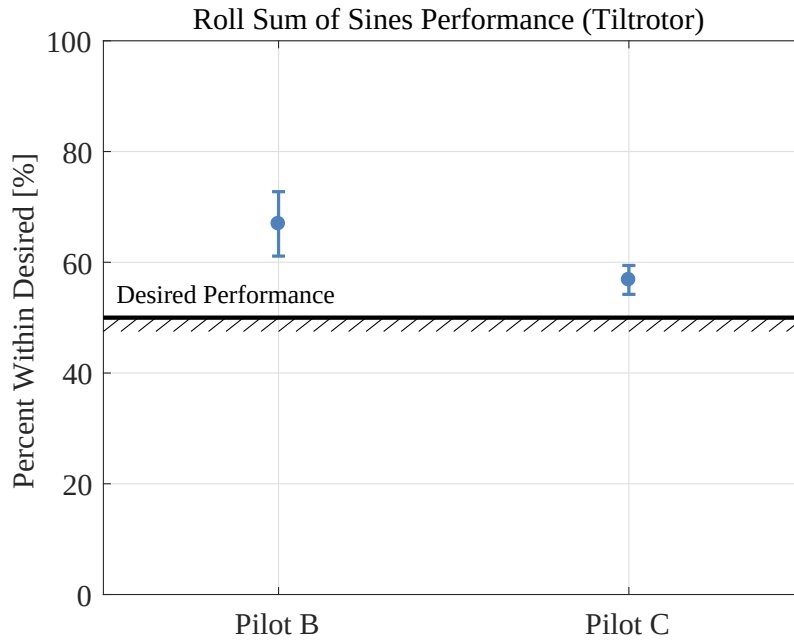


Figure 4.19. Roll Sum-of-Sines Tracking performance (tiltrotor, 180 kts).

noted that he was putting a step input on the thrust control lever one second before starting the turn, which is evident by the increase in airspeed in Pilot A's runs in Figure 4.20.

Pilot E rated this MTE an HQR 4, although he used a different control strategy than Pilot A. Pilot E noted that he decreased airspeed going into the turn in order to increase turn rate. This led to Pilot E having slightly shorter time to complete the maneuver than Pilot A, as seen by Pilot E's runs in Figure 4.20. Even though the airspeed dropped below desired performance during Pilot E's runs, he was able to bring airspeed back into desired at the end of the turn and therefore meet desired performance.

Figure 4.21 shows the pilot stick activity (cutoff frequency ω_{co} and RMS) for both the lateral and longitudinal axes during the Break Turn maneuver. Both pilots have similar stick activity between the lateral and longitudinal axes, and in Pilot E's case between the TCL axis, suggesting that although this is primarily a roll task, speed and altitude management requires comparable pitch inputs. Neither pilot had any activity on the pedals, thus suggesting that turn coordination worked well.

4.3.6 High-Speed Acceleration/Deceleration

Pilot E flew the High-Speed Acceleration/Deceleration maneuver with the tiltrotor. Figure 4.22 shows time histories for all of Pilot E's record runs for the acceleration portion of the maneuver. Pilot E was able to meet desired performance on all pa-

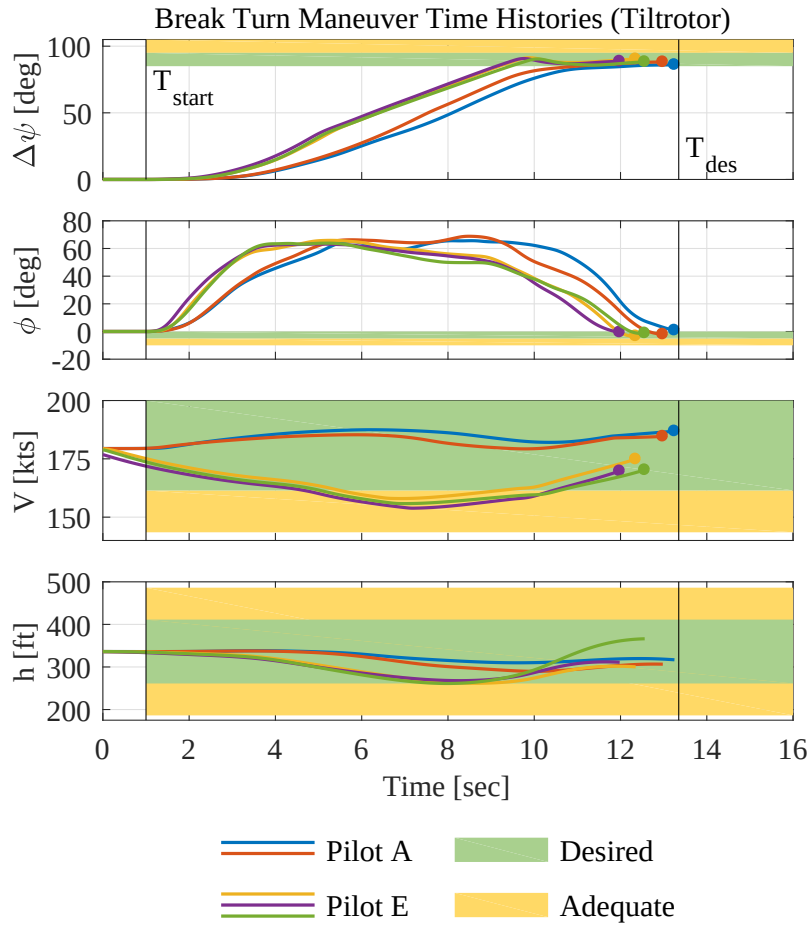


Figure 4.20. Break Turn task performance (tiltrotor, 180 kts).

rameters for both of his acceleration runs. Pilot E rated the acceleration portion of this MTE an HQR 4 and commented that “aggressiveness was required in power and longitudinal [stick inputs] to maintain altitude and airspeed.” Pilot E further commented that this was “primarily an altitude maneuver” that required “constant longitudinal control activity.” Figure 4.22 also shows the tiltrotor pitch attitude θ throughout the acceleration maneuver. Since the tiltrotor transitions from rotor-borne to wing-borne flight during this maneuver, the pilot must trim the aircraft angle of attack α (and therefore pitch attitude θ) as airspeed changes.

Figure 4.23 shows time histories for all of Pilot E’s record runs for the deceleration portion of the maneuver. Pilot E had an initial run that was beyond the desired time T_{des} but did meet the adequate time requirement. However, his remaining two record runs were all within desired. Pilot E assigned the deceleration portion of the maneuver an HQR 4. He commented that he was “able to meet desired” and be aggressive with the nacelle rates and power inputs. He also noted that he was “focused on height control” and “cross checked [the Vertical Speed Indicator]

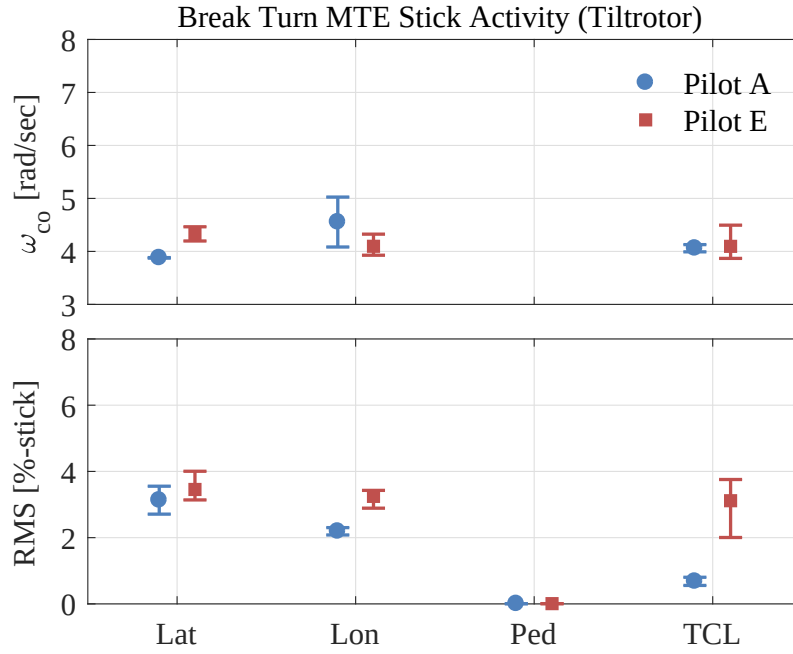


Figure 4.21. Break Turn stick activity (tiltrotor, 180 kts).

to meet the altitude tolerance.” Figure 4.23 also shows pitch attitude θ throughout the deceleration maneuver, and as with the acceleration portion, the pilot had to constantly adjust pitch attitude throughout the maneuver.

Figure 4.24 shows the pilot longitudinal stick and TCL activity (cutoff frequency ω_{co} and RMS) during the acceleration and deceleration maneuvers. The pilot had to trim pitch attitude with the longitudinal stick and control airspeed with the TCL throughout the maneuvers, and the resulting average stick RMS was about 3% of stick deflection, with a cutoff frequency $\omega_{co} \approx 2$ rad/sec.

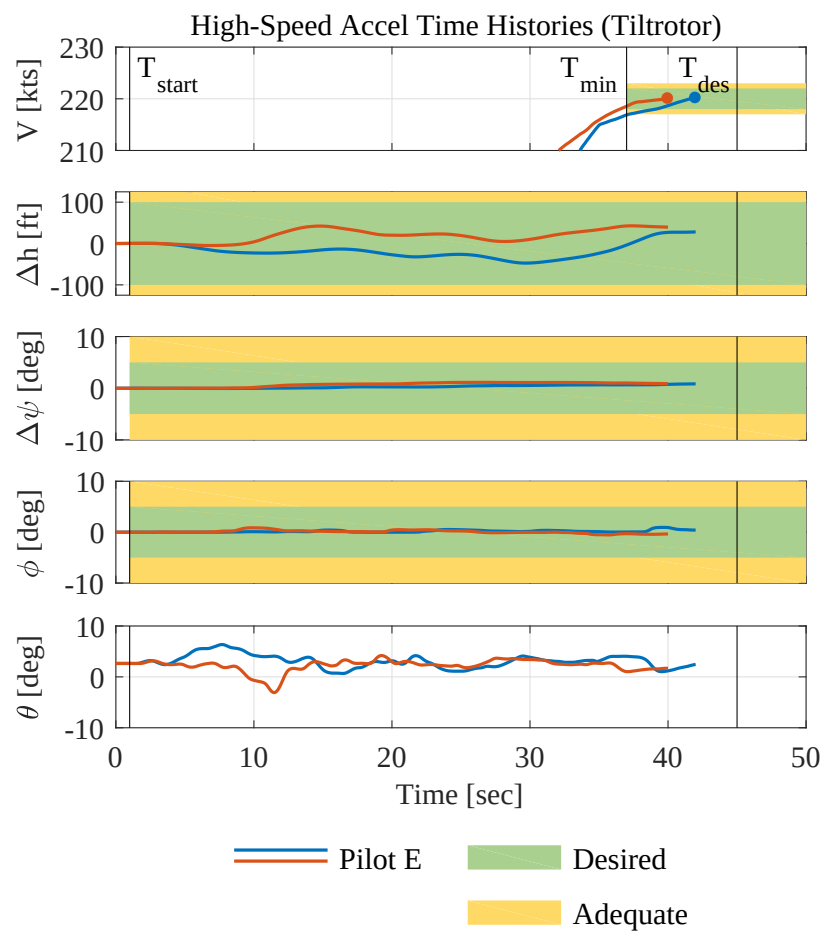


Figure 4.22. High-Speed Acceleration task performance (tiltrotor, 50–220 kts).

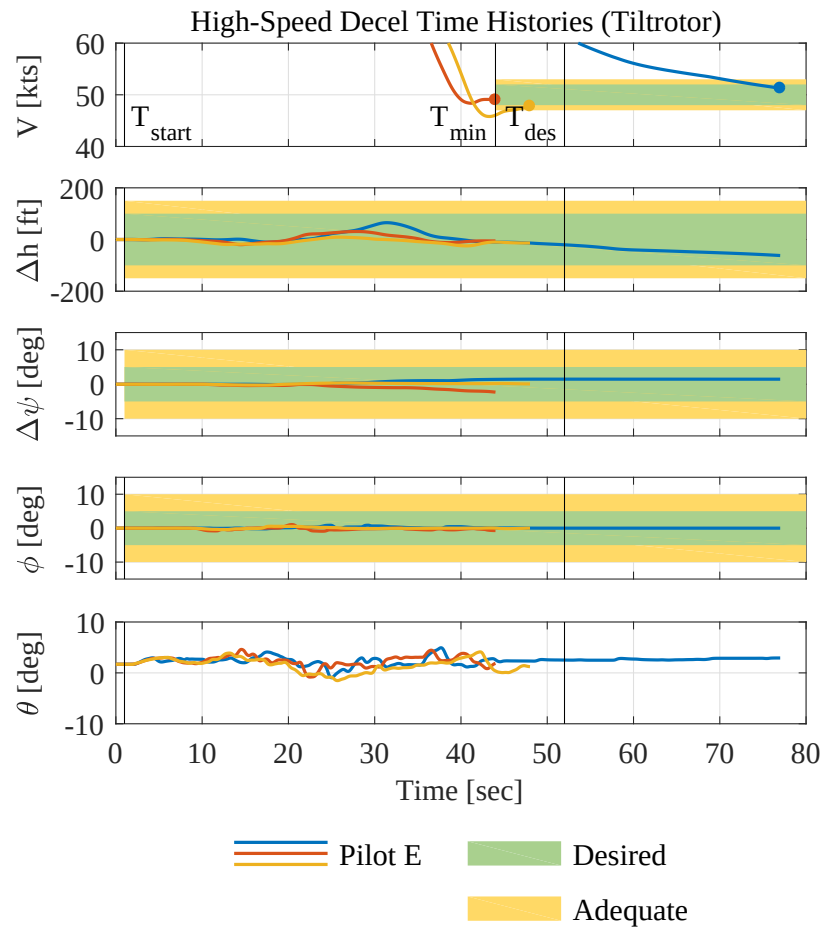


Figure 4.23. High-Speed Deceleration task performance (tiltrotor, 220–50 kts).

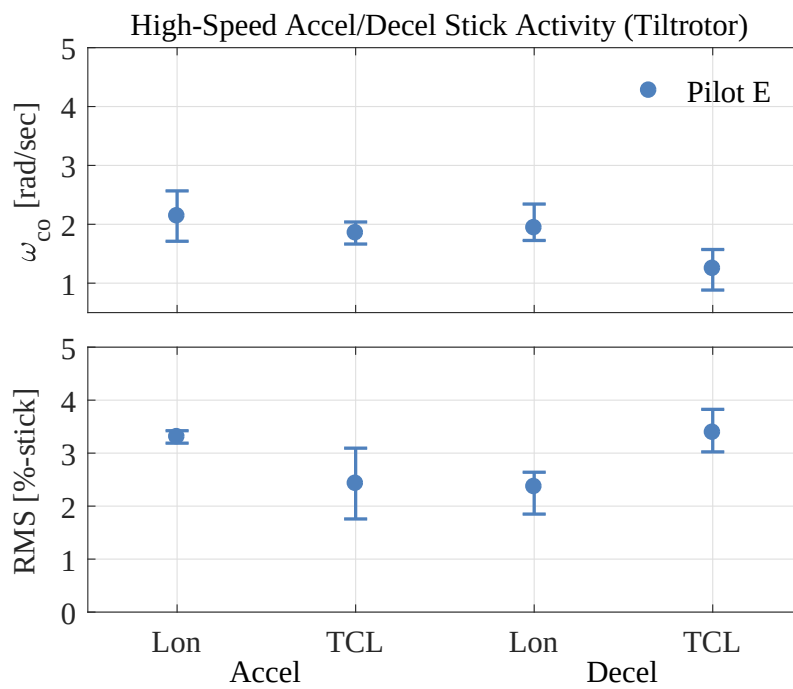


Figure 4.24. High-Speed Acceleration/Deceleration stick activity (tiltrotor).

4.4 Discussion

4.4.1 Coaxial-Pusher Tip Clearance Controller

Throughout the handling qualities experiment in the VMS, the coaxial-pusher model logged over four hours of simulation time, during which the minimum rotor separation was 1.2 ft. In a separate VMS simulation experiment using the models and control systems described here, the coaxial-pusher model logged over 20 hours of simulation time, during which the minimum rotor separation was 1.5 ft. This demonstrates the success of the tip clearance feed-forward controller designed for the coaxial-pusher (described in Sec. 3.5.1).

4.4.2 MIL-STD-1797B Specifications

Handling-qualities specifications from the fixed-wing requirement MIL-STD-1797B worked well to supplement ADS-33E high-speed requirements for both aircraft. In some cases where there is overlap, the MIL-STD-1797B requirements for Category A, Class I are more stringent than the ADS-33E High Agility/Target Acquisition & Tracking requirements, such as for pitch attitude bandwidth and flight path bandwidth/lag.

The lower-order equivalent system (LOES) specifications from MIL-STD-1797B were applied to both aircraft, when LOES fits had cost functions $J_{\text{LOES}} \leq 50$, indicating an excellent match between the lower-order system and high-order aircraft closed-loop response [83]. For both aircraft, pitch LOES costs were $J_{\text{LOES}} < 50$ above $V_{\text{CAS}} = 80$ kts and $J_{\text{LOES}} < 10$ above $V_{\text{CAS}} = 130$ kts, indicating a classical fixed-wing-like response for both RCAH and n_z -command response types. In the roll and yaw axes, $J_{\text{LOES}} < 10$ above $V_{\text{CAS}} = 80$ kts for both aircraft.

4.4.3 Pitch Rate Command Versus Normal Acceleration Command

The differences in average HQRs for the Pitch Angle Capture and Hold and Pitch Sum-of-Sines Tracking tasks between the pitch rate command response type and normal acceleration command response types was $\Delta\text{HQR} = 0.3$ for the coaxial-pusher and $\Delta\text{HQR} = 0.1$ for the tiltrotor. These small difference for both aircraft, shows that both response types work well for the MTEs tested.

For both aircraft, pilots did note during the Pitch Sum-of-Sines Tracking MTE that the pitch response was too heavily damped. This correlates well with the high equivalent short period damping ζ_{sp} seen in the Control Anticipation Parameter (CAP) specification, as well as the Level 2 flight path bandwidth values for the RCAH designs. However, since the pitch rate command model is first-order, there is no damping term to tune. A higher-order pitch rate command model, that tracks a classic

first-over-second-order pitch rate response, should be investigated (as done later in Chapter 7).

This result also indicates that the CAP or flight path bandwidth specifications should be Tier 1 specification in both coaxial-pusher and tiltrotor designs.

4.4.4 Pitch Stick Contamination During Roll MTEs

For both aircraft, three out of the four pilots that flew the Roll Sum-of-Sines MTE noted pitch excursions/coupling. This issue was not noted for the Bank Angle Capture and Hold MTE, though, which is a task that requires more discrete lateral stick inputs.

One pilot noted that during the Bank Angle Capture and Hold MTE, he inadvertently put in pitch inputs while applying large roll inputs. Since in all cases, the aircraft pitch rates tracked the commanded rates, it appears the coupling issues were caused by stick inputs. It is possible that the pilots did not notice the stick contamination during the continuous tracking of the Sum-of-Sines tasks, unlike the discrete inputs of the Capture and Hold task.

4.4.5 Pilot-Vehicle System Comparison

The control laws for both aircraft use the same architecture and were optimized to meet the same set of specifications. Therefore, it was expected that they would have similar Handling Qualities Ratings, which was the case for tasks that did not require a configuration change (e.g., changing nacelle position in the tiltrotor).

One way to quantitatively compare how pilots flew both aircraft is to investigate the characteristics of the pilot Y_p vehicle Y_c system (PVS) for a given task, shown in Figure 4.25. In the case of the Sum-of-Sines Tracking tasks, the forcing function of the PVS is known (the pitch attitude target signal θ_{cmd}), and so all elements of the PVS can be identified and compared between the two aircraft. Since Pilot C flew the Pitch Sum-of-Sines Tracking task using both aircraft and had similar performances (Table 4.2), his data will be used for this comparison.

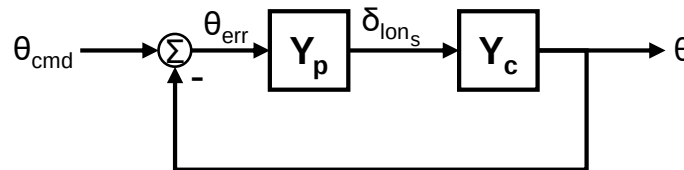


Figure 4.25. Pilot Y_p -Vehicle Y_c System (PVS) for pitch tracking task.

Table 4.2. Pilot-Vehicle Parameters for Pitch Sum-of-Sines Tracking Task (180 kts)

	Coaxial-	
	Pusher	Tiltrotor
<i>Pilot-Vehicle System Parameters ($Y_p Y_c$)</i>		
Pilot-Vehicle Crossover		
Frequency ω_c [rad/sec]	2.32	2.37
Gain Margin [dB]	3.84	3.35
Phase Margin [deg]	32.18	28.80
Pilot-Vehicle Equivalent		
Time Delay τ [sec]	0.43	0.45
<i>Pilot Parameters (Y_p)</i>		
Pilot Gain K_p	13.50	14.43
Time Delay τ_p [sec]	0.23	0.29
Cutoff Frequency ω_{co} [rad/sec]	1.25	1.23
Stick RMS [%]	11.32	12.23
<i>Performance</i>		
% Within Desired	75.14	79.54

Figure 4.26 shows the PVS broken-loop frequency response ($Y_p Y_c$) extracted from the time history data using CIPHER[®] at the discrete frequencies used in the tracking signal. The figure also shows a crossover model [108] fit to the each response, given by:

$$Y_p Y_c = \frac{\omega_c}{s} e^{-\tau s} \quad (4.7)$$

where:

- Y_p denotes the pilot response ($\delta_{lon_s}/\theta_{err}$),
- Y_c denotes the aircraft response (θ/δ_{lon_s}),
- ω_c is the PVS crossover frequency, and
- τ is the equivalent time delay.

The parameters of the crossover model are given in Table 4.2 for both aircraft.

The extracted PVS broken-loop frequency responses of both aircraft match very well, as do the crossover models fit to each response. In fact, as shown in Table 4.2, the PVS crossover frequency ω_c , which is interpreted as the fundamental frequency of piloted control inputs for closed-loop regulation [83], has only a 2% difference between both aircraft. In addition, the PVS crossover models for both aircraft have similar gain margins, phase margins, and time delays.

To further investigate if the pilot was altering his compensation between the two aircraft in order to get a similar PVS response, the individual elements of the PVS were

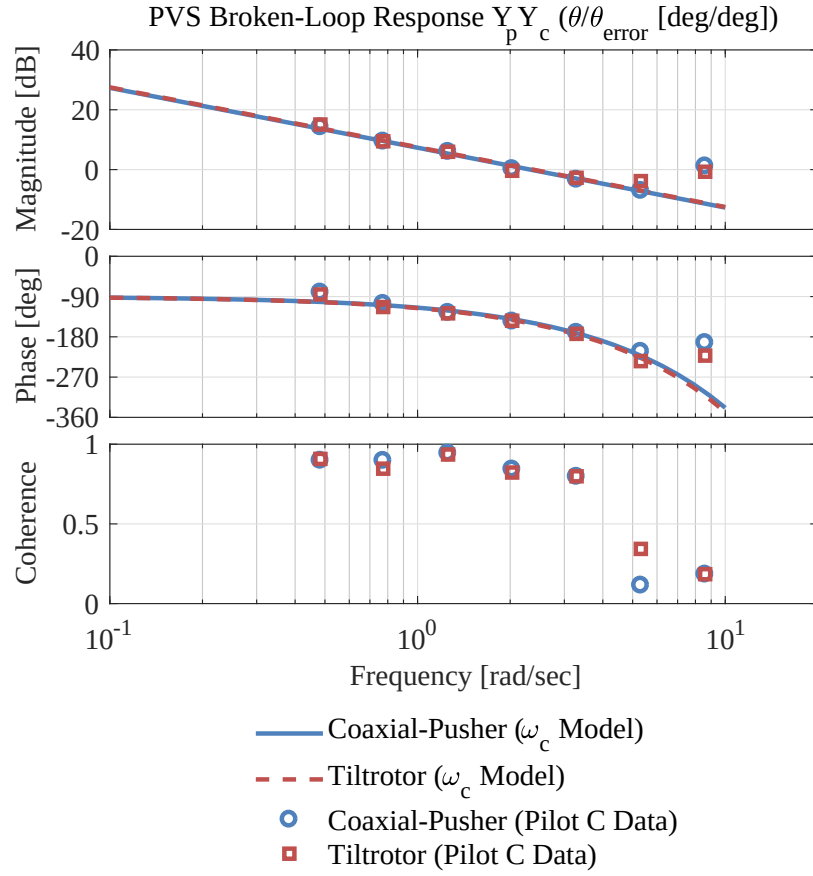


Figure 4.26. Pilot-Vehicle System (PVS) broken-loop frequency responses comparison for Pitch Sum-of-Sines Tracking task (coaxial-pusher and tiltrotor, 180 kts).

investigated. Figure 4.27 shows the frequency response of both aircraft Y_c extracted from their respective linear models and from the tracking task data at the discrete frequencies used in the tracking signal. As expected, both aircraft have nearly identical closed-loop frequency responses, since their control systems use the same command models and were tuned to meet the same set of specifications.

Finally, Figure 4.28 shows the pilot Y_p frequency response of both aircraft extracted from the tracking task data. In addition, Table 4.2 lists the parameters of a simple pilot model given by:

$$Y_p = K_p e^{\tau_p s} \quad (4.8)$$

that was fit to the data for both aircraft. Table 4.2 also lists the pilot cutoff frequency (ω_{co}) as well as the pilot stick RMS. Based on the pilot frequency responses and pilot model parameters, the pilot behavior between the two aircraft was very similar.

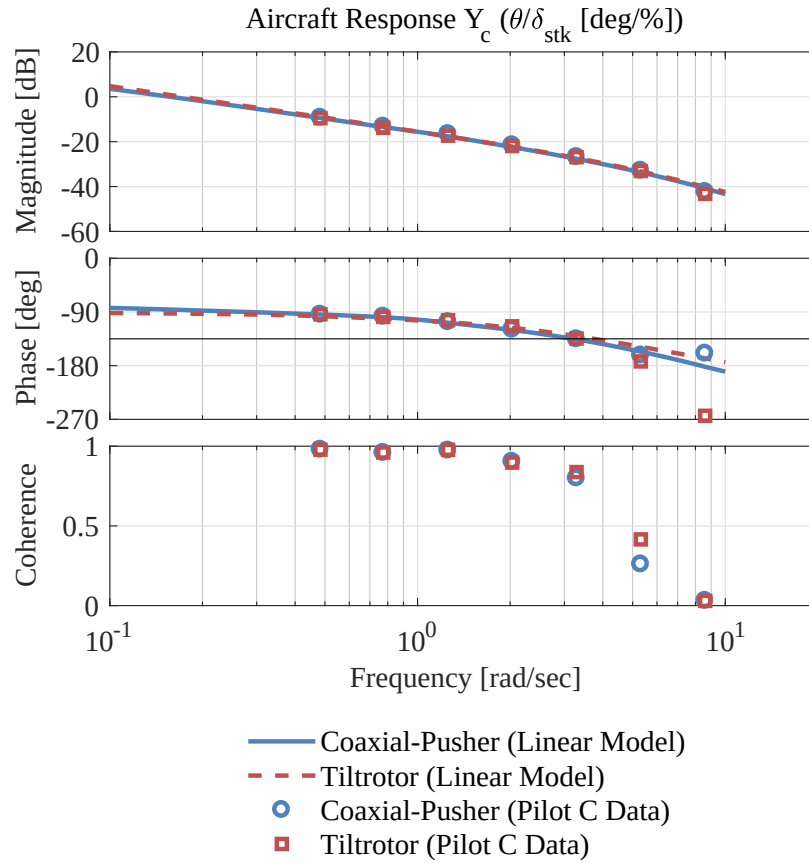


Figure 4.27. Vehicle frequency responses comparison for Pitch Sum-of-Sines Tracking task (coaxial-pusher and tiltrotor, 180 kts).

4.4.6 Outer Loops

For both aircraft, pilots were asked which outer-loop modes would help improve performance and/or reduce pilot workload for the MTEs. For the coaxial-pusher, pilots answered that airspeed hold would help for the Sum-of-Sines Tracking tasks and altitude hold for the Break Turn MTE. For the tiltrotor, pilots answered that altitude hold would help for the High-Speed Acceleration/Deceleration task. These answers correlate well with high pilot longitudinal stick RMS values for the Break Turn MTE in the coaxial-pusher and high longitudinal stick and TCL RMS values for the tiltrotor during the High-Speed Acceleration/Deceleration task.

4.4.7 High-Speed MTEs

This simulation experiment was the first use of the newly proposed high-speed MTEs in the VMS and by many of the participating pilots. When providing the pilot ratings

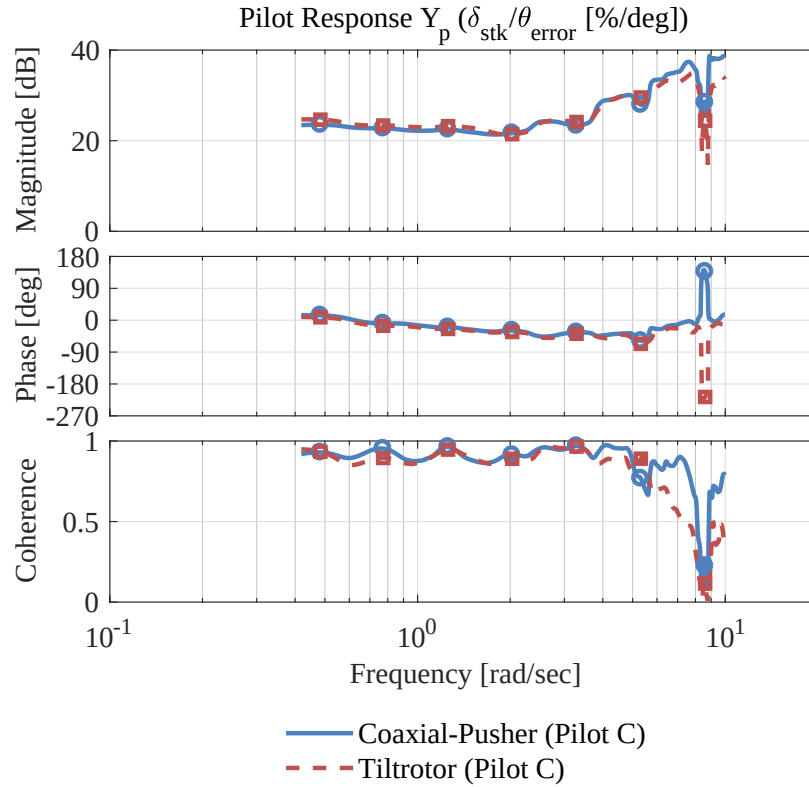


Figure 4.28. Pilot frequency responses comparison for Pitch Sum-of-Sines Tracking task (coaxial-pusher and tiltrotor, 180 kts).

and comments, pilots were also asked to comment on the appropriateness of the MTEs. In all cases, pilots commented that the MTEs were appropriate for assessing the handling qualities of these aircraft. Pilots commented that the BACH and PACH MTEs were appropriate to see stick characteristics, damping, and bandwidth/phase delay.

Pilots noted that the desired tolerance on the Pitch SOS Tracking MTE was too easy. It would be advisable to use the Precision, Aggressive version of this MTE moving forward, instead of the Precision, Non-Aggressive version used here. Pilots did note that the Roll SOS Tracking MTE seemed reasonable.

For the Break Turn MTE, it was noted that the desired timing tolerance may be too tight, and this MTE received the worst HQR recorded (5.5).

4.5 Summary

This chapter described the handling qualities assessment of inner-loop flight control systems for the generic lift offset coaxial and tiltrotor rotorcraft. A piloted sim-

ulation experiment was conducted in the NASA Ames Vertical Motion Simulator facility with participation from five Army experimental test pilots. Seven high-speed handling-qualities demonstration maneuvers were used: Pitch and Bank Attitude Capture and Hold, Pitch and Roll Sum-of-Sines Tracking, Break Turn, and High-Speed Acceleration/Deceleration. Overall, Level 1 handling qualities were assigned to the coaxial-pusher with inner-loop control laws in transition and high-speed, while the tiltrotor with inner-loop control laws was assigned Level 1 handling qualities at high-speed and Level 2 in transition.

Chapter 5 |

Outer-Loop Control Law Design

The control laws described thus far in Chapters 3 and 4 are considered *inner-loop* control laws in that they control the aircraft angular rates and attitudes. *Outer-loop* control laws to control speed and flight path/climb rate were developed to be wrapped around the inner-loop control laws. The outer-loop control laws were designed and optimized for both coaxial-pusher and tiltrotor aircraft across their full flight envelopes, however piloted simulation testing (described later in Chapter 6) concentrated on the high-speed and transition portion of the flight envelopes.

This chapter provides a description of the outer-loop control laws that were designed for both coaxial-pusher and tiltrotor aircraft. First, the design approach and architecture used for the outer loops will be described. Next, the design specifications used for control law optimization will be provided. Control law optimization results will be presented for both coaxial-pusher and tiltrotor, first at one flight condition, and then at several flight conditions spanning their respective flight envelopes. Finally, a discussion will be provided.

5.1 Design Approach

For both aircraft configurations, with the inner-loop control laws closed around the bare-airframe there are three inputs to control airspeed V and climb rate $\dot{h}$:

1. Pitch attitude θ
2. Symmetric main rotor collective δ_{col}
3. Pusher propeller collective δ_{prop} (coaxial-pusher) or nacelle tilt angle δ_{nac} (tiltrotor)

Which of these inputs affects which state (airspeed or climb rate) and by how much is a function of flight condition and aircraft configuration/orientation. During manual control of the closed inner-loop system, the pilot must control the two states with the

three available controls, which was seen to increase workload in the VMS handling qualities simulation experiment described in Chapter 4. Therefore, outer-loop control laws were designed for both coaxial-pusher and tiltrotor to control airspeed and climb rate.

A dynamic inversion (DI) control system architecture [50, 109] was used for both aircraft. Dynamic inversion was used for the outer loop instead of explicit model following as was done for the inner loop for two reasons: 1) It provides the ideal feed-forward and feedback cross-feeds between the two coupled control channels and 2) To stay consistent with previous work done at Penn State (e.g., [60, 110]).

A top level representation of the DI outer-loop control laws is shown in Figure 5.1. The control system consists of five main elements:

1. Inner-Loop Control System
2. Dynamic Inversion [CA and $(CB)^{-1}$ blocks] and Control Allocation
3. Command Model
4. Command Delay
5. Feedback

The following sections will describe each element in more detail.

5.1.1 Inner-Loop Control System

The inner-loop control system is based on that presented in Chapter 3 with two modifications. First, a second pitch axis command model was added to the inner-loop control system for the outer-loop control system to command. This command model can be tuned separately from the inner-loop pitch axis command model that governs the response to pilot input, and was used to achieve an attitude command response type. For the outer-loop pitch attitude command θ_{cmOL} passed to the inner loop, a second-order pitch attitude command model was used (as opposed to the first-order pitch rate command model used for pilot command):

$$\frac{\theta_{\text{cm}}}{\theta_{\text{cmOL}}} = \frac{\omega_{\text{loncm}}^2}{s^2 + 2\zeta_{\text{loncm}}\omega_{\text{loncm}} + \omega_{\text{loncm}}^2} \quad (5.1)$$

where $\omega_{\text{loncm}} = 10$ rad/sec and $\zeta_{\text{loncm}} = 1$. These values for frequency and damping were chosen to ensure the inner loop was a fast enough actuator for the outer loop based on the anticipated outer-loop crossover frequency ($\omega_c = 1$ rad/sec), and to ensure that the inner-loop pitch axis stability margins did not degraded into Level 2 with the outer loops closed.

The second modification made to the inner-loop control system was to extend the pitch Rate-Command/Attitude-Hold (RCAH) range for the coaxial-pusher from hover-200 kts to hover-260 kts, and eliminate the normal acceleration command mode. In the case of the tiltrotor, pitch RCAH was kept for an airspeed range of

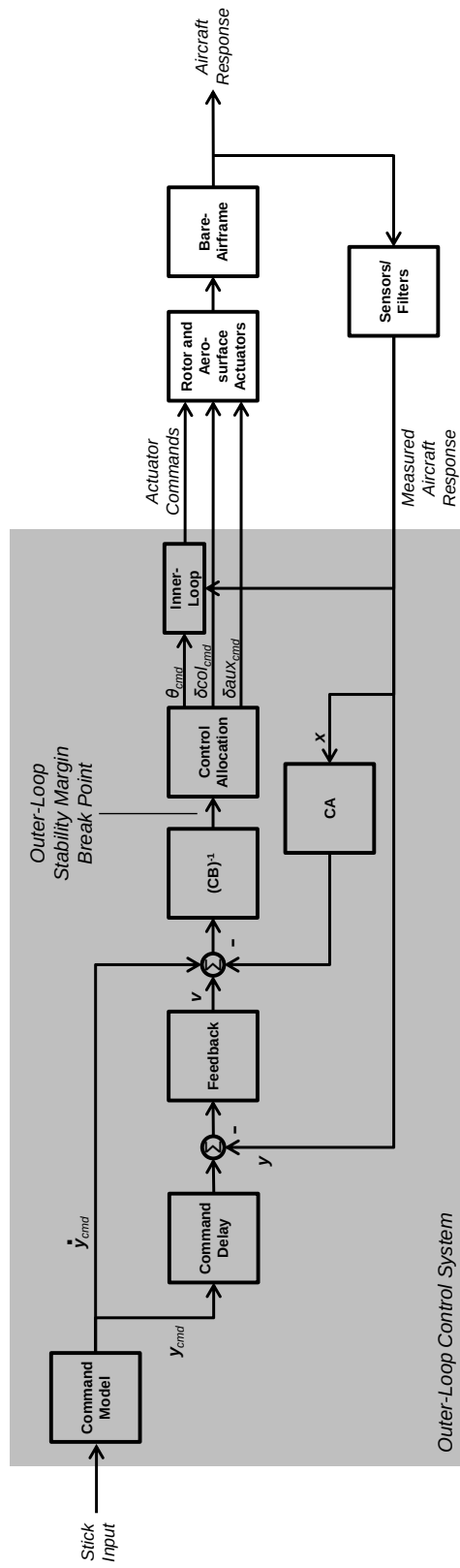


Figure 5.1. Top level block diagram representation of outer-loop control system.

hover–200 kts, and normal acceleration command was kept for 200–300 kts. For the normal acceleration command portion of the tiltrotor flight envelope, the pitch attitude command generated by the outer-loop control system θ_{cmOL} was converted to a normal acceleration command $n_{z\text{cmOL}}$ using the following kinematic relationship [95]:

$$\frac{n_{z\text{cmOL}}}{\theta_{\text{cmOL}}} = \frac{(V_{\text{TAS}}/g)s}{T_{\theta_2}s + 1} \quad (5.2)$$

where T_{θ_2} is the flight path-attitude lag, and is taken as equal to $T_{\theta_2} = -1/Z_w$ based on the bare-airframe rigid-body (or quasi-steady) value of Z_w . (See Sec. 2.1.2.3 for derivation of the quasi-steady model.)

Figure 5.2 shows the actual normal acceleration n_z to pitch attitude θ frequency response for the tiltrotor with inner-loop control system closed at $V_{\text{CAS}} = 250$ kts, determined by:

$$\frac{n_z}{\theta} = \frac{n_z/\delta_{\text{lon}s}}{\theta/\delta_{\text{lon}s}} \quad (5.3)$$

as well as the response of the lower order model of Eq. 5.2. The transfer function from the simple kinematic relationship matches the actual aircraft response well up to a frequency of $\omega = 10$ rad/sec. This well covers the frequency range at which the outer loops will operate.

The outer-loop normal acceleration command $n_{z\text{cmOL}}$ was then passed to the inner loop through a second-order normal acceleration command model:

$$\frac{n_{z\text{cm}}}{n_{z\text{cmOL}}} = \frac{\omega_{\text{loncm}}^2}{s^2 + 2\zeta_{\text{loncm}}\omega_{\text{loncm}}s + \omega_{\text{loncm}}^2} \quad (5.4)$$

where $\omega_{\text{loncm}} = 7$ rad/sec and $\zeta_{\text{loncm}} = 1$.

Figure 5.3 shows the tiltrotor pitch attitude θ response to the outer-loop pitch attitude command θ_{cmdOL} at $V_{\text{CAS}} = 250$ kts, demonstrating the excellent command tracking performance of the inner loop.

These modifications to the inner-loop control system make the inner loop appear as a pitch attitude actuator to the outer-loop control system. Next, the dynamic inversion controller closed around the inner loop will be discussed.

5.1.2 Dynamic Inversion and Control Allocation

The dynamic inversion (DI) section of the outer-loop control system is comprised of the $\mathbf{CA}$ and $(\mathbf{CB})^{-1}$ blocks in Figure 5.1. These form a feedback loop meant to cancel the plant dynamics (in this case the closed-loop bare-airframe plus inner-loop control system) and make the relationship between the output variables to be controller ($\mathbf{y} = [V \dot{h}]^T$) and the inputs ($\mathbf{u}$) appear as a set of decoupled integrators [50, 109].

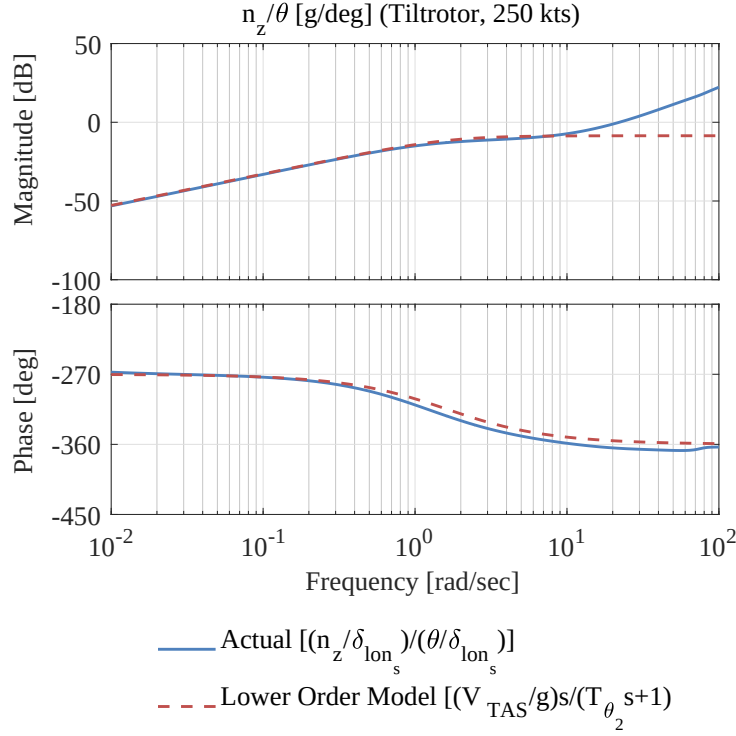


Figure 5.2. Normal acceleration response to pitch attitude (tiltrotor, 250 kts).

5.1.2.1 Dynamic Inversion Model

Since dynamic inversion requires full state feedback, a lower-order (two state, three input) model is used to represent the airspeed V and climb rate $\dot{h}$ dynamics of the bare-airframe with inner loop closed:

$$\begin{bmatrix} \dot{V} \\ \ddot{h} \end{bmatrix} = \underbrace{\begin{bmatrix} X_V & X_{\dot{h}} \\ Z_V & Z_{\dot{h}} \end{bmatrix}}_{\mathbf{A}} \begin{bmatrix} V \\ \dot{h} \end{bmatrix} + \underbrace{\begin{bmatrix} X_{\theta} & X_{\delta_{col}} & X_{\delta_{aux}} \\ Z_{\theta} & Z_{\delta_{col}} & Z_{\delta_{aux}} \end{bmatrix}}_{\mathbf{B}} \begin{bmatrix} \theta \\ \delta_{col} \\ \delta_{aux} \end{bmatrix} \quad (5.5)$$

where δ_{aux} represents pusher propeller collective δ_{prop} in the case of the coaxial-pusher and nacelle angle δ_{nac} in the case of the tiltrotor. Since the two states of the system V and $\dot{h}$ are also the outputs to be controller, $\mathbf{C} = \mathbf{I}$.

There are several methods to determine the parameters of the system in Eq. 5.5 from the bare airframe and inner-loop control system. The first method is a simplified analytical method [60,110] where the bare-airframe model is reduced to the rigid-body six degrees-of-freedom (quasi-static) dynamics (Sec. 2.1.2.3). Then, the stability and control derivatives of the reduced-order bare-airframe are used directly, dropping cross

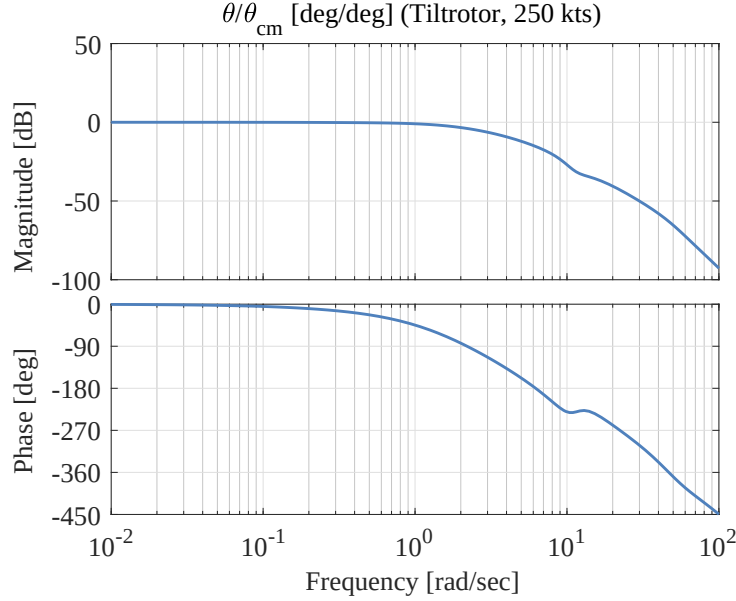


Figure 5.3. Pitch attitude response to outer loop pitch attitude command (tiltrotor, 250 kts).

coupling stability derivatives as in Refs. 60 and 110:

$$\begin{bmatrix} \dot{V} \\ \ddot{h} \end{bmatrix} = \begin{bmatrix} X_u & 0 \\ 0 & Z_w \end{bmatrix} \begin{bmatrix} V \\ \dot{h} \end{bmatrix} + \begin{bmatrix} -g \cos \theta_0 & X_{\delta_{col}} & X_{\delta_{aux}} \\ -(Z_w V - g \sin \theta_0) & -Z_{\delta_{col}} & -Z_{\delta_{aux}} \end{bmatrix} \begin{bmatrix} \theta \\ \delta_{col} \\ \delta_{aux} \end{bmatrix} \quad (5.6)$$

The second method to determine the stability and control derivatives in Eq. 5.5 is by starting with the reduced-order (rigid body dynamics only) bare-airframe and applying a constrained coupling-numerator solution [95] to account for the fact that the inner-loop control system will be closed and regulating the aircraft rotational rates (p , q , and r). Then the stability and control derivatives of the constrained bare-airframe are used in Eq. 5.6.

Finally, system identification (ID) can be performed on the closed inner-loop system to identify the derivatives in Eq. 5.5. In this case, a frequency domain approach using the CIFER [83] tool was used to identify the derivatives across a frequency range of $\omega = 0.001 - 1$ rad/sec. All three methods were investigated, and example results are shown here for the coaxial-pusher at $V_{CAS} = 180$ kts, with the measure of goodness being how well the dynamic inversion worked to make the relationship between the output variables and the inputs appear as a set of decoupled integrators.

Table 5.1 lists the parameter values using the three methods described for the

Table 5.1. Dynamic Inversion System Parameters (Coaxial-Pusher, 180 kts)

Parameter	Analytical	Coupling Numerator	CIFER ID
<i>A-matrix</i>			
X_V	-0.0446	-0.0505	-0.0631
$X_{\dot{h}}$	0	-0.0234	-0.0501
Z_V	0	0.000718	0.243
$Z_{\dot{h}}$	-1.394	-1.394	-1.198
<i>B-matrix</i>			
X_θ	-0.562	-0.437	-0.308
$X_{\delta_{\text{col}}}$	-0.0539	0.0157	0.0910
$X_{\delta_{\text{prop}}}$	0.440	0.438	0.443
Z_θ	7.39	7.39	6.45
$Z_{\delta_{\text{col}}}$	-1.0	-1.0	-2.816
$Z_{\delta_{\text{prop}}}$	-0.00121	-0.00121	0.0144

coaxial-pusher at $V_{\text{CAS}} = 180$ kts. Figures 5.4 through 5.8 show the frequency responses of the actual closed-loop bare-airframe plus inner-loop control system (labeled Actual), overlaid with the responses of the lower-order model (Eq. 5.5) as determined using the analytical method (labeled Analytical), coupling numerator method (labeled Coupling Numerator), and system identification method (labeled CIFER ID).

The largest discrepancies are seen in the responses to collective input δ_{col} . For the airspeed response to collective V/δ_{col} (Figure 5.5), the analytical method predicts a response with the wrong sign (phase off by 180 deg), while the coupling numerator (which assumed infinitely tight pitch rate control by the inner loop, and thus ignores the existing collective to pitch coupling) underestimates the magnitude of the response. The CIFER ID method accurately predicts the response across a broad frequency range. For the climb rate response to collective $\dot{h}/\delta_{\text{col}}$ (Figure 5.8), the Analytical and Coupling Numerator methods both under-predict the magnitude of the response. The actual response has a higher magnitude due to the bare-airframe pitch to collective coupling that is not fully eliminated by the inner pitch loop. Finally, the CIFER ID method accurately captures the low frequency droop in the climb rate to pitch attitude $\dot{h}/\theta$ frequency response (Figure 5.7).

Figure 5.10 shows airspeed and climb rate step responses of the actual and lower-order systems. As with the frequency responses, the CIFER ID method produced the most accurate model.

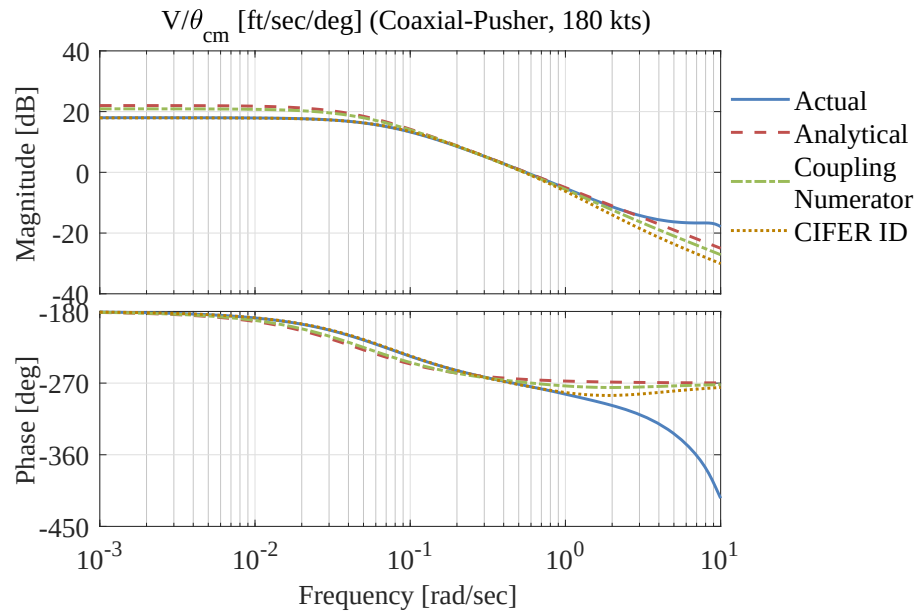


Figure 5.4. Airspeed response to pitch attitude command (coaxial-pusher, 180 kts).

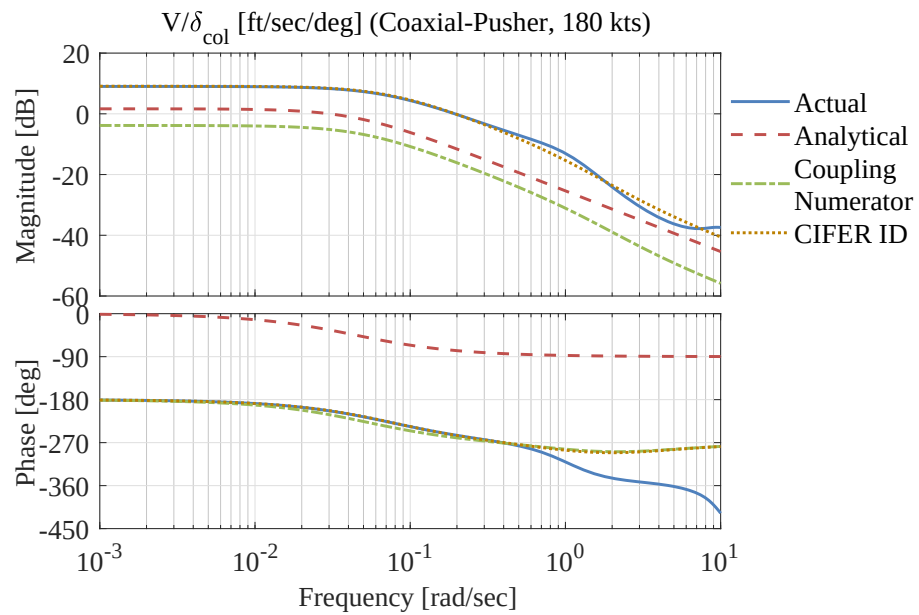


Figure 5.5. Airspeed response to symmetric main rotor collective input (coaxial-pusher, 180 kts).

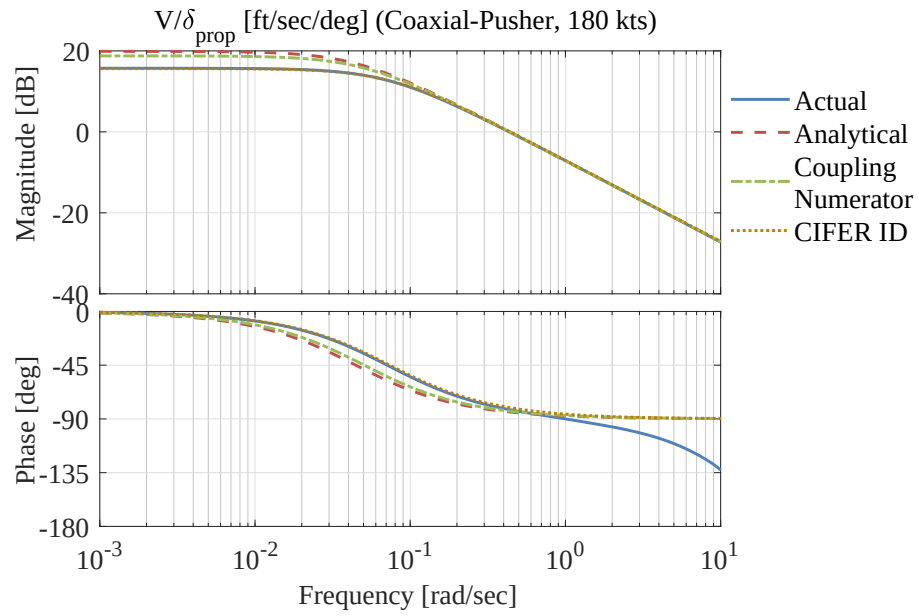


Figure 5.6. Airspeed response to pusher propeller collective input (coaxial-pusher, 180 kts).

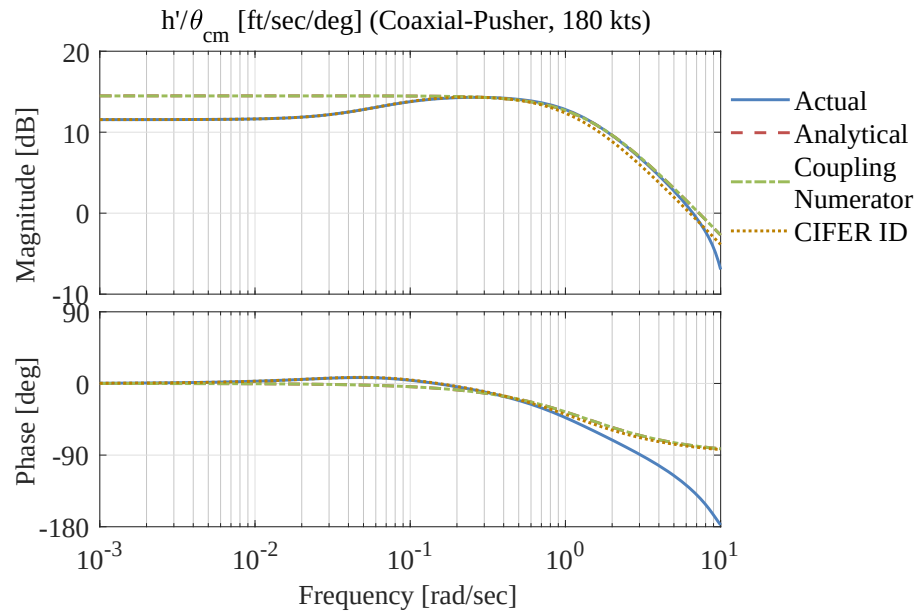


Figure 5.7. Climb rate response to pitch attitude command (coaxial-pusher, 180 kts).

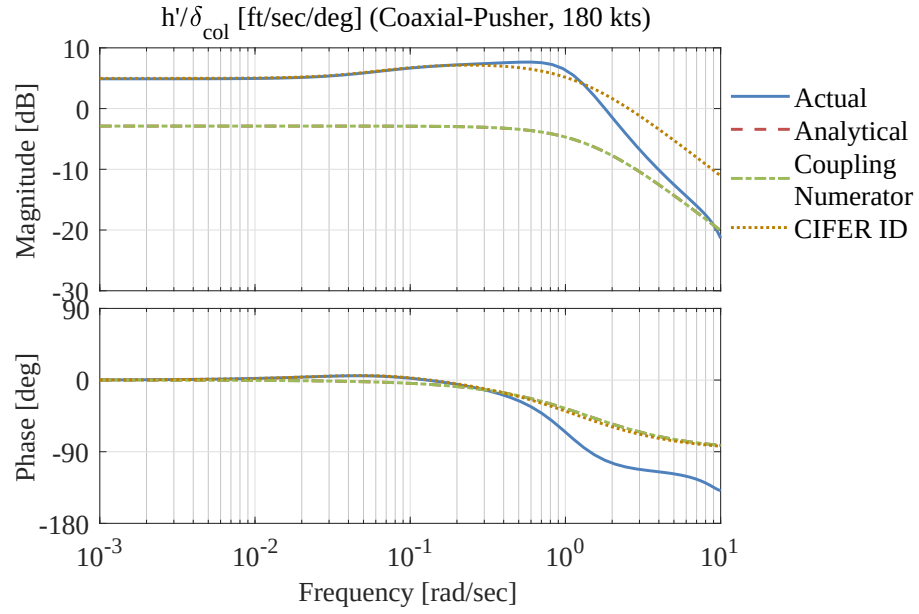


Figure 5.8. Climb rate response to symmetric main rotor collective input (coaxial-pusher, 180 kts).

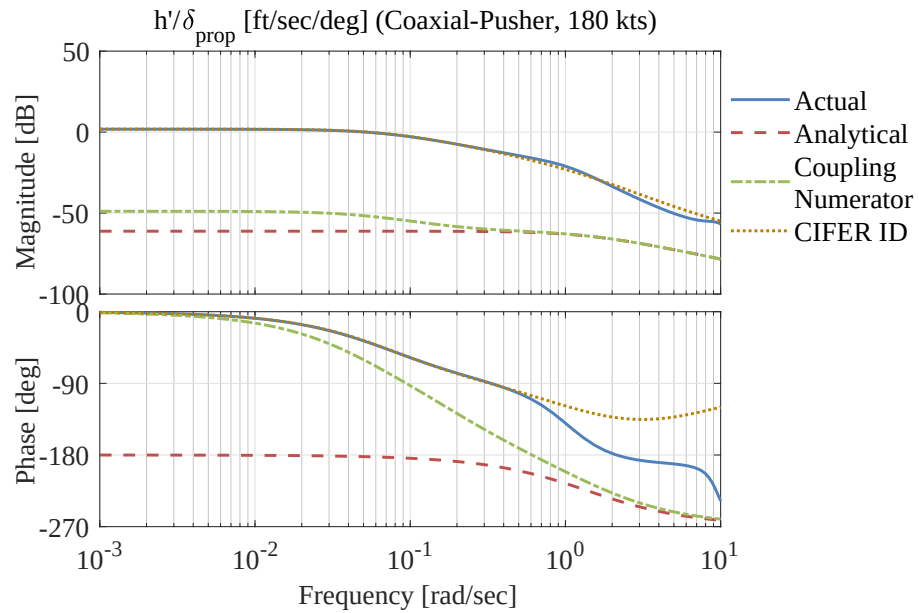


Figure 5.9. Climb rate response to pusher propeller collective input (coaxial-pusher, 180 kts).

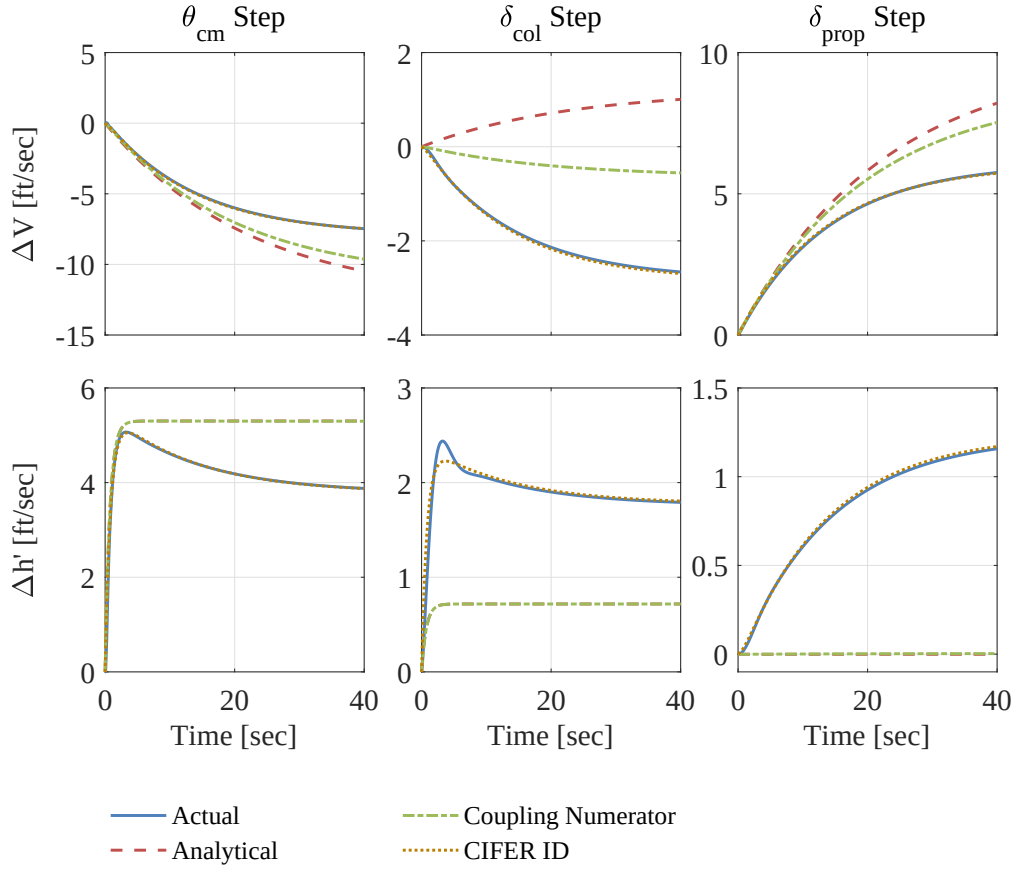


Figure 5.10. Airspeed and climb rate step responses to pitch attitude command, symmetric main rotor collective, and pusher propeller collective inputs (coaxial-pusher, 180 kts).

5.1.2.2 Control Allocation

A square system is required for dynamic inversion, however the system representing the closed-loop bare-airframe plus inner-loop control system shown in Eq. 5.5 has a redundant input. Therefore, a control allocation scheme is required. Different approaches were used for the coaxial-pusher and tiltrotor, which are discussed in the following sections.

5.1.2.2.1 Coaxial-Pusher

For the coaxial-pusher, a weighted pseudo-inverse method [91, 93] was used to account for the redundant control (similar to what was done for the inner loop). The control allocation matrix $\mathbf{M}$ is given by:

$$\mathbf{M} = \mathbf{W}^{-1} \mathbf{B}^T (\mathbf{B} \mathbf{W}^{-1} \mathbf{B}^T)^{-1} \quad (5.7)$$

where $\mathbf{B}$ is the control derivative matrix from Eq. 5.5 and the weighting matrix $\mathbf{W} = \text{diag}\{1 \ 20 \ 1\}$ was chosen to fade out collective input at around the minimum power speed $V_{\text{CAS}} \approx 100$ kts (Figure 2.4). The $\mathbf{B}$ matrix from Eq. 5.5 is then multiplied by the control allocation matrix $\mathbf{M}$ to get a square matrix that can be inverted for the $(\mathbf{CB})^{-1}$ block.

Figure 5.11 shows the outer-loop control allocation matrix $\mathbf{M}$ as a function of airspeed V for the coaxial-pusher. As expected, in hover, the outer-loop airspeed command is allocated primarily to pitch attitude θ (similar to a typical helicopter) and some to the pusher propeller δ_{prop} . As airspeed increases, the outer-loop airspeed command is allocated less to pitch attitude and more to pusher propeller (similar to a typical fixed-wing aircraft). The climb rate command is allocated to symmetric main rotor collective δ_{col} in hover, and pitch attitude as airspeed increases. Again, this behavior is similar to a helicopter at hover/low-speed and converts to airplane-like behavior at high speed.

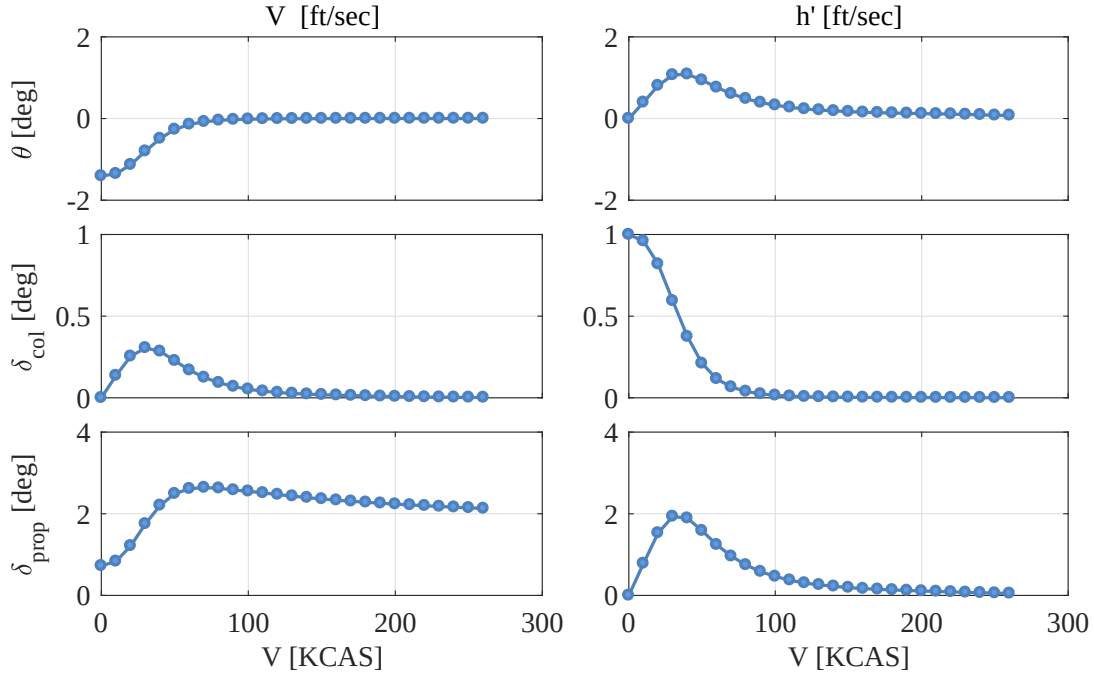


Figure 5.11. Outer-loop control allocation (coaxial-pusher).

5.1.2.2.2 Tiltrotor

In the case of the tiltrotor, the outer-loop airspeed and climb rate commands were allocated to pitch attitude and symmetric main rotor collective commands only. The nacelles were scheduled with commanded acceleration, instead of being used as a dynamic input for the outer loops to control. This was done to avoid excessive

hub moments and overdriving the nacelle actuators. Since the tiltrotor lower-order dynamic inversion system is square, the $\mathbf{B}$ matrix can be inverted, and the $(\mathbf{CB})^{-1}$ block acts as the control allocation. Figure 5.12 shows the tiltrotor outer-loop control allocation as a function of airspeed and nacelle angle. The control allocation is similar to that for the coaxial-pusher with airspeed command primarily being allocated to pitch attitude at hover/low-speed and main rotors collective at high speed. Climb rate command is primarily allocated to collective at hover/low-speed and pitch attitude at high speed.

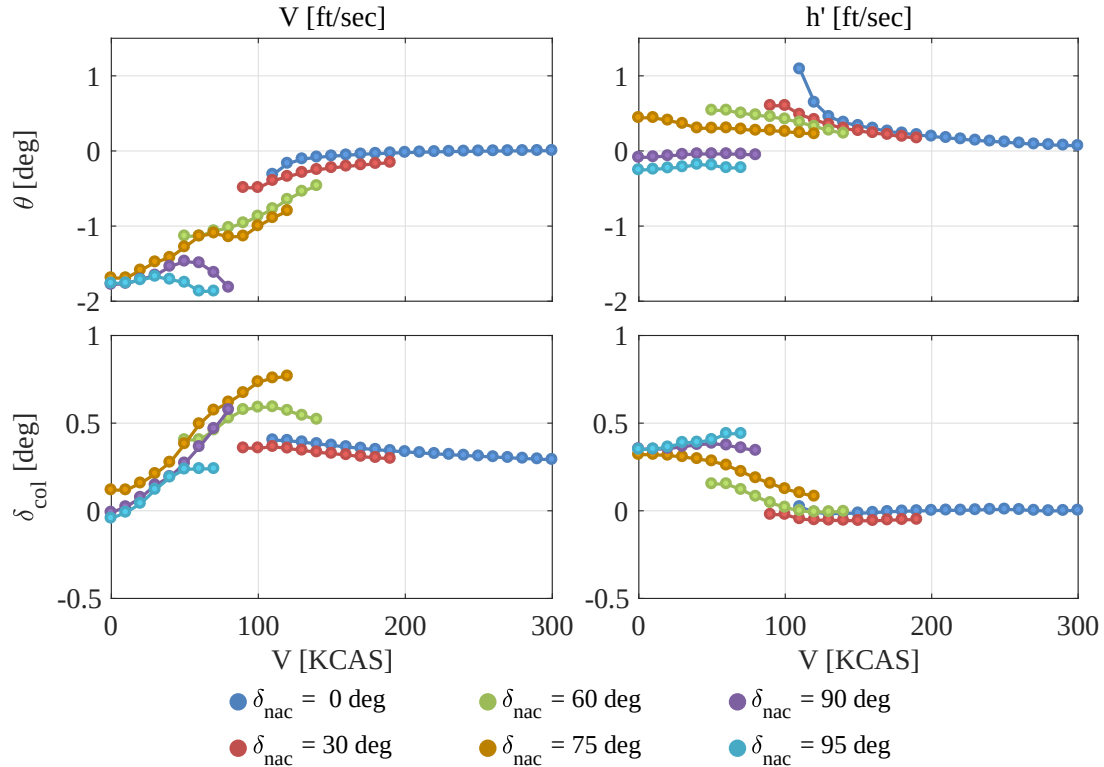


Figure 5.12. Outer-loop control allocation (tiltrotor).

Figure 5.13 shows the outer-loop nacelle command schedule. The figure shows the conversion corridor of the tiltrotor as well as the path through the corridor (combinations of airspeed and nacelle angle) that results in a pitch trim attitude of $\theta_0 = 0$ deg (dashed blue line in Figure 5.13). Three nacelle schedules are used based on the sign (positive, negative, or zero) of the acceleration command $\dot{V}_{cm}$ from the airspeed loop command model. When no acceleration is commanded ($\dot{V}_{cm} = 0$, red line in Figure 5.13), the nacelle schedule follows the $\theta_0 = 0$ deg schedule from hover to $V_{CAS} = 140$ kts. At speeds between $V_{CAS} = 140 - 180$ kts, the nacelle schedule linearly approaches $\delta_{nac} = 0$ deg. Above $V_{CAS} = 180$ kts the nacelle schedule is $\delta_{nac} = 0$ deg.

During acceleration ($\dot{V}_{cm} > 0$, green line in Figure 5.13), the nacelle schedule is closer to the bottom edge of the conversion corridor. This tracks the path through the

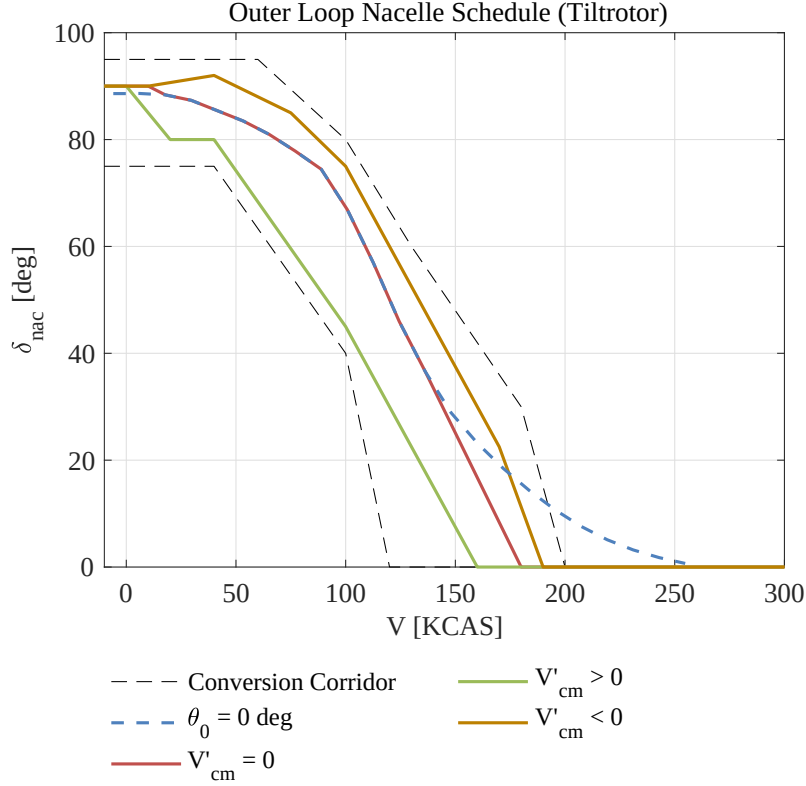


Figure 5.13. Outer-loop nacelle schedule (tiltrotor).

corridor for fastest acceleration. In addition, this path has a trim pitch attitude that is $\theta_0 > 0$ deg. During acceleration, the outer loops command a negative pitch attitude as seen by the negative values in the first subplot in Figure 5.12. This ensures that the total pitch attitude which is the sum of trim and outer loop command will not be excessively nose-down.

Conversely, during deceleration ($\dot{V}_{cm} < 0$, orange line in Figure 5.13), the nacelle schedule is closer to the upper edge of the conversion corridor. As with the acceleration nacelle schedule, this ensures that the total pitch attitude will not be excessively nose-up.

Below $V_{CAS} = 10$ kts, all three nacelle schedules command $\delta_{nac} = 90$ deg.

5.1.2.3 Comparison of Methods

As discussed in Sec. 5.1.2.1, the measure of goodness of the dynamic inversion system used to determine (CA) and $(CB)^{-1}$ is how well it makes the relationship between the output variables and the inputs appear as a set of decoupled integrators. Figures 5.14 and 5.15 show the airspeed and climb rate response to their outer-loop feedback signals for the coaxial-pusher at $V_{CAS} = 180$ kts. With ideal dynamic inversion, these

responses should be $1/s$. Additional phase roll-off can be seen above $\omega = 1$ rad/sec for both responses due to the actuators, sensors, and higher-order state not account for in the dynamic inversion. Overall, the CIPHER ID method produces responses which most closely resemble $1/s$. The responses from the Analytical method have a low frequency mode that is not eliminated, likely due to dropping of the cross coupling stability derivatives in the $(\mathbf{A})$ matrix. However, this mode would likely be suppressed by the integral gain in the feedback path.

Figures 5.16 and 5.17 show the airspeed and climb rate response to the opposite outer-loop feedback signals for the coaxial-pusher at $V_{CAS} = 180$ kts. These are the off-diagonal frequency responses that should be 0 in the case of perfect dynamic inversion. Here again, the CIPHER ID method produces responses with the lowest overall magnitude.

Therefore, the CIPHER ID method was used to generate the stability derivative $(\mathbf{A})$ and control derivative $(\mathbf{B})$ matrices.

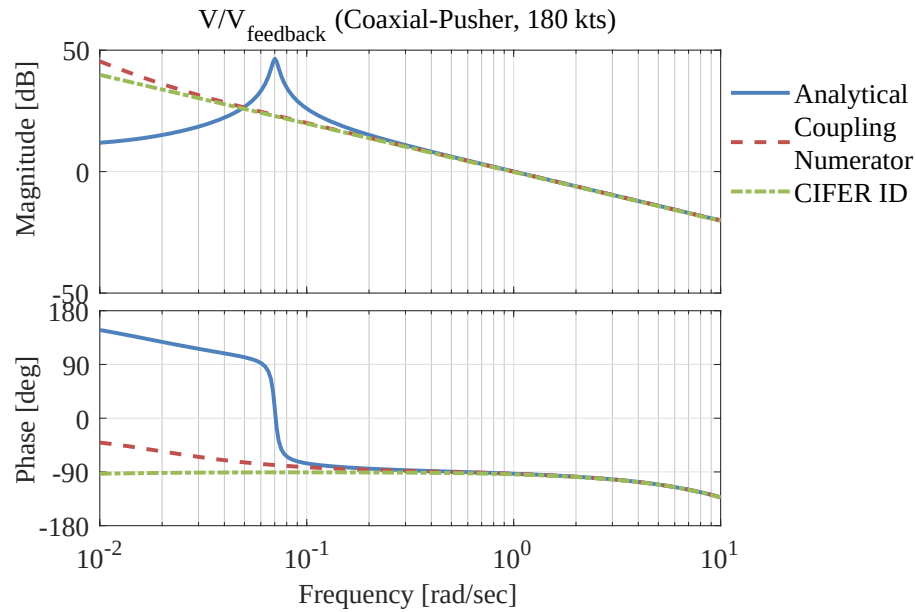


Figure 5.14. Dynamics inversion airspeed response to airspeed feedback (coaxial-pusher, 180 kts).

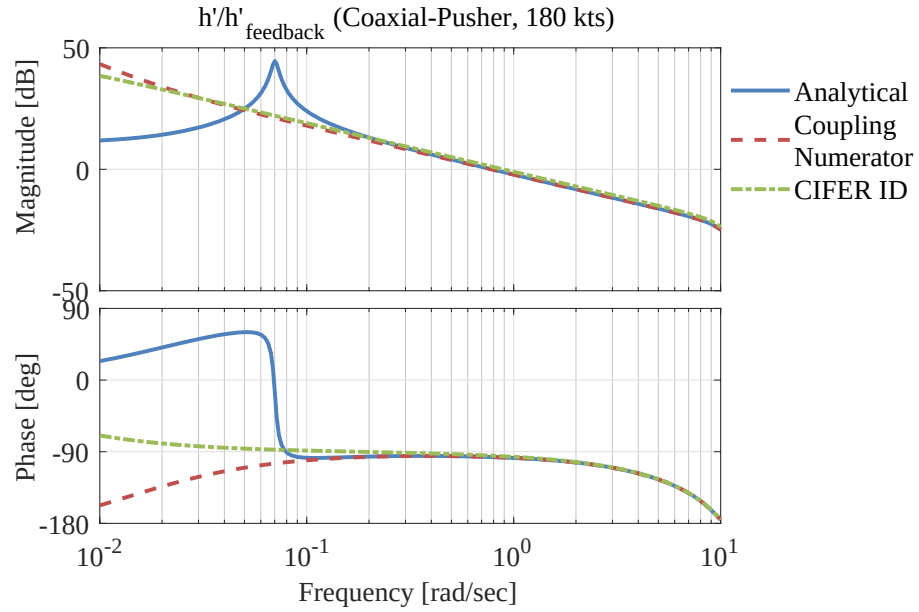


Figure 5.15. Dynamics inversion climb rate response to climb rate feedback (coaxial-pusher, 180 kts).

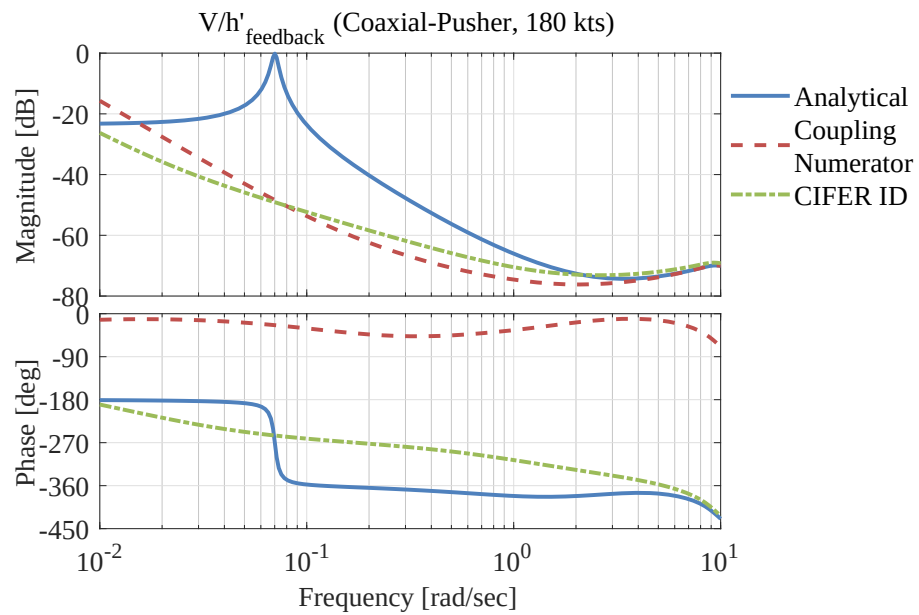


Figure 5.16. Dynamics inversion airspeed response to climb rate feedback (coaxial-pusher, 180 kts).

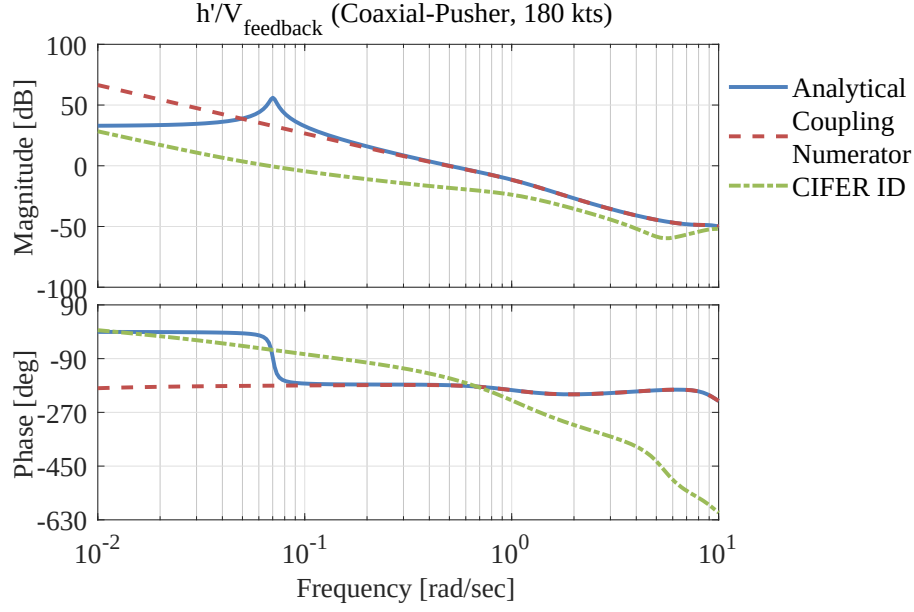


Figure 5.17. Dynamics inversion climb rate response to airspeed feedback (coaxial-pusher, 180 kts).

5.1.3 Command Model

The command model in each axis sets the aircraft response characteristics to pilot inputs (i.e., response type, magnitude, and bandwidth). Table 5.2 lists the response type in each axis as a function of airspeed. The collective stick δ_{col_s} generates an acceleration command response type using a first-order command model:

$$\frac{\dot{V}_{cm}}{\delta_{col_s}} = \frac{K_{col_{cm}}}{\tau_{col_{cm}} s + 1} \quad (5.8)$$

The command model gain was tuned based on the maximum acceleration and deceleration values at each airspeed. Maximum commanded acceleration $\dot{V}_{cm_{max+}}$ values are based on excess power P_{ex} :

$$K_{col_{cm}} = \dot{V}_{cm_{max+}} = \frac{P_{ex}}{(W/g)V} \quad (5.9)$$

In this case, the calculation of excess power as a function of airspeed was done offline, however, the control laws can be modified to dynamically change the command model gain $K_{col_{cm}}$ as a function of the actual excess power, as done in Ref. 110.

Maximum commanded deceleration $\dot{V}_{cm_{max-}}$ are based on other aircraft limitation and were determined off-line as well.

Figures 5.18 and 5.19 show the airspeed command model parameters for the

Table 5.2. Outer-Loop Control System Response Types

Speed Range [kts]	Longitudinal Cyclic	Thrust Control Lever
0–40	Vertical Rate Command/ Height Hold	Ground Speed Acceleration Command/ Speed Hold
40–300	Flight Path Rate Command/ Attitude Hold	Airspeed Acceleration Command/ Speed Hold

coaxial-pusher and tiltrotor, respectively. Note that for the stick gain $K_{\text{col}_{\text{cm}}}$ (top axis), two lines are shown: one for acceleration (positive values) and one for deceleration (negative values).

For the climb rate response, longitudinal stick commands climb rate for airspeeds between hover and $V_{\text{CAS}} = 40$ kts using a second-order command model:

$$\frac{\dot{h}_{\text{cm}}}{\delta_{\text{lon}_s}} = \frac{K_{\text{lon}_{\text{cm}}} \omega_{\text{lon}_{\text{cm}}}^2}{s + 2\zeta_{\text{lon}_{\text{cm}}} \omega_{\text{lon}_{\text{cm}}} s + \omega_{\text{lon}_{\text{cm}}}^2} \quad (5.10)$$

Above $V = 40$ kts, longitudinal stick commands flight path rate using a first order command model:

$$\frac{\dot{\gamma}_{\text{cm}}}{\delta_{\text{lon}_s}} = \frac{K_{\text{lon}_{\text{cm}}}}{\tau_{\text{lon}_{\text{cm}}} s + 1} \quad (5.11)$$

Flight path rate and attitude are commanded and held, respectively, above $V = 40$ kts to aid in decelerating descending approaches. In these instances, if climb rate was held, the pilot would have to continuously adjust the commanded climb rate as airspeed varied to maintain the flight path vector.

Figures 5.20 and 5.21 show the flight path command model parameters for the coaxial-pusher and tiltrotor, respectively. Here, the stick gain $K_{\text{lon}_{\text{cm}}}$ was set to match the inner-loop $K_{\text{lon}_{\text{cm}}}$. The inner-loop $K_{\text{lon}_{\text{cm}}}$ commands 30 deg/sec of pitch rate between hover and $V_{\text{CAS}} = 110$ kts. Above $V_{\text{CAS}} = 110$ kts, the maximum pitch rate command is set to equal a normal acceleration of $n_z = 2.5$ g.

The flight path command model time constant, $\tau_{lon_{cm}}$ was chosen to have a flight path bandwidth of $\omega_{BW_\gamma} = 1.3$ rad/sec throughout the flight envelope.

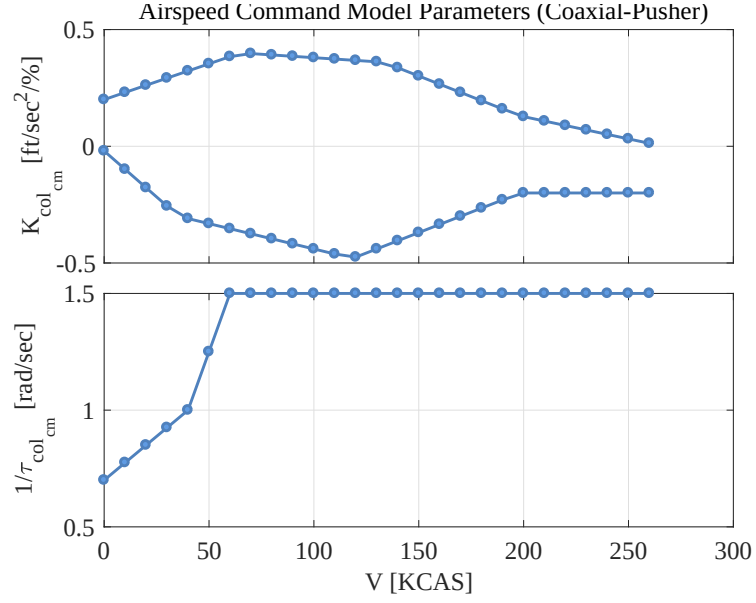


Figure 5.18. Airspeed command model parameters (coaxial-pusher).

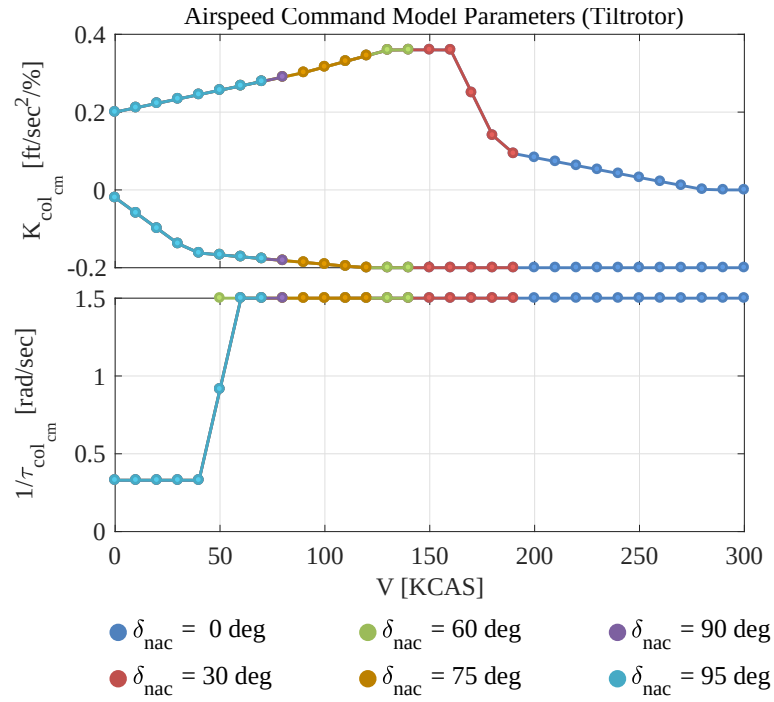


Figure 5.19. Airspeed command model parameters (tiltrotor).

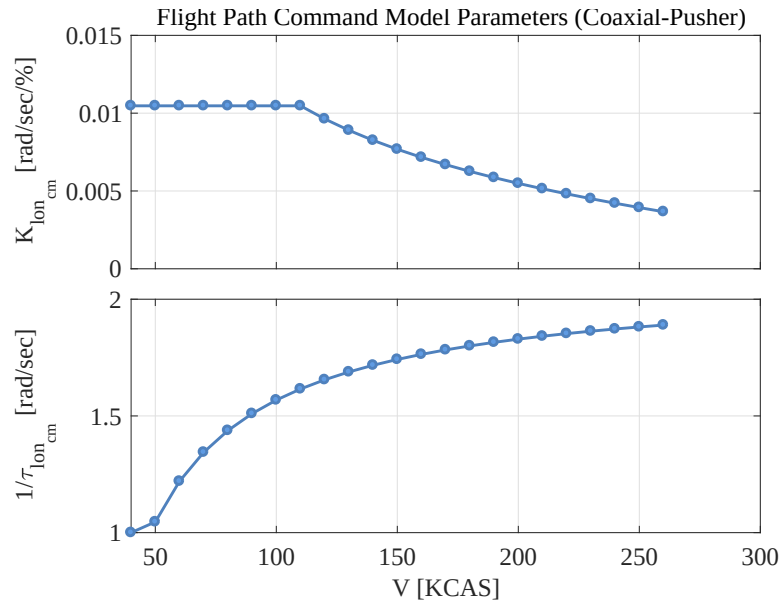


Figure 5.20. Flight path command model parameters (coaxial-pusher).

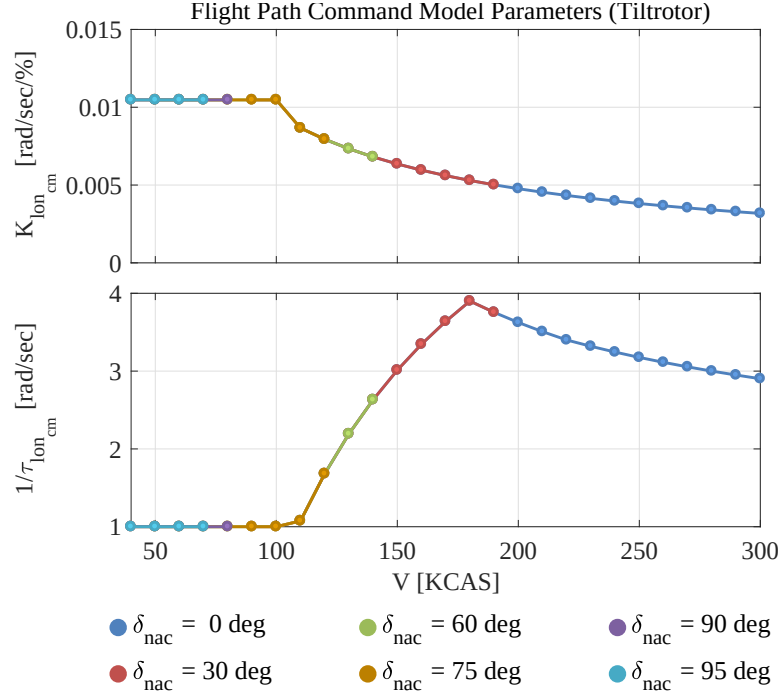


Figure 5.21. Flight path command model parameters (tiltrotor).

5.1.4 Command Delays

As with the inner loop, command delays are used to synchronize the commanded and actual states in time, before determining the error used for feedback. The addition of command delays is typical in model following control laws [50], and is done to account for higher-order dynamics and delays from the actuators, sensors, filters, and flight control computer processing time that are not accounted for by the inverse model. Accounting for this additional delay before comparing the commanded states with the actual states is useful to not overdrive the actuators. The addition of the command delays also reduces the amount of overshoot in the closed-loop end-to-end response with no added phase loss [99].

5.1.5 Feedback

Proportional-plus-integral (PI) feedback is used in both channels of the outer loops, with the integral gain K_I constrained as a ratio of the proportional gain K_P . The ratio K_I/K_P sets the zero in the feedback, and is set to be:

$$\frac{K_I}{K_P} = \frac{\omega_c}{5} \quad (5.12)$$

where ω_c is the crossover frequency of the loop broken at the “Outer-Loop Stability

Margin Break Point” in Figure 5.1. This ensures that the integral gain is effective, without overly degrading phase margin [50].

The feedback gains were optimized to meet a comprehensive set of stability, handling qualities, and performance specifications using a multi-objective optimization approach in CONDUIT[®] [50]. The Specifications section below lists the specifications used for the optimization.

5.2 Specifications

A common set of stability, handling qualities, and performance specifications, shown in Table 5.3, was used to optimize both coaxial-pusher and tiltrotor outer-loop control laws using the CONDUIT[®] software tool [50]. The specifications were divided into two categories—Tier 1 and Tier 2 specifications [50]. Tier 1 specifications are key flight control and handling qualities requirements that drive the design optimization, and are guaranteed to be met for an optimized design. Tier 2 specifications are those which are evaluated only at the end of the optimization. These are typically alternate requirements that give insight into the design and generally overlap with Tier 1 specifications. Because they are not evaluated during the optimization, due to computational time consideration, they are not always met.

5.2.1 Tier 1 Specifications

Twenty-three Tier 1 specifications were used for the outer-loop control laws divided into three categories which differentiate how they are handled during the control system optimizing [50]. The first category are the Hard Constraints, composed of the stability and stability margin requirements. During the first phase of the optimization, CONDUIT[®] ensures that these specifications are met. The Hard Constraints are met first to ensure that a stable system is used to evaluate all subsequent specifications. Once all of the Hard Constraints are met, the CONDUIT[®] optimization ensures that the Soft Constraints are met, while not violating the already met Hard Constraints. The Soft Constraints are composed of the handling-qualities requirements. Finally, once all Hard and Soft Constraints are met, CONDUIT[®] works to minimize the Summed Objective requirements while not violating the Hard and Soft Constraints. The Summed Objectives are composed of actuator RMS and crossover frequency ω_c . Minimizing the Summed Objectives results in a *Pareto optimum* solution. This is the solution that has minimum over-design, and therefore makes the most economical use of the actuators and minimizes sensitivity to noise.

For the coaxial-pusher and tiltrotor outer-loop control laws, Tier 1 specifications were selected primarily from SAE AS94900 [97] (stability margins) and ADS-33E [98] (handling qualities requirements).

Tier 1 specifications include absolute eigenvalue stability (CONDUIT[®] name EigLcG1), stability margins (StbMgG1), and Nichols margins (NicMgG1) requirements

(Hard Constraints). These specifications ensure that the design is stable with sufficient stability margins for each control loop broken at the input to the control allocation matrix.

Standard stability margin were evaluated for both airspeed and climb rate/flight path loops by breaking the loop at the outer-loop stability margin break point marked in Figure 5.1. In addition, stability margins of the inner pitch loop were evaluated with the outer loops closed by breaking the inner loop at the stability margin loop break point marked in Figure 3.1. Standard stability margin boundaries of gain margin $GM \geq 6$ dB and phase margin $PM \geq 45$ deg [97] are used throughout the flight envelope for the outer loops. In the case of the inner loop the standard stability margin boundaries were used for the coaxial-pusher throughout the flight envelope. For the tiltrotor, the inner-loop phase margin boundary was relaxed to $PM \geq 35$ deg for the mid-speed ($V_{CAS} = 50 - 100$ kts) speed range, since the standard phase margin requirement could not be met.

Soft Constraints include the model following cost specification (ModFoG2) which compares the closed-loop frequency response in each axis with the frequency response of the command model. A cost function J_{MF} is computed based on the weighted difference in the magnitude and phase of the responses, and a value of $J_{MF} \leq 50$ is enforced, ensuring good command model following in each axis [50].

Disturbance rejection bandwidth (DRB, DstBwG1) and peak (DRP, DstPkG1) specifications [102] were enforced in each axis for the appropriate hold variable using boundaries of $DRB \geq 0.75$ rad/sec and $DRP \leq 6$ dB.

A minimum crossover frequency specification (CrsMnG2) was included for each axis. This specification ensures that the frequency response for each control loop broken at the input to the control allocation matrix has a crossover frequency above a specified value. The minimum crossover requirements are $\omega_c = 1.0$ rad/sec for both airspeed and climb rate/flight path channels. As shown in Eq. 5.12, the integral gains were constrained to their respective proportional gains using the minimum crossover frequency specification value $\omega_c = 1.0$ rad/sec.

The Eigenvalue damping specification (EigDpG1) was used which evaluates the damping ratio of all closed-loop eigenvalues within a specified frequency range and compares them to the minimum required value.

The Open Loop Onset Point (OLOP, OlpOpG1) specification [104] is included to evaluate the control law's susceptibility to Pilot Induced Oscillations (PIOs) and limit cycle oscillations that can result from actuator rate limiting. Linear analysis methods ignore the nonlinear effects of actuator position and rate limiting. However, the OLOP specification is based on frequency domain describing function concepts, and is useful to include in the design process to not push the design beyond the actuator limits of the aircraft.

5.2.2 Tier 2 Specifications

Six Tier 2 specifications were used as “check only” and not enforced by the control system optimization. The ADS-33E pitch attitude piloted bandwidth specification (BnwPiF1) and flight path response to pitch attitude specification (FlpPiF1) were included for high-speed flight ($V_{CAS} > 100$ kts). The pitch attitude bandwidth specification is included in the Tier 2 because the response type for the outer loops is flight path rate. Therefore, the pilot does not control pitch attitude directly. However, it is still desirable to have Level 1 values of pitch attitude bandwidth in case the pilot does decide to close the loop around pitch attitude while in this mode. In addition, since pitch attitude is an actuator to control climb rate/flight path for the outer loops, it is desirable to have a sufficiently fast pitch response.

The flight path response to pitch attitude specification ensures that the flight path or vertical rate response does not lag the pitch attitude response by more than $\varphi = 45$ deg at all frequencies below $\omega = 0.4$ rad/sec. The relationship between flight path and pitch attitude is given approximately by [95]:

$$\frac{\gamma}{\theta} = \frac{1}{T_{\theta_2}s + 1} \quad (5.13)$$

where $1/T_{\theta_2}$ is the high-frequency pitch rate transfer function numerator zero. This parameter is a function of the bare-airframe, and since it shows up as a zero in the aircraft dynamics, it cannot be affected by feedback (although it can be through control allocation). Since this requirement does not drive the optimization of the feedback gains it is included as a Tier 2 specification.

Tier 2 specifications from MIL-STD-1797B consist of the pitch attitude (BnwPiL4) and flight path (BnwFpL1) bandwidth, pitch attitude dropback (DrpPiL1), and Control Anticipation Parameter (CAP, CapPiL1) specification, which were described in Sec. 3.2.2.

5.3 Optimization Strategy

The control law parameters are gain scheduled as a function of airspeed for the coaxial-pusher. Parameters were determined at 10 kt increments from hover to $V_{CAS} = 260$ kts, for a total of 27 design points. For the tiltrotor, control law parameters are gain scheduled as a function of airspeed (from hover to $V_{CAS} = 300$ kts) and nacelle angle. Figure 3.3 shows the tiltrotor design points (71 total design points), which are also in increments of 10 kts and span the conversion corridor.

At each design point, the command model parameters (stick gains and break frequencies) were hand-tuned to meet the piloted bandwidth requirements (as discussed in Sec. 5.1.3). The feedback gains were optimized in CONDUIT[®] using a multi-objective optimization approach to meet all of the Hard, Soft, and Summed Objective

Table 5.3. Outer-Loop Control System Optimization Specifications

CONDUIT [®]		Description (Motivation)	Axis*	Speed Range [kts]	Source
Spec Name					
Tier 1	Hard Constraints (Stability Requirements)				
	EigLcG1	Eignevalues in L.H.P. (Stability)	All	All	Generic
	StbMgG1	Gain and Phase Margin broken at outer loop break point (Stability)	V, γ	All	AS94900
	StbMgG1	Gain and Phase Margin broken at inner loop break point (Stability)	P	All	AS94900
	NicMgG1	Nichols Margins broken at outer loop break point (Stability)	V, γ	All	Ref. 105
	Soft Constraints (Handling Qualities Requirements)				
	ModFoG2	Command model following cost (HQ)	V, γ	All	Generic
	DstBwG1	Disturbance rejection bandwidth ≥ 0.75 rad/sec (Loads, Ride Quality)	V, γ	All	Ref. 102
	DstPkG1	Disturbance rejection peak ≤ 6 dB (Loads, Ride Quality)	V, γ	All	Ref. 102
	CrsMnG2	Minimum $\omega_c \geq 1.0$ rad/sec (Robustness)	V, γ	All	Generic
	EigDpG1	Eigenvalue Damping $\zeta \geq 0.5$ (HQ, Loads)	All	All	Generic
	OlpOpG1	Open Loop Onset Point, pilot input (PIO)	V, γ	All	Ref. 104
		Open Loop Onset Point, disturbance input (PIO)	V, γ	All	Ref. 104
	Summed Objective (Performance Requirements)				
	CrsLnG1	Crossover Frequency (Act. Activity)	V, γ	All	Generic
	RmsAcG1	Actuator RMS (Act. Activity)	V, γ	All	Generic
Tier 2	Check Only				
	BnwPiF1	Pitch attitude bandwidth and phase delay, forward flight (HQ)	P	60-300	ADS-33E
	FlpPiF1	Flight path response to pitch attitude (HQ)	γ	100-300	ADS-33E
	BnwPiL4	Bandwidth, phase delay (HQ)	P	60-300	MIL-STD-1797B
	BnwFpL1	Transient flight-path response (HQ)	P, γ	60-300	MIL-STD-1797B
	DrpPiL1	Pitch dropback (HQ)	P	60-300	MIL-STD-1797B
	CapPiL1	LOES Control Anticipation Parameters (HQ)	P	60-300	MIL-STD-1797B

* P = Pitch, V = Velocity loop, γ = flight path loop

constraints [50] listed in Table 5.3

5.4 Optimization Results

5.4.1 Coaxial-Pusher

5.4.1.1 Point Design

Figures 5.22 through 5.26 show several of the key frequency responses of the optimized outer-loop control system design for the coaxial-pusher at one flight condition ($V_{CAS} = 180$ kts). Figures 5.22 and 5.23 show the airspeed and flight path broken loop frequency responses, respectively. Both loops have crossover frequencies of $\omega_c = 1.0$ rad/sec, corresponding to the value enforced by the minimum crossover frequency specification. In addition, both loops have sufficient stability margins. Figure 5.24 shows the broken-loop response for the loop broken at the inner pitch loop with the outer loops closed. Although these stability margins are slightly degraded from their values without the outer loops closed, as expected, they are still in Level 1.

Figures 5.25 and 5.26 show the closed-loop airspeed and flight path responses to pilot stick input. In addition, the figures also show the commanded airspeed and flight path frequency responses and demonstrate the excellent model following capabilities of the control system. As discussed in Sec. 5.1.3, the flight path command model was tuned to give as flight path bandwidth $\omega_{BW_\gamma} = 1.3$ rad/sec, which can be seen in Figure 5.26.

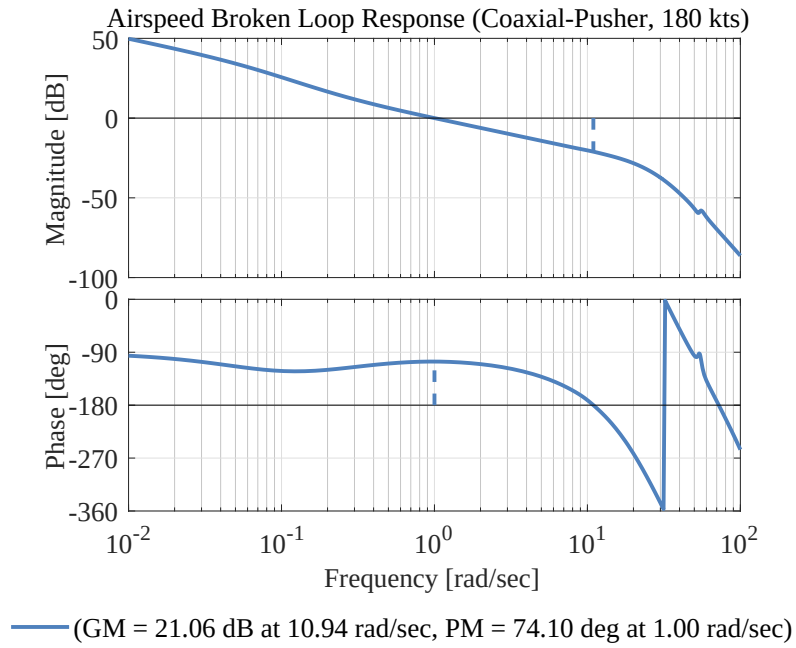


Figure 5.22. Airspeed outer-loop broken-loop responses (coaxial-pusher, 180 kts).

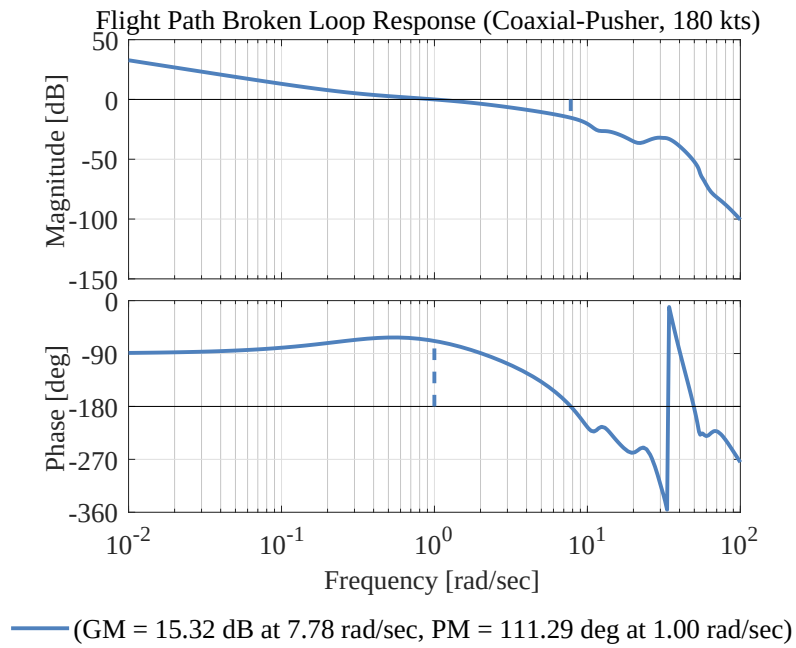


Figure 5.23. Flight path outer-loop broken-loop responses (coaxial-pusher, 180 kts).

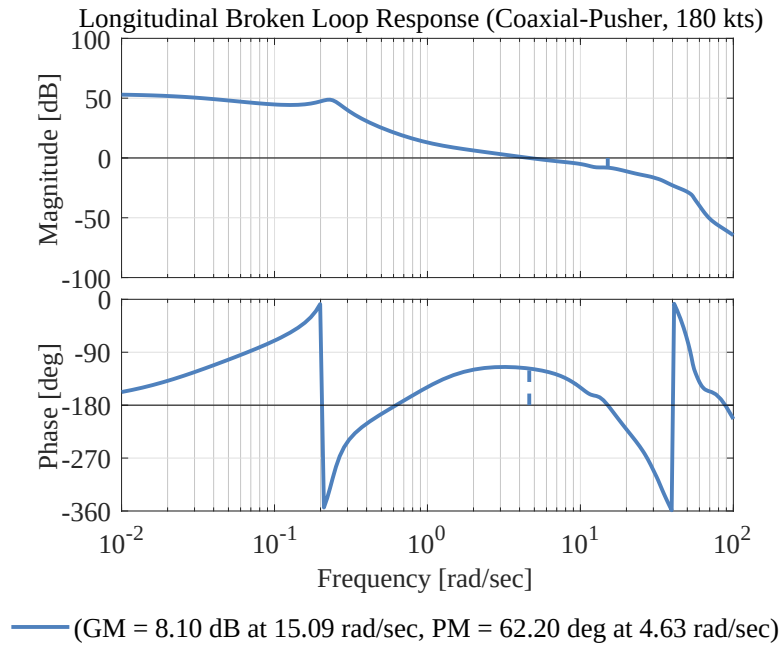


Figure 5.24. Longitudinal inner-loop broken-loop responses (coaxial-pusher, 180 kts).

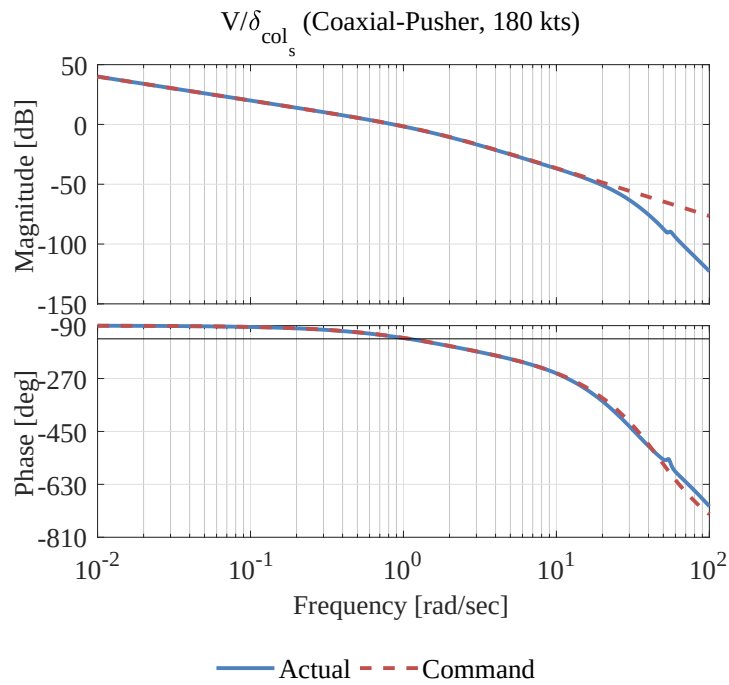


Figure 5.25. Closed-loop airspeed frequency responses (coaxial-pusher, 180 kts).

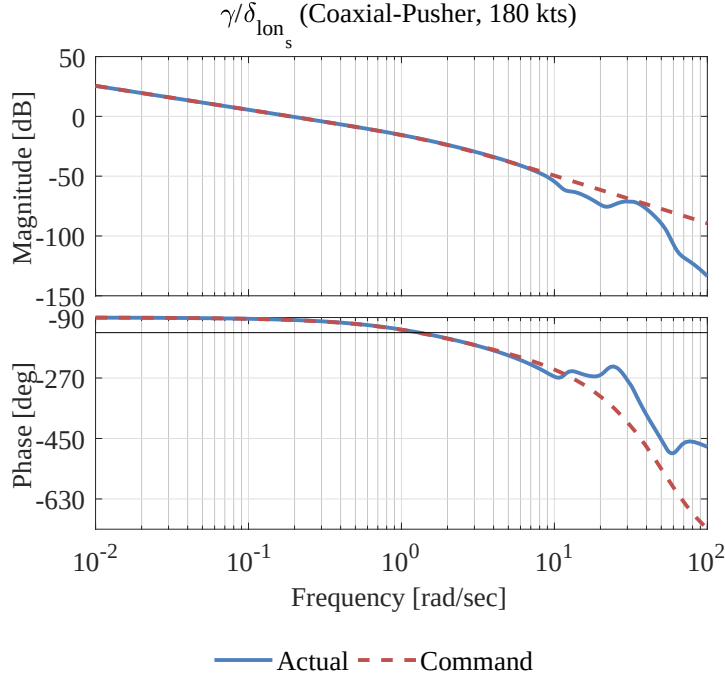


Figure 5.26. Closed-loop flight path frequency responses (coaxial-pusher, 180 kts).

Figure 5.27 shows time histories of flight path and airspeed step responses. The figure shows the commanded and actual flight path γ , airspeed V , pitch attitude θ , symmetric main rotor collective δ_{col} , and pusher propeller collective δ_{prop} . The control system provides excellent model following capabilities and decoupled airspeed and flight path responses. For example, the flight path step response is achieved by pitching the aircraft up (as seen in the pitch attitude θ response, Figure 5.27, first column, third subplot) and also increasing pusher propeller collective δ_{prop} to maintain airspeed (Figure 5.27, first column, last subplot).

Similar model following and decoupled response performance was seen throughout the entire flight envelope.

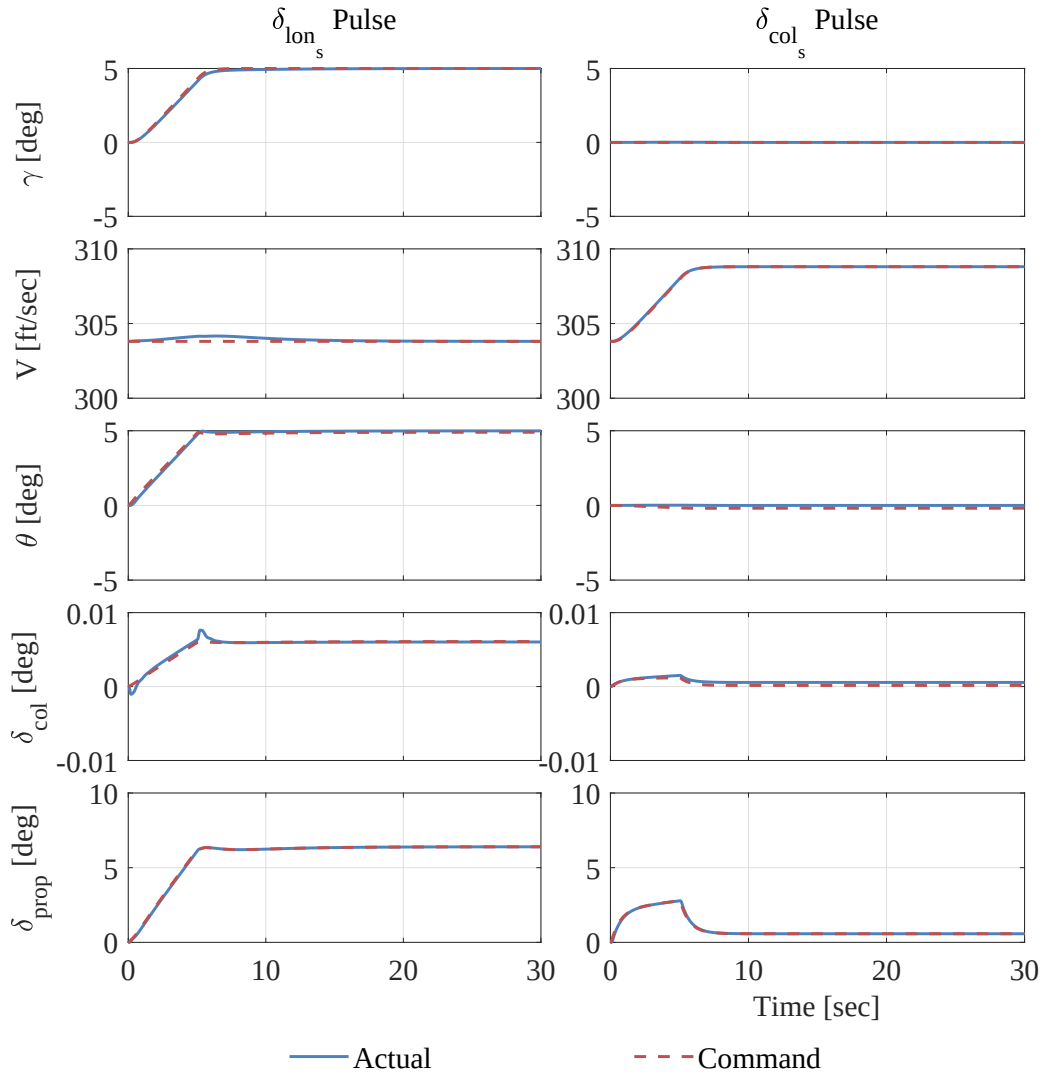


Figure 5.27. Outer-loop airspeed and flight path step responses (coaxial-pusher, 180 kts).

5.4.1.2 Gain Schedule

Figure 5.28 shows the coaxial-pusher outer-loop feedback gains as a function of airspeed. As expected, the gains vary less than the inner-loop feedback gains, since the bare-airframe with inner loop closed has less varying dynamics as a function of airspeed than the bare-airframe alone. In addition, the dynamic inversion loop further suppresses variations in the aircraft response as a function of flight condition. The gains are not constant though, even with the dynamic inversion architecture, due to enforcing a minimum crossover frequency for the full outer-loop feedback at the “Outer-Loop Stability Margin Break Point” in Figure 5.1 and not at the output of the

“Feedback” block.

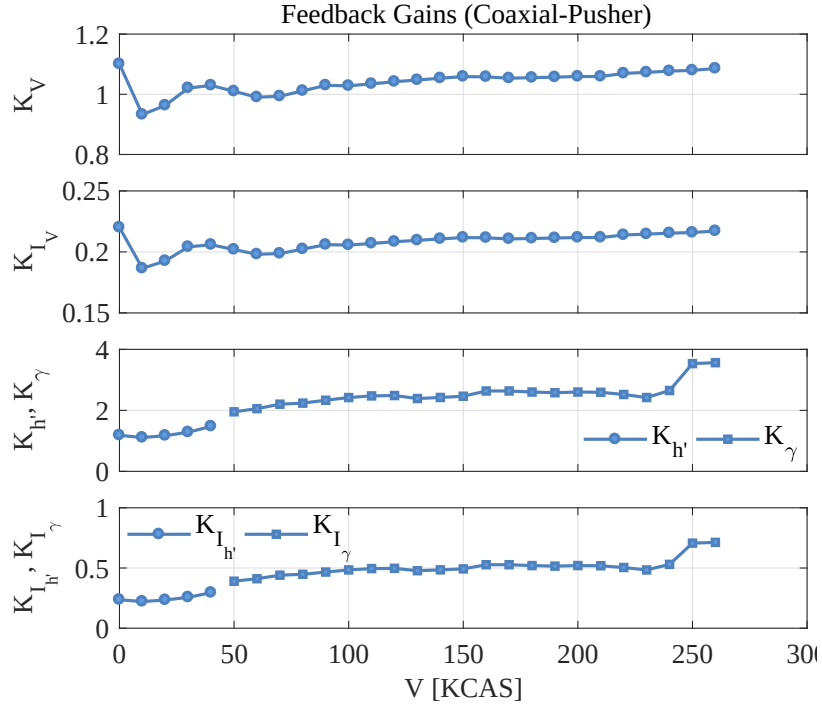


Figure 5.28. Outer-loop feedback gain schedule (coaxial-pusher).

Figure 5.29 shows several of the specifications and the optimized design values for several airspeeds ranging from hover to $V_{CAS} = 240$ kts. The first two rows of subplots show Tier 1 specifications: Stability margin, eigenvalue damping, minimum crossover frequency, and DRB/DRP. These are met for all of the designs through the optimization process.

The last row of subplots in Figure 5.29 shows several of the Tier 2 specifications. The first subplot is the ADS-33E flight path response to pitch attitude specification, which is met for all of the designs. Recall from Sec. 5.2.2 that this specification is equal to the bare-airframe $1/T_{\theta_2}$ (inverse flight path-attitude lag) and is not affected by the feedback.

The last three specifications in the third row of subplots in Figure 5.29 are the Tier 2 MIL-STD-1797B flight path versus pitch attitude bandwidth, CAP, and pitch attitude dropback specification. As described in Sec. 5.1.3, the flight path command model was tuned to give a flight path bandwidth $\omega_{BW_\gamma} = 1.3$ rad/sec, which corresponds to the flight path bandwidth values on the specification.

Since a constant flight path bandwidth ω_{BW_γ} value is enforced for all of the designs, the resulting pitch attitude bandwidth ω_{BW_θ} and dropback characteristics are a function of the bare-airframe $1/T_{\theta_2}$. Higher values of $1/T_{\theta_2}$ correspond to a shorter flight path to pitch attitude time lag. Therefore, for the same flight path bandwidth

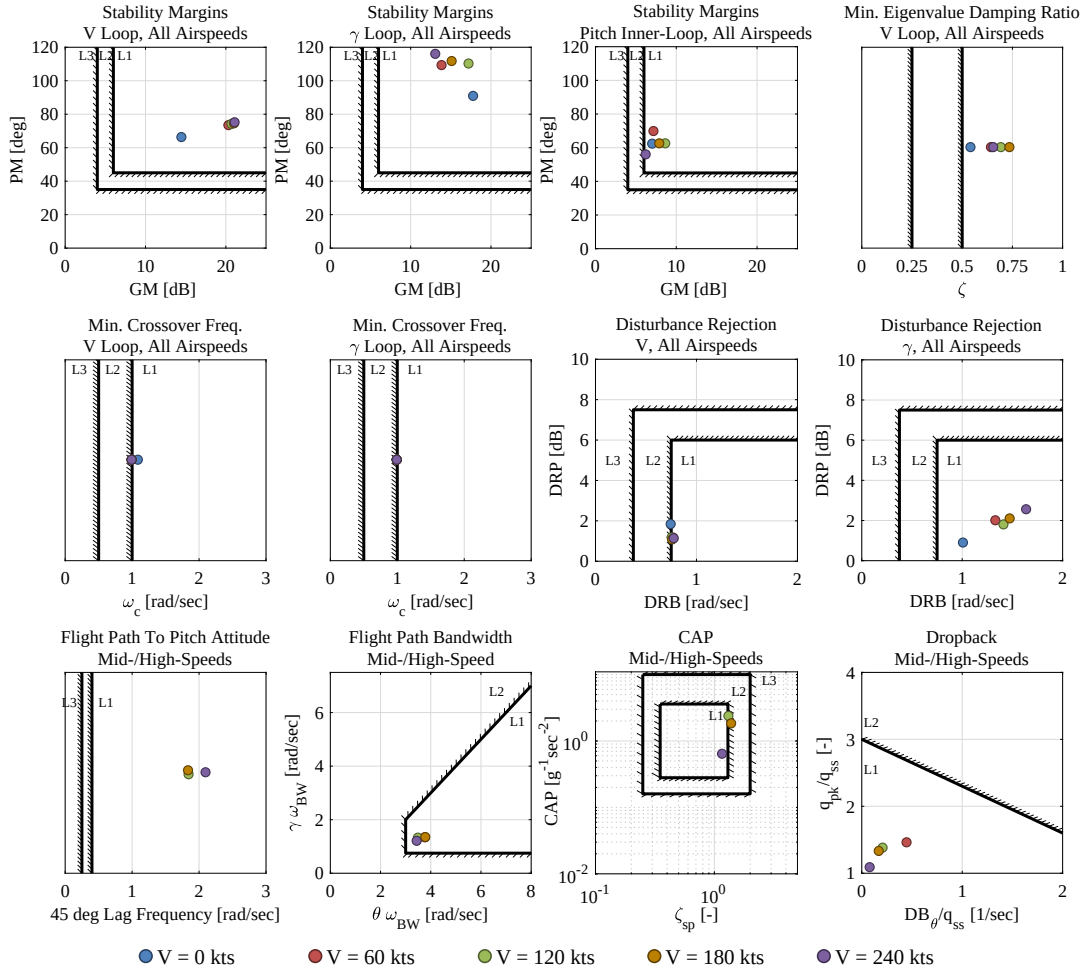


Figure 5.29. Example control system design specification and optimized design results (coaxial-pusher, hover-240 kts).

frequency, higher values of $1/T_{\theta_2}$ result in designs with lower required values of pitch attitude bandwidth and lower value of pitch attitude dropback (as seen by comparing the designs at $V_{CAS} = 180$ and 240 kts in Figure 5.29).

5.4.2 Tiltrotor

5.4.2.1 Point Design

Figures 5.30 through 5.34 show several of the key frequency responses of the optimized outer-loop control system design for the tiltrotor at one flight condition ($V_{CAS} = 180$ kts, $\delta_{nac} = 0$ deg). Figures 5.30 and 5.31 show the airspeed and flight path broken loop frequency responses, respectively. Both loops have crossover frequencies of $\omega_c = 1.0$

rad/sec, corresponding to the value enforced by the minimum crossover frequency specification. In addition, both loops have sufficient stability margins. Figure 5.32 shows the broken-loop response for the loop broken at the inner pitch loop with the outer loops closed. Although these stability margins are slightly degraded from their values without the outer loops closed, as expected, they are still in Level 1.

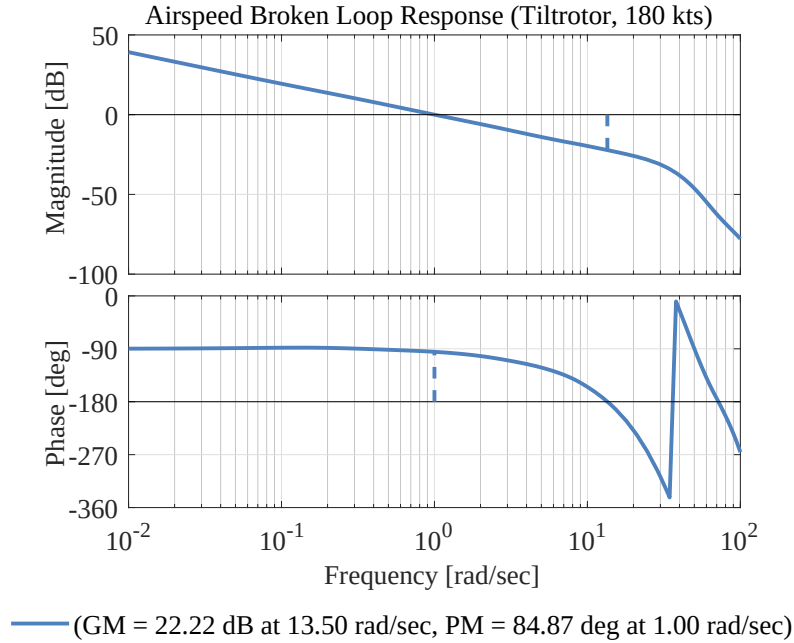


Figure 5.30. Airspeed outer-loop broken-loop responses (tiltrotor, 180 kts).

Figures 5.33 and 5.34 show the closed-loop airspeed and flight path responses to pilot stick input. In addition, the figures also show the commanded airspeed and flight path frequency responses and demonstrate the excellent model following capabilities of the control system. As discussed in Sec. 5.1.3, the flight path command model was tuned to give as flight path bandwidth $\omega_{BW_\gamma} = 1.3$ rad/sec, which can be seen in Figure 5.34.

Figure 5.35 shows time histories of a flight path and airspeed step responses. The figure shows the commanded and actual flight path γ , airspeed V , pitch attitude θ , and symmetric main rotors collective δ_{col} . The control system provides excellent model following capabilities and decoupled airspeed and flight path responses. Similar model following and decoupled response performance was seen throughout the entire flight envelope.

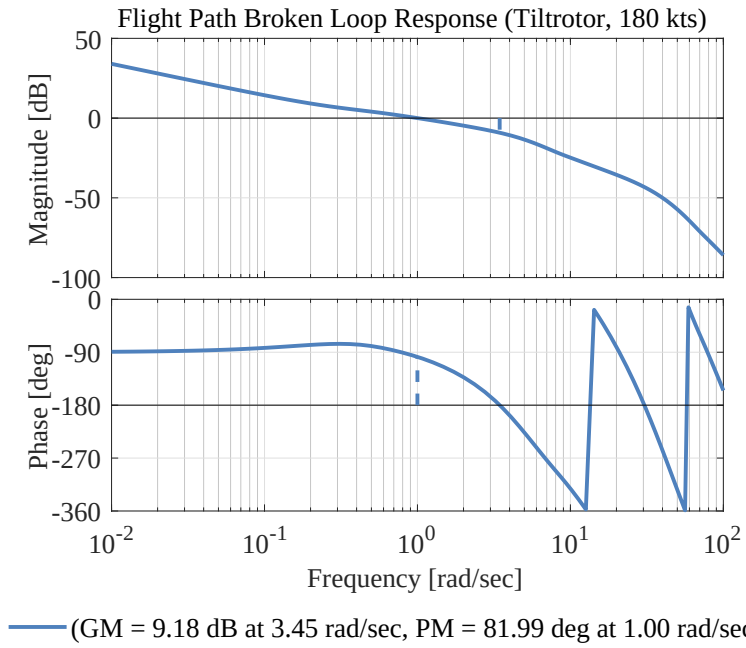


Figure 5.31. Flight path outer-loop broken-loop responses (tiltrotor, 180 kts).

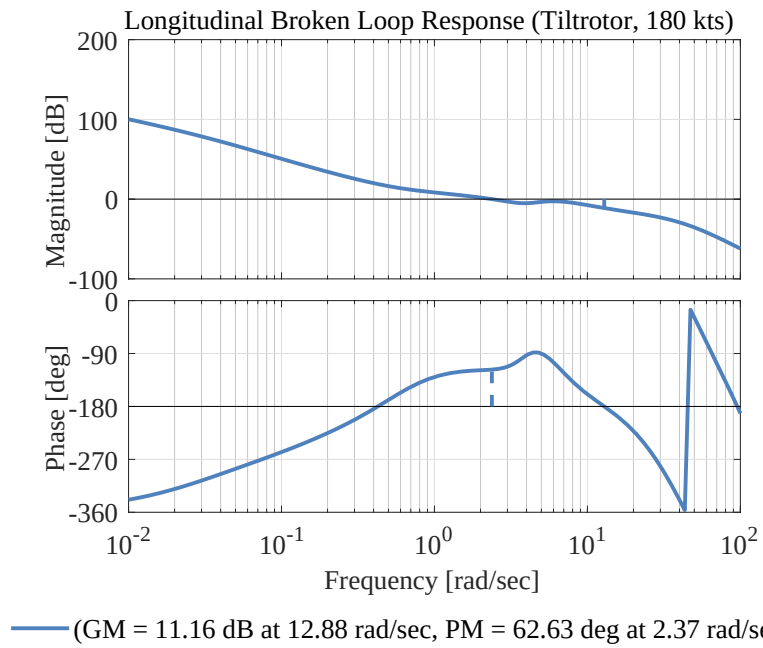


Figure 5.32. Longitudinal inner-loop broken-loop responses (tiltrotor, 180 kts).

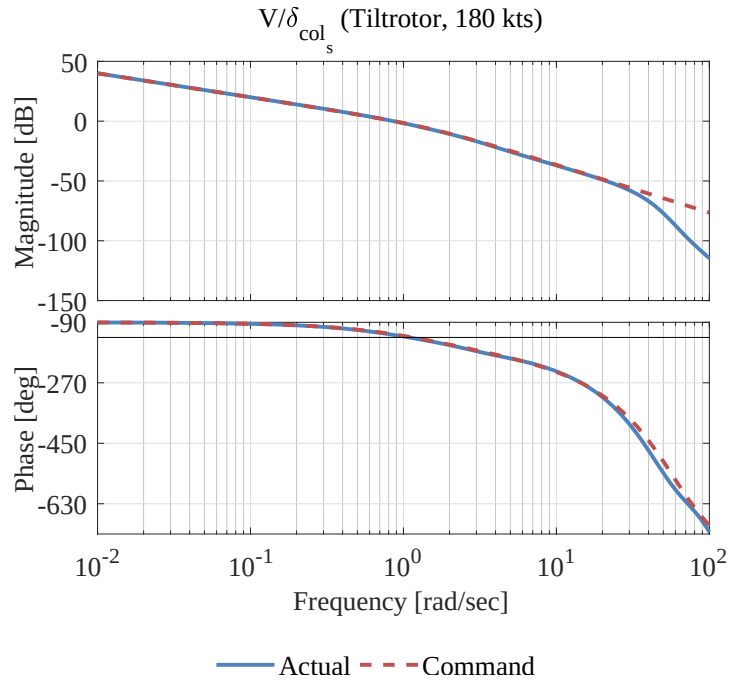


Figure 5.33. Closed-loop airspeed frequency responses (tiltrotor, 180 kts).

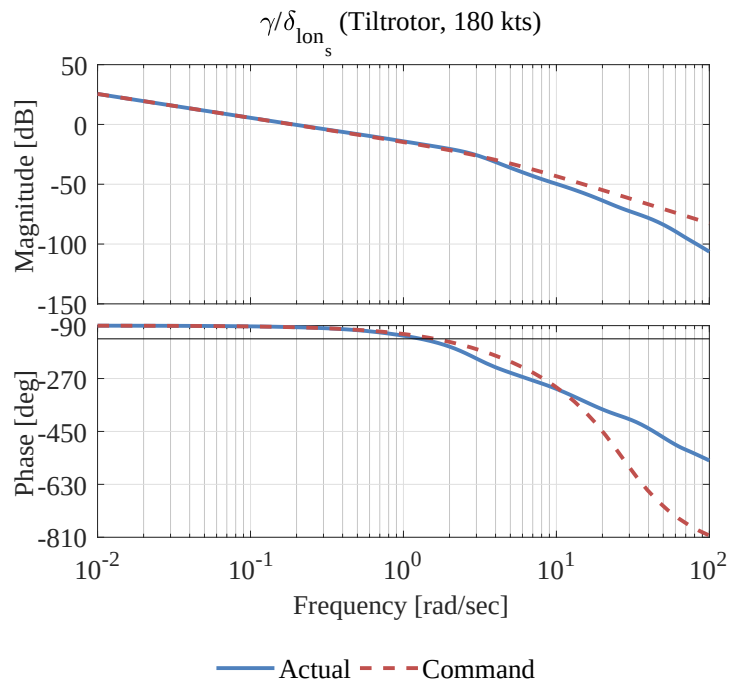


Figure 5.34. Closed-loop flight path frequency responses (tiltrotor, 180 kts).

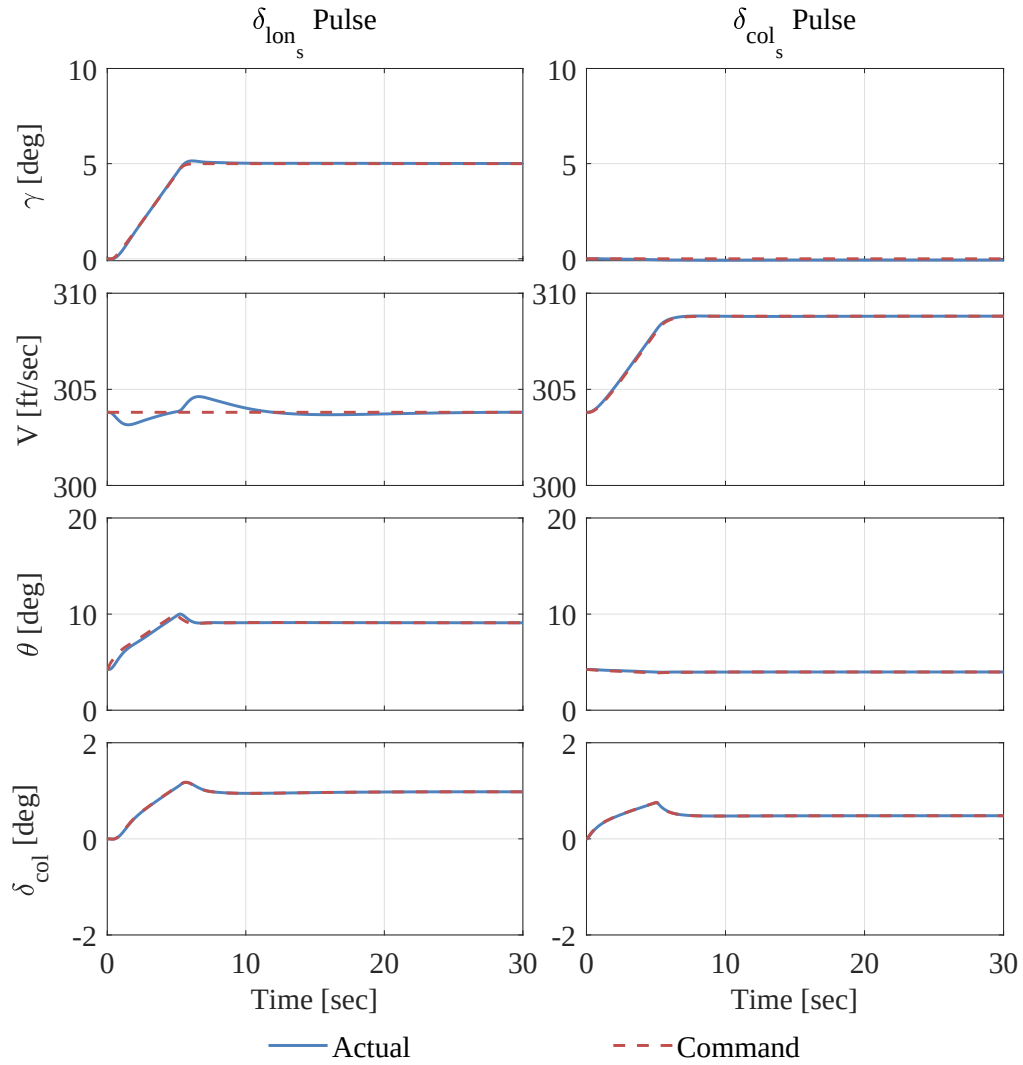


Figure 5.35. Outer-loop airspeed and flight path step responses (tiltrotor, 180 kts).

5.4.2.2 Gain Schedule

Figure 5.36 shows the tiltrotor outer-loop feedback gains as a function of airspeed and nacelle angle. As expected, the gains vary less than the inner-loop feedback gains, since the bare-airframe with inner loop closed has less varying dynamics as a function of airspeed than the bare-airframe alone. As with the coaxial-pusher, the gains are not constant, even with the dynamic inversion architecture due to enforcing a minimum crossover frequency for the full outer loop feedback at the “Outer-Loop Stability Margin Break Point” in Figure 5.1 and not at the output of the “Feedback” block.

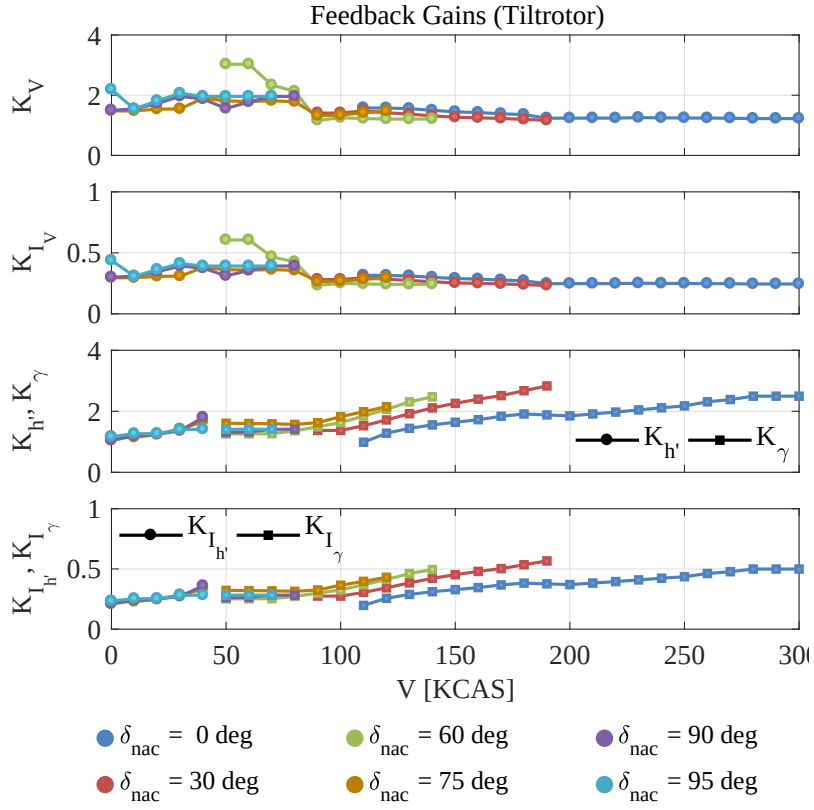


Figure 5.36. Outer-loop feedback gain schedule (tiltrotor).

Figure 5.37 shows several of the specifications and the optimized design values for several airspeeds ranging from hover to $V_{CAS} = 240$ kts. The first two rows of subplots show Tier 1 specifications: Stability margin, eigenvalue damping, minimum crossover frequency, and DRB/DRP. These are met for all of the designs though the optimization process.

The last row of subplots in Figure 5.37 shows several of the Tier 2 specifications. The first subplot is the ADS-33E flight path response to pitch attitude specification. Recall from Sec. 5.2.2 that this specification is equal to the bare-airframe $1/T_{\theta_2}$ (the inverse flight path-attitude lag) and is not affected by the feedback. The specification is met for all of the designs, except slow speed airplane mode ($V_{CAS} = 120$ kts, $\delta_{nac} = 0$ deg), which is at the edge of the conversion corridor (Figure 3.3).

The second subplots in the last row in Figure 5.37 is the MIL-STD-1797B flight path versus pitch attitude bandwidth specification. All of the designs have a flight path bandwidth close to the design value of $\omega_{BW_\gamma} = 1.3$ rad/sec (Sec. 5.1.3). The slower designs ($V_{CAS} = 60 - 120$ kts) have pitch attitude bandwidth in the Level 2 region of the specification.

The third subplots in the last row in Figure 5.37 is the MIL-STD-1797B Control Anticipation Parameter (CAP) specification. This requirement is met for all of the

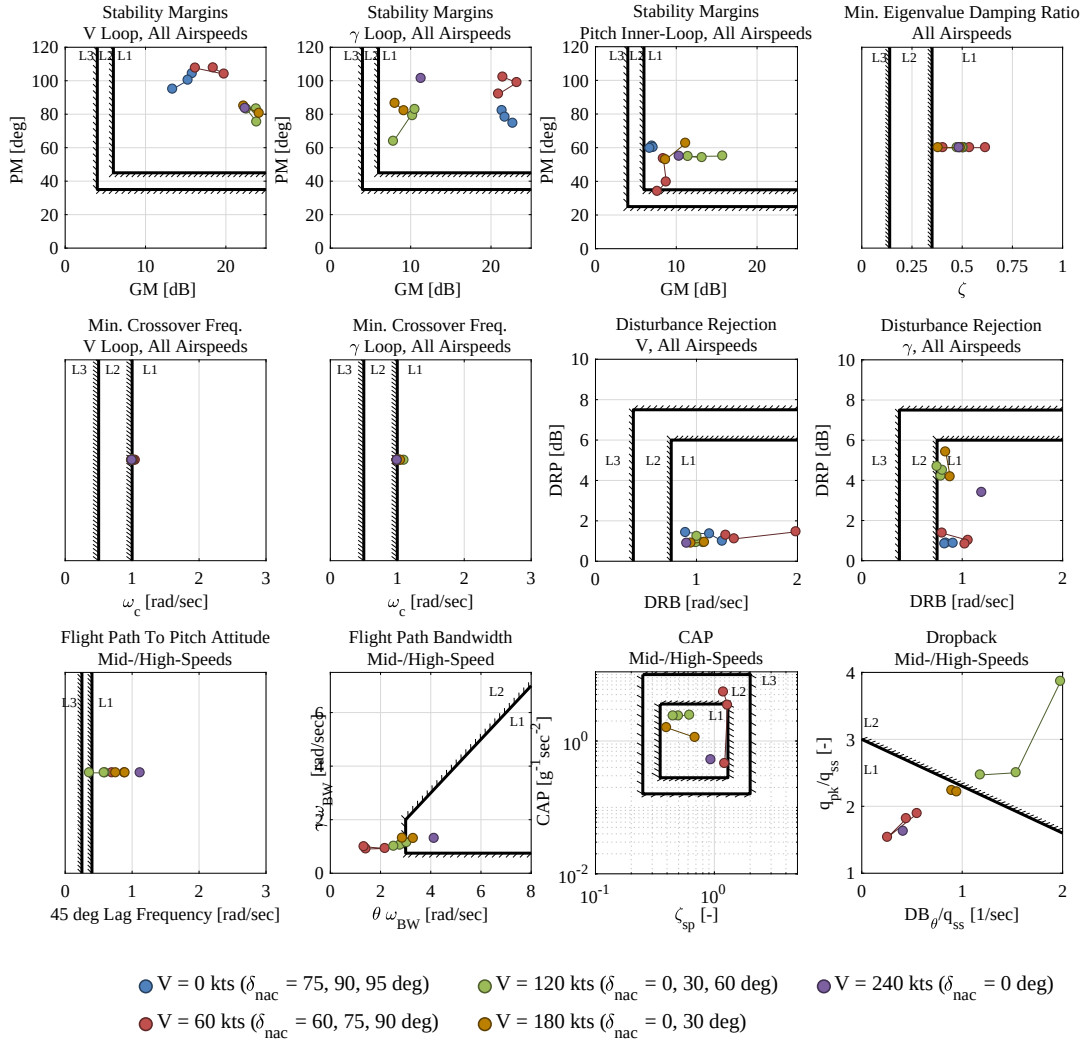


Figure 5.37. Example control system design specification and optimized design results (tiltrotor, hover–240 kts).

flight condition except for $V_{CAS} = 60$ kts, $\delta_{nac} = 60$ deg which is also on the edge of the conversion corridor.

Finally, the last subplots in the last row in Figure 5.37 is the MIL-STD-1797B pitch attitude dropback specification. Since a constant flight path bandwidth ω_{BW_γ} value is enforced for all of the designs, the resulting dropback characteristics are a function of the bare-airframe $1/T_{\theta_2}$, with lower values of $1/T_{\theta_2}$ resulting in higher values of dropback. This characteristic will be discussed in more detail in Sec. 5.6.

5.5 Additional Considerations

5.5.1 Pilot Inceptors

Handling qualities cannot be investigated without consideration to the pilot inceptors. However, a detailed study of what inceptor configuration pilots prefer to use with the outer-loop control laws is outside of the scope of this work. The outer-loop control laws described here were intended to be used with standard helicopter inceptors (sidestick, collective, and pedals). As such, a “Unified” control strategy [65,66] was used where the sidestick commands rate of climb $\dot{h}$ / flight path rate $\dot{\gamma}$ and the collective commands longitudinal acceleration $\dot{V}$ throughout the flight envelope, as described in more detail in Sec. 6.1.1.1. Although this control strategy works well for the F-35B and its mission [66], since it is geared for and more intuitive in the high-speed flight regime, it is not necessarily the best solution for a helicopter and its specific mission. Since the research here is focused on transition and high-speed flight, this control strategy was adopted, however future research should be focused on how to extend or blend this strategy with one more suited for the hover/low-speed flight regime.

5.5.2 Improved Inner-Loop Turn Compensation

To improve the outer-loop altitude hold capabilities during turns, the inner-loop turn compensation was modified for the pitch rate-command/attitude-hold portion of the inner-loop control laws. Previously, a pitch rate turn compensation command q_{TC} was added to the pitch rate command generated by the inner-loop command model based on [111]:

$$q_{TC} = \frac{g}{V} \tan \phi \sin \phi \quad (5.14)$$

Since $\dot{\theta} = 0$ in a coordinated turn, no command was summed with the pitch attitude command generated by the inner-loop command model, such that $\theta_{TC} = 0$.

To maintain altitude during a coordinated turn (i.e., $\gamma = 0$), the incremental normal acceleration Δn_z required is:

$$\Delta n_{z_{TC}} = \frac{1}{\cos \phi} - 1 \quad (5.15)$$

which in steady-state corresponds to an angle of attack α of:

$$\alpha_{TC} = \frac{\Delta n_{z_{TC}}}{n/\alpha} \quad (5.16)$$

where n/α is the steady-state normal acceleration per angle of attack, given by [95]:

$$\frac{n}{\alpha} = \frac{V}{gT_{\theta_2}} \quad (5.17)$$

Turn compensation can be improved by setting $\theta_{TC} = \alpha_{TC}$, especially when the inner-loop control system has an integral feedback gain on pitch attitude error. Figure 5.38 shows a time history plot for the tiltrotor at $V_{CAS} = 180$ kts performing a steady turn with a bank angle of $\phi = 45$ deg. Aircraft responses are shown for both pitch attitude turn compensation off ($\theta_{TC} = 0$) and on ($\theta_{TC} = \alpha_{TC}$) cases. With the addition of the pitch attitude turn compensation, there is significantly less altitude loss ($\Delta h = -32$ ft instead of $\Delta h = -290$ ft after 40 sec).

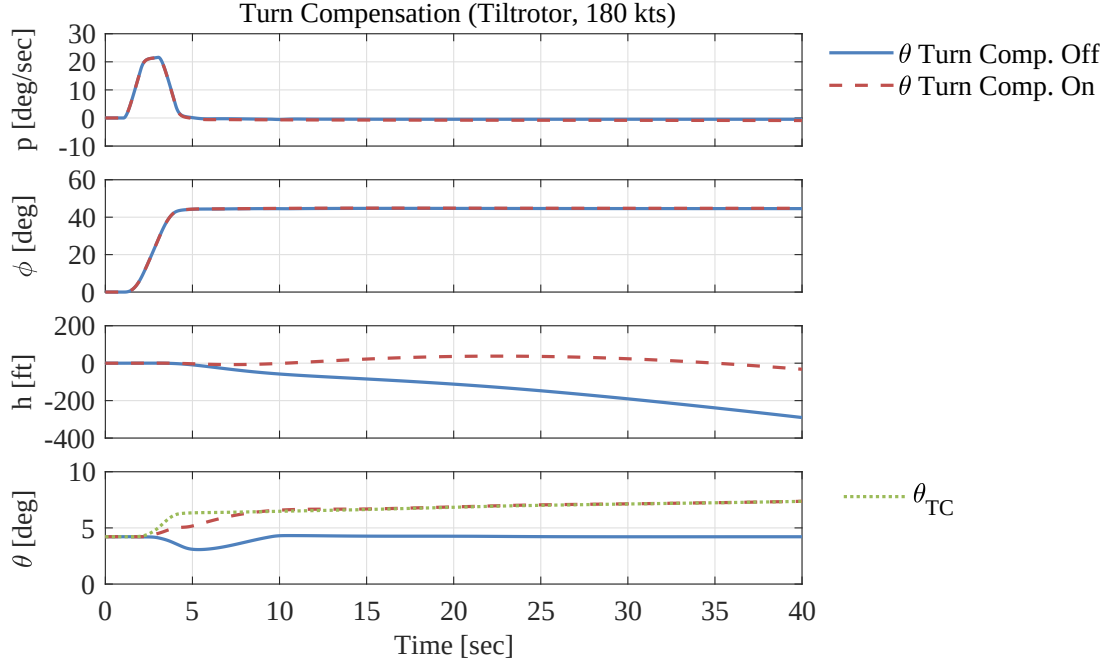


Figure 5.38. Inner-loop turn compensation performance (tiltrotor, 180 kts).

5.6 Discussion

As discussed earlier, the flight path-attitude lag parameter T_{θ_2} is the time in seconds that the flight path γ response of the aircraft lags behind the pitch attitude θ response. This parameter is a function of the bare-airframe and cannot be changed using feedback to one control input alone. Figures 5.39 and 5.40 show the values of flight path-attitude lag T_{θ_2} for the coaxial-pusher and tiltrotor, respectively, as a function of airspeed and nacelle angle (tiltrotor only). Overall, the coaxial-pusher has lower values of T_{θ_2} , especially in the mid-airspeed range ($V_{CAS} = 100 - 200$ kts) as compared to the tiltrotor in airplane mode ($\delta_{nac} = 0$ deg). This is consistent with the findings of Cameron and Padfield [57], who noted that large pitch rate overshoot is more severe in tiltrotors than conventional fixed- or rotary-wing aircraft due to proprotors making positive contributions to both the pitch damping derivative M_q and the static stability

derivatives M_w , resulting in large values T_{θ_2} .

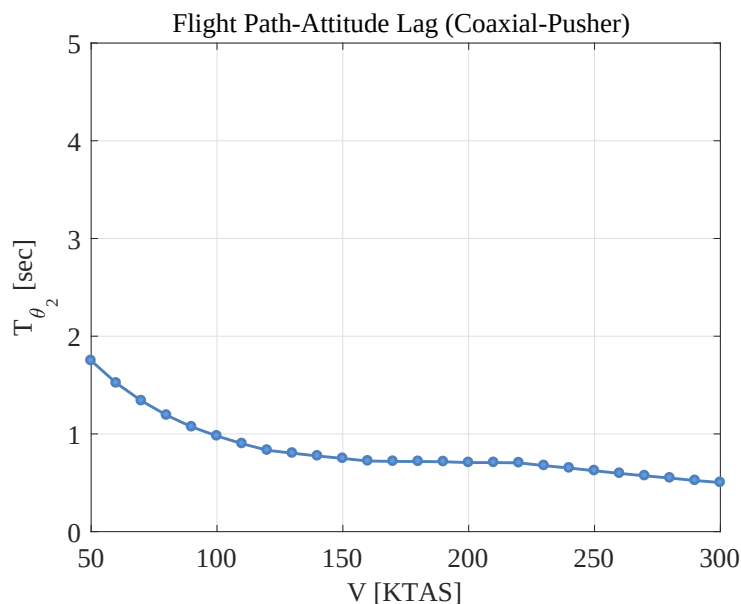


Figure 5.39. Bare-airframe flight path-attitude lag T_{θ_2} values (coaxial-pusher).

The consequences of the larger values of T_{θ_2} can be seen by comparing the outer-loop control system performance of the coaxial-pusher and tiltrotor at a common flight condition ($V_{CAS} = 180$ kts). Figure 5.41 shows the flight path response to pilot input γ/δ_{lon_s} . The flight path command models of both aircraft were tuned to give $\omega_{BW_\gamma} = 1.3$ rad/sec (Sec. 5.1.3), which can be seen in Figure 5.41 as the phase curves of the γ/δ_{lon_s} responses of both aircraft cross $\varphi = -135$ deg at $\omega = 1.3$ rad/sec.

To get similar flight path responses, though, the tiltrotor pitch rate response has a larger overshoot (associated with the larger value of T_{θ_2} , i.e., lower frequency $1/T_{\theta_2}$ zero), as seen in Figure 5.42. This leads to more pitch rate overshoot and pitch attitude dropback, as seen in the time histories in Figure 5.43.

Figure 5.44 shows the values of pitch attitude dropback for the inner- and outer-loop control laws for both aircraft, plotted against the specification boundary. Both inner-loop designs have zero dropback, as is expected for a pitch rate-command/attitude-hold response type. The outer-loop control laws dropback increases for the coaxial-pusher, but is well within the Level 1 region of the specification. For the tiltrotor, the outer-loop control laws have a value of pitch attitude dropback close to the Level 1/Level 2 boundary, significantly higher than the coaxial-pusher. Piloted evaluation of the outer loops will determine if the tiltrotor value of dropback is in fact acceptable as the dropback specification suggests.

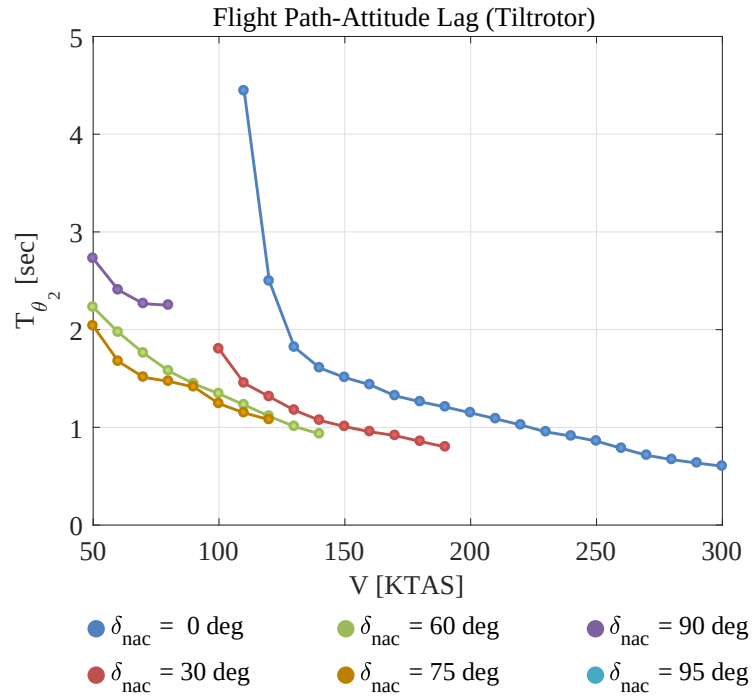


Figure 5.40. Bare-airframe flight path-attitude lag T_{θ_2} values (tiltrotor).

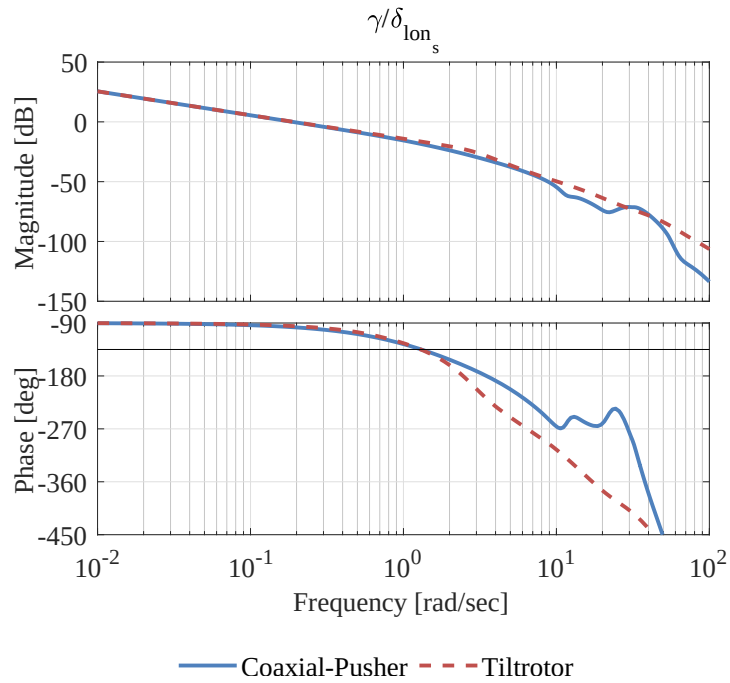


Figure 5.41. Flight path response to pilot stick input (coaxial-pusher and tiltrotor, outer loops, 180 kts).

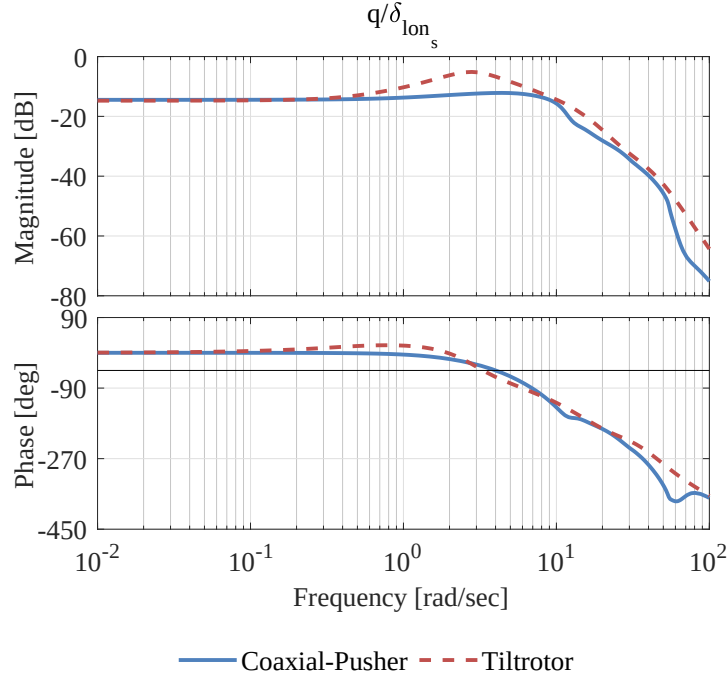


Figure 5.42. Pitch rate response to pilot stick input (coaxial-pusher and tiltrotor, outer loops, 180 kts).

5.7 Summary

This chapter discussed the development and optimization of full-flight envelope outer-loop control systems for both coaxial-pusher and tiltrotor aircraft. A dynamic inversion control system architecture was used for the outer loops of both aircraft, and was tuned to meet a common comprehensive set of stability, handling-qualities, and performance specifications. A lower-order (two state, three input) model was used to represent the airspeed V and climb rate $\dot{h}$ dynamics of the closed-loop bare-airframe and inner-loop control system. System identification of the closed-loop bare-airframe and inner-loop control system was used to determine the parameters of the lower-order model used for dynamic inversion. Weighted pseudo inverse control allocation was used to allocate the outer loop pitch attitude, symmetric main rotor collective, and pusher propeller collective command to the inner-loop control laws for the coaxial-pusher. In the case of the tiltrotor, the nacelle tilt angle was scheduled as a function of airspeed and commanded acceleration while the outer loops controlled pitch attitude and symmetric main rotor collective. Optimization results were shown for both aircraft at a range of airspeeds spanning the flight envelope and demonstrate the predicted Level 1 handling qualities of both configurations.

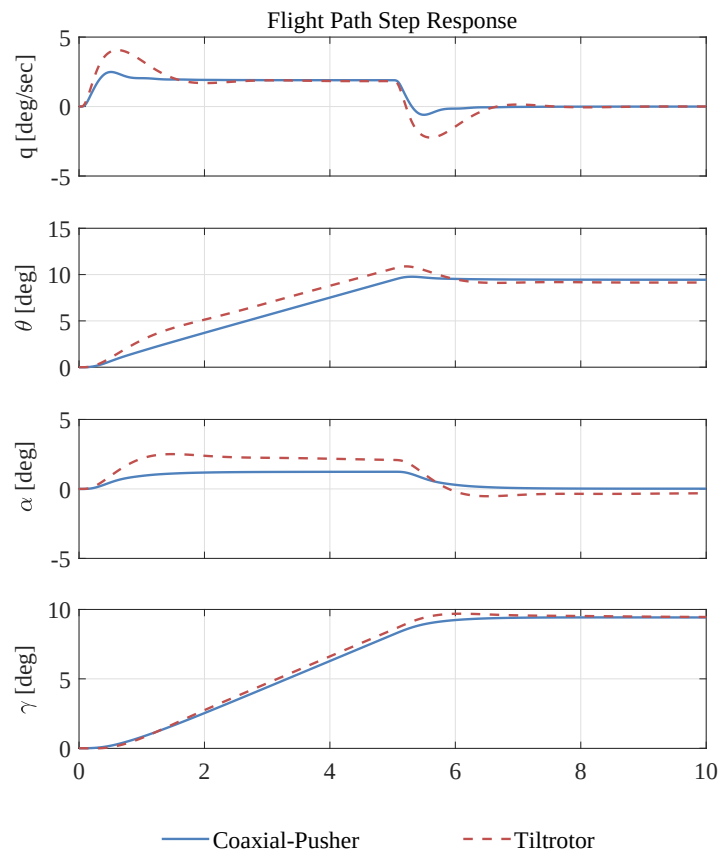


Figure 5.43. Flight path step response time history (coaxial-pusher and tiltrotor, outer loops, 180 kts).

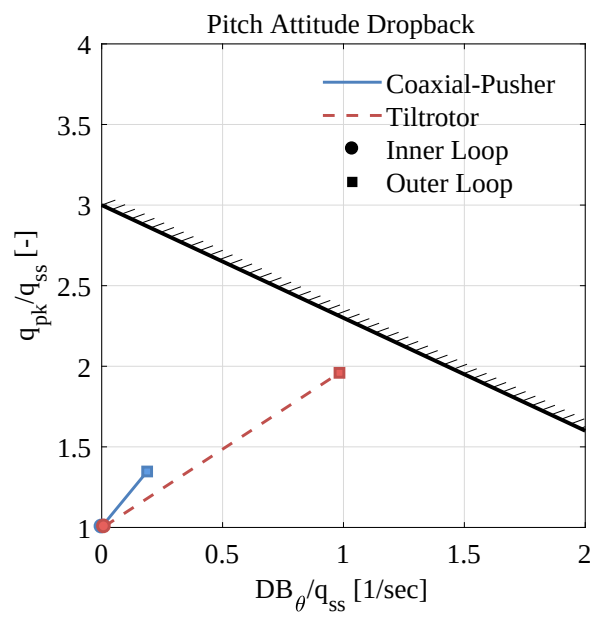


Figure 5.44. Pitch attitude dropback (coaxial-pusher and tiltrotor, inner and outer loops, 180 kts).

Chapter 6 |

Outer-Loop Control Law Handling Qualities Simulation

This chapter describes the results of a piloted handling qualities simulation experiment that was carried out at the Pennsylvania State University (PSU) Flight Simulator facility to test the outer-loop control laws described in Chapter 5. First, a description of the experimental setup will be provided, including a descriptions of the facility and the tasks flown by the pilots. Next, the results for the coaxial-pusher configuration will be provided followed by the results for the tiltrotor. Finally, a discussion of the results will be provided.

6.1 Experimental Setup

6.1.1 Simulation Facility

The handling qualities experiment was conducted in the PSU Flight Simulator facility, shown in Figure 6.1. The simulator consists of a raised Bell Helicopter BA609 simulation cab and a 5 m diameter spherical screen which provides 210 deg horizontal field of view and 50 deg vertical field of view. The simulator has motion capabilities, but they were not exercised for this experiment (i.e., experiment was carried out fixed-based).

6.1.1.1 Inceptors

The inceptor configuration consisted of a passive sidestick [shown in Figure 6.2(b)] attached on the right-hand side by the pilot seat, standard active pedals, and a standard active collective stick [shown in Figure 6.2(a)] using pull-for-power logic.



Figure 6.1. PSU Flight Simulator facility external view

6.1.1.1.1 *Inner-Loop Control*

Button control for the inner-loop control laws was discussed in Secs. 3.5.1 and 3.5.2, but will be discussed here for the PSU simulator inceptors. The collective had a discrete thumb hat switch [marked “1” in Figure 6.2(a)] to control the coaxial-pusher pusher propeller and the tiltrotor nacelle angle. Pushing forward on the hat switch increases pusher propeller collective and tilts the nacelles down towards airplane mode (increase airspeed in both cases). Pushing backward on the hat switch has the reverse effect. The sidestick inceptor also had a thumb hat switch [marked as “2” in Figure 6.2(b)] which provided the same functionality. However, pilots preferred to use hat switch on the collective to decouple attitude/altitude (right hand) control from airspeed control (left hand).

In addition to the hat switch on the sidestick, two special function push buttons for each platform were implemented in the switch marked “3” in Figure 6.2(b). For the coaxial-pusher, pressing button “3” back was the Set Zero Thrust button and commanded the pusher propeller to its zero thrust setting, thus allowing the pilot a way to perform a quick deceleration. Pressing button “3” up was the Couple Thrust to Airspeed button. This button set the pusher propeller collective to roughly match the value required to hold the current airspeed with ship level pitch attitude. Note that this is not an airspeed hold mode as there is no airspeed feedback, and is simply implemented as a pusher propeller versus airspeed trim map in the inner-loop control laws.

For the tiltrotor, button “3” was used to cycle the nacelles between preset angles $\delta_{\text{nac}} = 0, 30, 60, 75, 90, 95$ deg.

For the inner-loop control laws, force feel characteristics on the collective were different between the two aircraft configurations. For the coaxial-pusher, due to the collective trim map in the control laws (Sec. 3.5.1), the collective stick is always trimmed in the center position. Therefore, a detent and force gradient was present on the collective stick. For the tiltrotor, the collective serves the purpose of a throttle and in general is not trimmed in the center position. Therefore, the force gradient was disabled on the collective.

6.1.1.1.2 Outer-Loop Control

The outer-loop control laws used the unified control inceptor approach, as discussed in Sec. 5.5.1. Here, the collective stick is used to control longitudinal acceleration/deceleration throughout the flight envelope, as discussed in Sec. 5.1.3. In addition, pilots were also able to use the thumb hat switch on the collective [“1” in Figure 6.2(a)] to control acceleration and deceleration. Since this switch is discrete (commands 0 or 1), it commands maximum acceleration/deceleration when pressed.

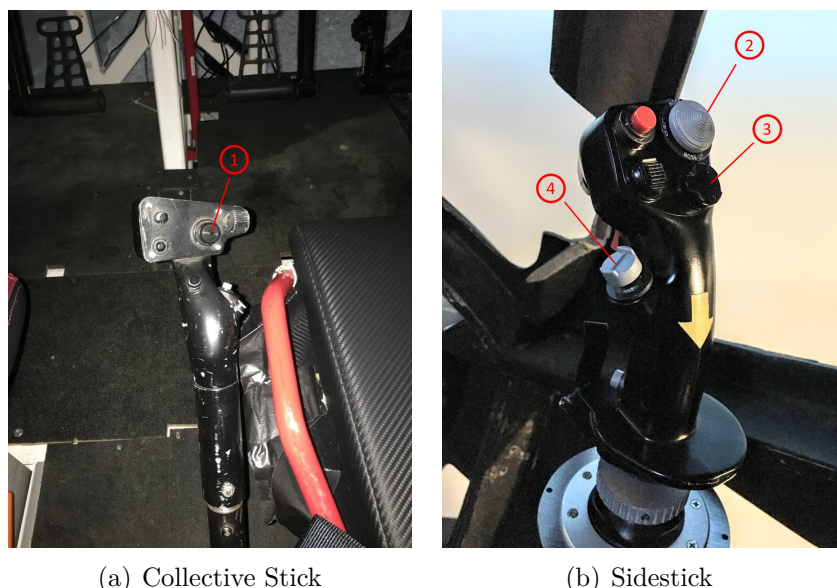


Figure 6.2. PSU Flight Simulator facility inceptors.

The sidestick is used to command climb rate $\dot{h}$ between hover and $V_{CAS} = 40$ kts and flight path rate $\dot{\gamma}$ above $V_{CAS} = 40$ kts (Sec. 5.1.3). The button marked “4” in Figure 6.2(b) was used to set the commanded flight path angle or climb rate to zero, thus giving the pilots an easy way to null out their climb rate at high-speed, rather than hunting for $\gamma = 0$ with the sidestick. In addition, if the sidestick was in detent, and $|\gamma| < 1$ deg, the flight path rate command model automatically commanded $\gamma = 0$.

Buttons “2” and “3” on the sidestick were not active when the outer-loop control laws were engaged.

For the outer-loop control laws, the force gradient was enabled on the collective for both aircraft, since it controls acceleration. Therefore, to hold airspeed, the pilots simply had to release the collective and it would return to the detent position and null out the acceleration command.

6.1.1.2 Display

Figure 6.3 shows the primary flight display (PFD) used in the PSU simulator. The display used standard symbology with two configuration-specific gauges on the left side. The top gauge shows propeller thrust for the coaxial-pusher configuration. The magenta ball displays the commanded propeller thrust, while the red and white needle displays the actual current propeller thrust. For the tiltrotor, the lower gauge displays the nacelle tilt angle. The magenta needle displays the current nacelle position while a green shaded section (not shown in the figure) shows the current allowable nacelle range based on airspeed. The torque display in the right was driven by the power required output of the models (described in Chapter 2).

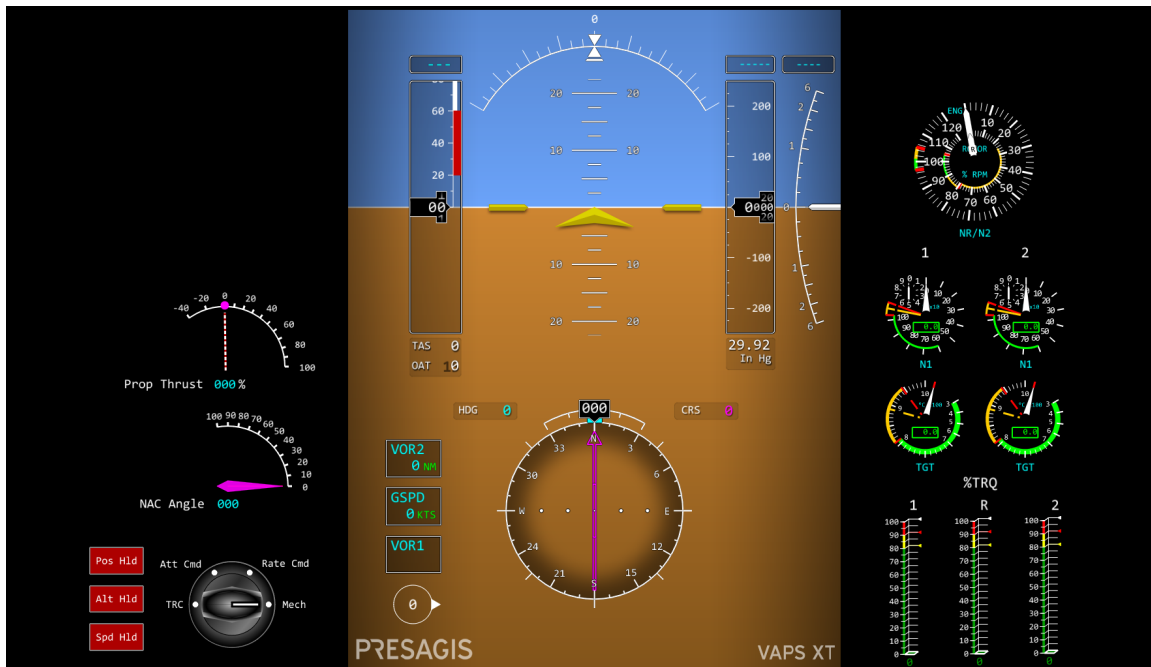


Figure 6.3. PSU Flight Simulator facility primary flight display (PFD)

Airspeed and climb rate bugs (not shown) were used to display the commanded values of the outer-loop control laws. In addition, airspeed and heading bugs were used during the Break Turn and High-Speed Acceleration/Deceleration MTEs to provide the pilots with additional cueing. For the Break Turn MTE, a heading bug was placed on the heading to be captures at the end of the turn and an airspeed bug was placed at the lower airspeed value required to maintain desired performance. For

the High-Speed Acceleration/Deceleration MTEs, an airspeed bug was placed on the airspeed to be captured at the end of the maneuver.

6.1.2 Nonlinear Simulation Model Validation

Before beginning handling qualities evaluations, implementation of models in the simulator was validated. This was done by conducting both closed-loop and broken-loop automated frequency sweeps of the nonlinear simulation models. The frequency sweep simulation data were analyzed using CIPHER[®] [83] to extract the appropriate frequency responses and compare to those of the linear point model used in the control law development. Three responses were analyzed in each axis: closed-loop piloted response, closed-loop disturbance response, and broken-loop response. The comparisons validate the implementation of the feed-forward and feedback sections of the control laws.

Figures 6.4 and 6.5 show the airspeed and flight path broken-loop frequency responses for both the nonlinear simulation model and linear analysis models of the coaxial-pusher at $V_{CAS} = 180$ kts. In both axes, there is an excellent agreement between the nonlinear simulation and linear models, validating the implementation of the stitched model, gain schedule, and control laws in the simulation model.

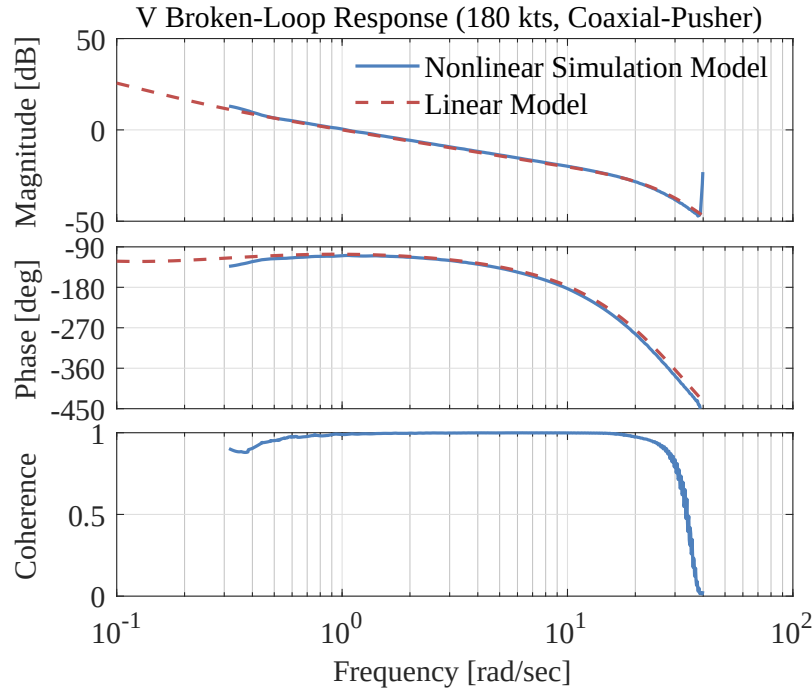


Figure 6.4. Airspeed broken-loop frequency response comparison (coaxial-pusher, 180 kts).

Figures 6.6 shows the closed-loop flight path response to pilot stick input for both the nonlinear simulation model and linear analysis models of the coaxial-pusher at

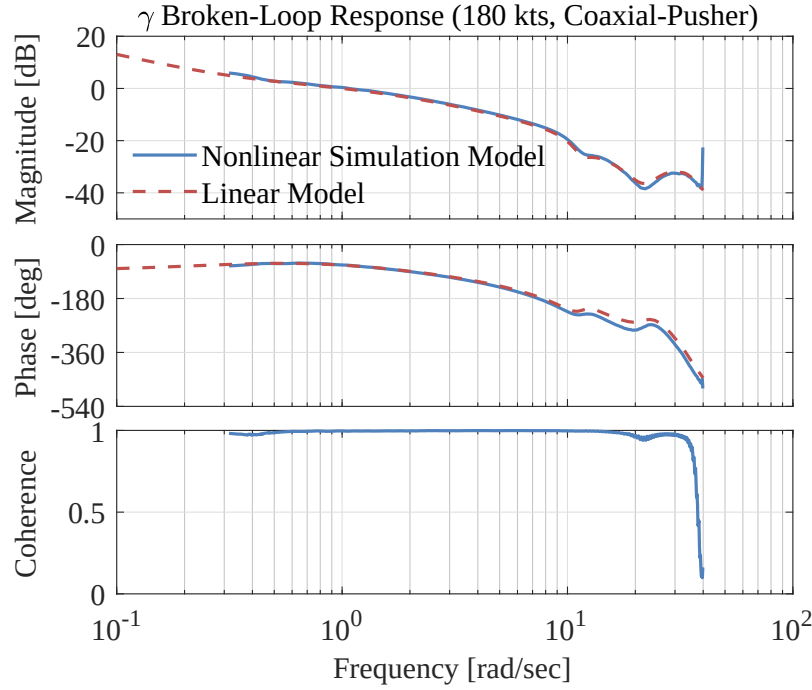


Figure 6.5. Flight path broken-loop frequency response comparison (coaxial-pusher, 180 kts).

$V_{CAS} = 180$ kts. Again, there is excellent agreement between the nonlinear simulation and linear models, validating the implementation of the simulation model.

Similar comparisons were done for the other axes and at additional airspeeds for the coaxial-pusher and tiltrotor simulation models. All validations showed excellent agreement with the analysis models.

6.1.3 Handling Qualities Task Definitions

A subset of the MTEs used in the inner loop handling qualities simulation experiment (Chapter 4) was used to test the outer loops. Since the outer loops only affect the longitudinal and heave axes, the MTEs tested were: Pitch Attitude Capture and Hold [44], Pitch Sum-of-Sines Tracking [45], Break Turn [42] and High-Speed Acceleration/Deceleration [43]. The tasks are described in detail in Sec. 4.1.3 and Appendix A which includes tables of desired and adequate performance bounds.

For both aircraft, Pitch Attitude Capture and Hold, Pitch Sum-of-Sines Tracking, and Break Turn MTEs were flown at an airspeed of $V_{CAS} = 180$ kts, while a speed range of $V_{CAS} = 50 - 220$ kts was used for the High-Speed Acceleration/Deceleration.

For the High-Speed Acceleration/Deceleration MTE, an analytical method (discussed in Sec. A.2.5) was used to determine the minimum time to complete the maneuver prior to testing. The following values of T_{minimum} were determined for the

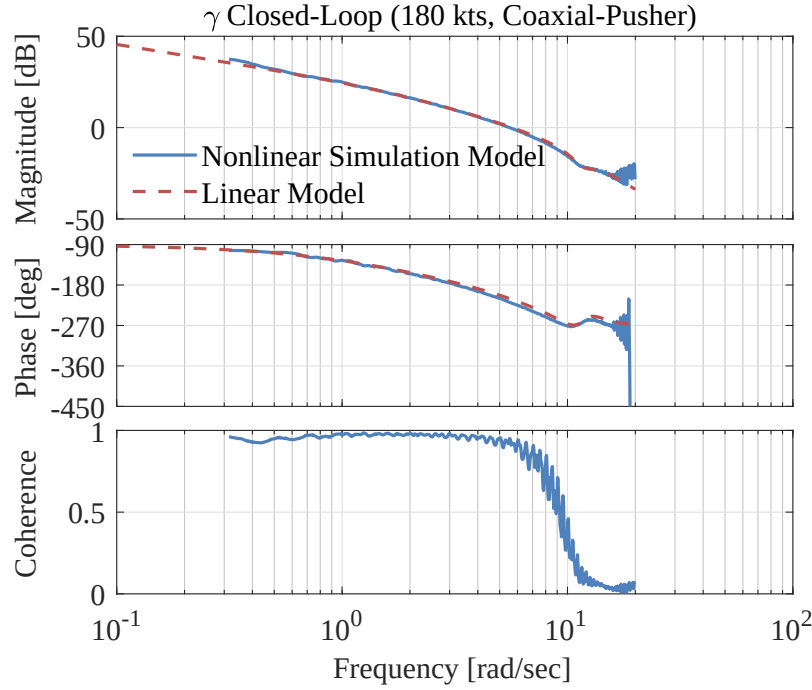


Figure 6.6. Flight path closed-loop frequency response comparison (coaxial-pusher, 180 kts).

coaxial-pusher:

$$\text{Acceleration: } T_{\text{minimum}} = 30 \text{ sec} \quad (6.1)$$

$$\text{Deceleration: } T_{\text{minimum}} = 22 \text{ sec} \quad (6.2)$$

and for the tiltrotor:

$$\text{Acceleration: } T_{\text{minimum}} = 30 \text{ sec} \quad (6.3)$$

$$\text{Deceleration: } T_{\text{minimum}} = 30 \text{ sec} \quad (6.4)$$

As an additional test of the benefits of the outer-loop control laws, a fifth task was added (Formation Flying) which was not a formally defined MTE. The task involved getting into formation with a KC-130 aircraft with a trailing refueling probe, as shown in Figure 6.7. The KC-130 was flying straight and level at $V_{\text{CAS}} = 180$ kts for this task (although some initial investigations had the KC-130 dynamics driven by a sum-of-sines signal). The aircraft being tested started 10 kts slower, 450 ft in back, 250 ft to the left, and 200 ft below the KC-130. A heads-up display (HUD) was used (as shown in Figure 6.8) as a reference for the pilot to determine the formation position, where the waterline symbol on the HUD was over the drogue (as shown in Figure 6.8). Since this was not a formal MTE with defined desired and adequate

standards, only comments were collected from the pilots for this task (with no HQRs taken).

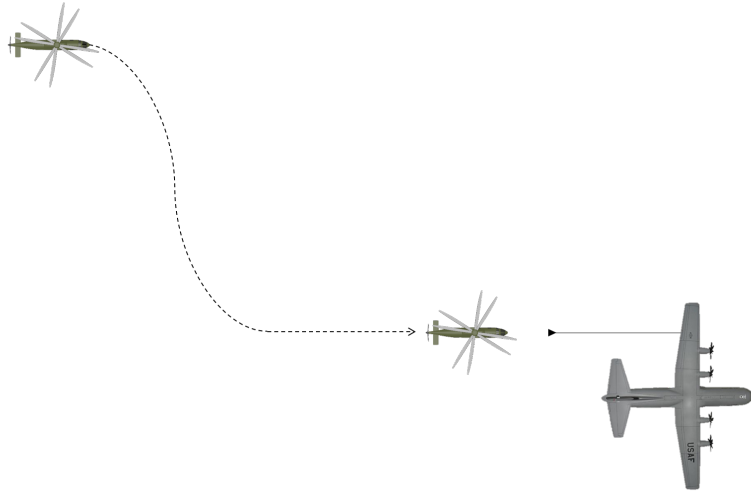


Figure 6.7. Top-down view of formation task (roughly to scale).

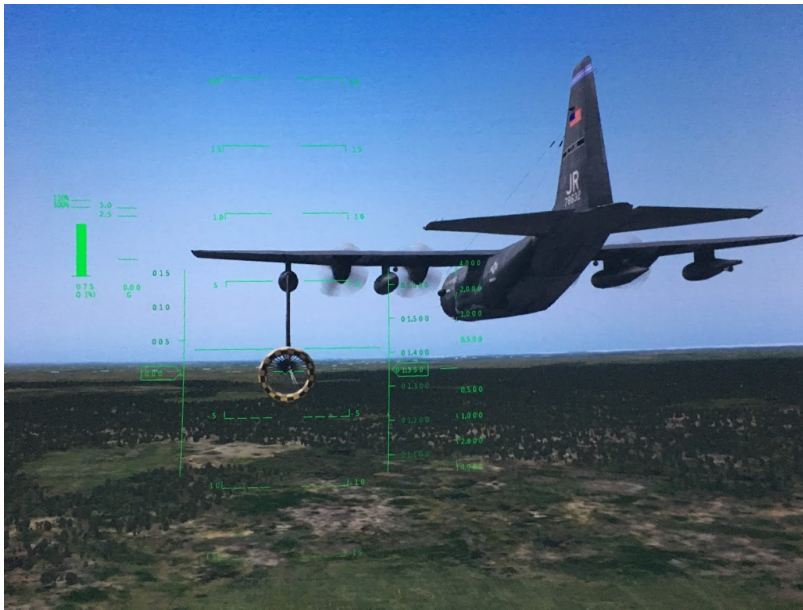


Figure 6.8. Pilot's view of formation position (note waterline symbol over refueling basket).

6.1.4 Pilot Questionnaire and Rating Scales

Appendix B shows the questionnaire and pilot rating scales that were provided to the pilots during their handling qualities evaluation. After several familiarization runs of

each maneuver with each aircraft/control laws combination, pilots would conduct two to three runs for record. Subsequently, they would answer a questionnaire specifically tailored for this experiment (given in Appendix B) and then use the Cooper-Harper Handling Qualities Rating Scale [30] to provide a Handling Qualities Rating (HQR). For the Break Turn, and High-Speed Acceleration/Deceleration MTEs, pilot also used the Bedford Workload Scale [112] to provide a Bedford Workload Rating (BWL). Finally, pilots were provided with a pilot induced oscillations (PIO) rating scale to use in case they encountered any PIO, however none were encountered during testing.

6.1.5 Pilot Demographics

Two U.S. Army experimental test pilots (XPs) participated in the handling qualities assessment. Both had over 2,400 hours of total flight time and primarily flight experience in H-1, H-60, H-58, and H-47, but other aircraft have been evaluated. One of the two pilots also participated in the VMS simulation (discussed in Chapter 4) and was familiar with the tasks, aircraft models, and inner-loop control systems. The other pilot had experience flying the high-speed MTEs on a UH-60M Black Hawk [46] and so was familiar with the tasks. He also had simulator tiltrotor experience.

6.2 Coaxial-Pusher Handling Qualities Results

Figure 6.9 shows the Handling Qualities Ratings (HQRs) collected for the coaxial-pusher aircraft for the tasks flow in the simulator. Results are shown for the inner-loop control laws (red squares) and outer-loop control laws (green triangles). In addition, the relevant results from the VMS simulation (discussed in Chapter 4) are shown for comparison (blue circles). The error bars represent average, maximum, and minimum ratings collected.

Average HQRs for the inner loop from the VMS experiment and PSU simulator experiment are within 1 HQR for all of the tasks, suggesting that the PSU simulator provided comparable fidelity to the VMS for conducting high-speed handling qualities evaluations.

Comparing the PSU simulator results for the inner- and outer-loop control laws, the outer-loop control laws improve the Break Turn ratings from borderline Level 1/Level 2 to Level 1. In addition, a half HQR point improvement is seen for the High-Speed Acceleration task, while the ratings for the High-Speed Deceleration are the same between the inner and outer loops. For both the Pitch Sum-of-Sines Tracking and Pitch Attitude Capture and Hold MTEs, the results for the outer loops are borderline Level 1/Level 2 while the result for the inner loop are Level 1. The following sections will discuss the results from the PSU simulator experiment for each MTE in more detail.

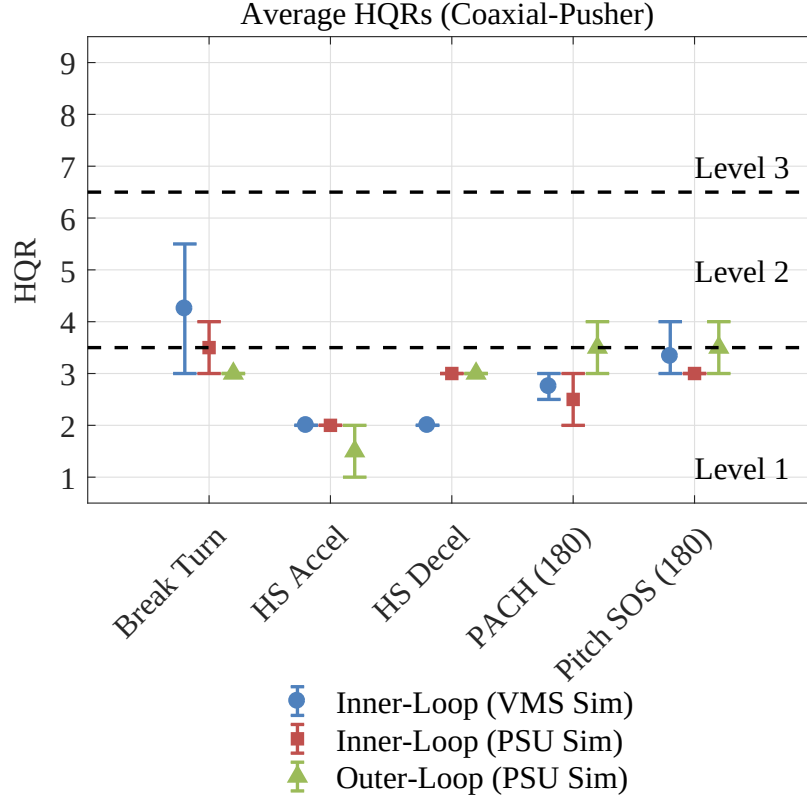


Figure 6.9. Handling qualities rating summary (coaxial-pusher).

6.2.1 Break Turn

Figure 6.10(a) shows the Break Turn MTE performance of the coaxial-pusher inner-loop control laws for all data runs for both pilots, plotted against the desired and adequate bounds listed in Table A.1. Both pilots were able to meet desired performance, although both also had one record run with the time to complete beyond desired (i.e., adequate performance).

Figure 6.10(b) shows the Break Turn MTE performance of the coaxial-pusher outer-loop control laws for all data runs for both pilots. With the outer-loops engaged, times to complete for both pilots were shorter. In addition, there was less variability in airspeed V and altitude h during the maneuver, and the bank angle ϕ traces were qualitatively smoother.

Pilot A rated the Break Turn with the inner-loop control laws an HQR 3 and a Bedford Workload Rating (BWL) 4. For the outer-loop control laws, Pilot A rated the Break Turn MTE an HQR 3 and BWL 3. He commented that in the roll axes, there was no change between the inner- and outer-loop control laws. Furthermore, although he was consistently able to make desired performance with both control laws, there was “increased workload” with the inner loop, while it was “very easy”

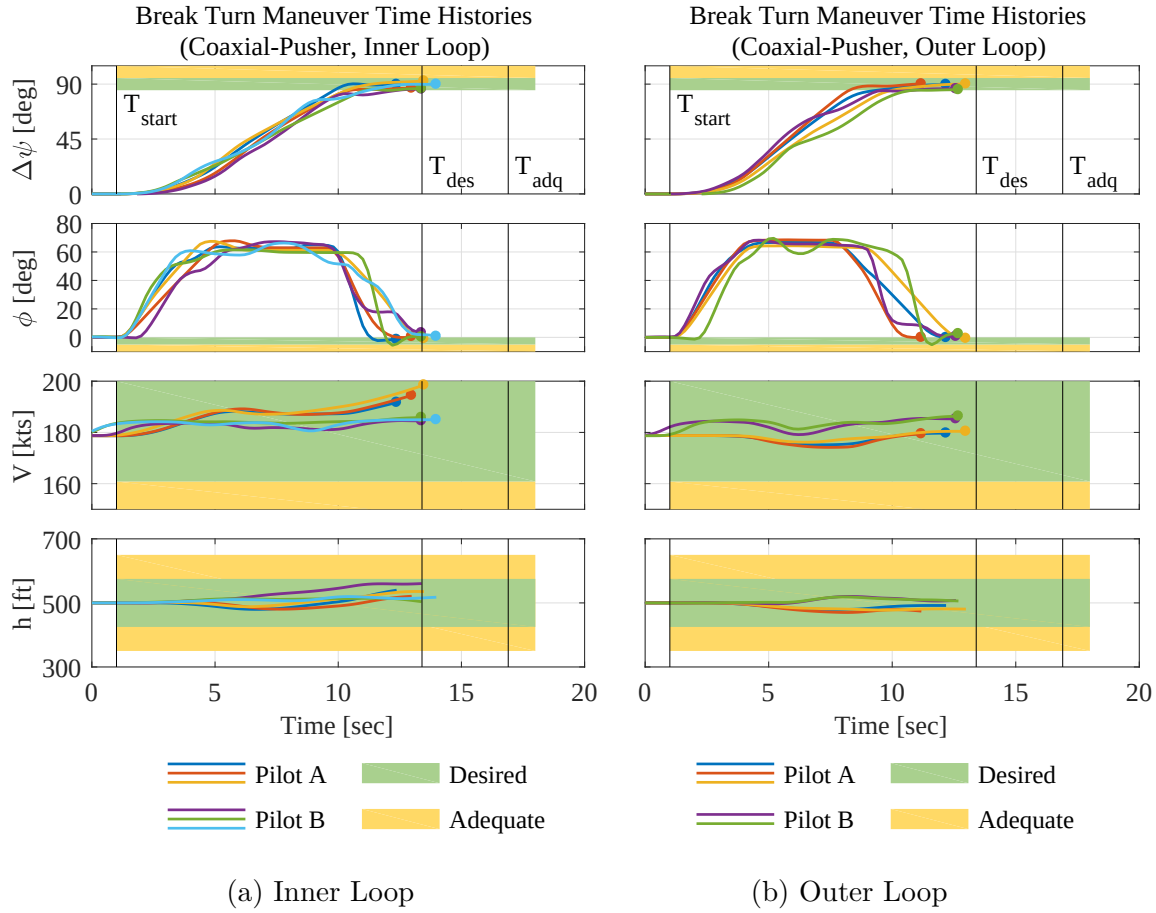


Figure 6.10. Break Turn task performance for (a) inner-loop control laws and (b) outer-loop control laws (coaxial-pusher, 180 kts).

with the outer loop (as reflected in the BWLR difference). Pilot A noted that when flying the task with the inner-loop control laws, he had to worry about airspeed and altitude in addition to the roll attitude/heading task. Pilot A’s technique was to max out the pusher propeller thrust before entering the turn, since there is no high-speed penalty for this task (only airspeed loss is specified in the task performance, although increasing airspeed does increase the time to complete the maneuver). This can be seen in Pilot A’s airspeed traces in Figure 6.10(a), which increase throughout the turn.

With the outer-loop control laws, Pilot A commented that “airspeed maintenance was not required at all” and that the control laws do a “good job of holding altitude.” He commented that his inputs were almost purely on the lateral stick, with several small pressure/counter-pressure inputs on the longitudinal stick to take out occasional altitude deviations. Pilot A commented that he rated the task an HQR 3 with the outer-loops because the most critical phase of the maneuver, which was targeting the desired heading and not overshooting bank angle on the roll out was similar to with the inner-loop control laws (as expected since the roll axis is identical between the

two sets of control laws).

Pilot B rated the Break Turn an HQR 4 and a BWLR 4 for the inner-loop control laws and an HQR 3 and BWLR 4 for the outer-loop control laws. He commented that with the inner-loop control laws, the most challenging part of the maneuver was altitude control (which resulted in the HQR 4 rating), as well as not overshoot bank angle on the heading capture at the end of the maneuver. With the outer-loop control laws, Pilot B was able to be more aggressive and get shorter times to complete.

Figure 6.11 shows the pilot stick activity (cutoff frequency ω_{co} and RMS) for all axes during the Break Turn maneuver for both inner- and outer-loop control laws. The error bars represent the average, maximum, and minimum values for all of the runs. For control axes with RMS = 0, ω_{co} was not calculated. The cutoff frequency ω_{co} is defined as the half-power frequency of the pilot stick signal determined via spectral analysis, and is a measure of the pilot's operating frequency [83]. Pilots did not have to use the pedals or collective with either set of control laws. Pilots also had comparable ω_{co} and stick RMS values for the lateral stick for both control laws, which correlates well with their comments that the rolling part of the maneuver was similar for both sets of control laws. The main difference is for the longitudinal stick. For the outer-loop control laws, the longitudinal stick RMS is on average about half of what it was for the inner-loop control laws, since the control laws maintain altitude during the turn. Finally, pilots did not have to use the thumb hat to adjust airspeed before or during the turn with the outer-loop control laws, although since this was a discrete inceptor, a cutoff frequency and RMS analysis was not done.

Overall, the results demonstrate the excellent airspeed and altitude hold performance of the outer-loop control laws. In addition, the pilot ratings show an improvement from borderline Level 1/Level 2 handling qualities with the inner-loop control laws to Level 1 handling qualities with the outer loop control laws. An average half-point reduction was also seen in the Bedford Workload Rating.

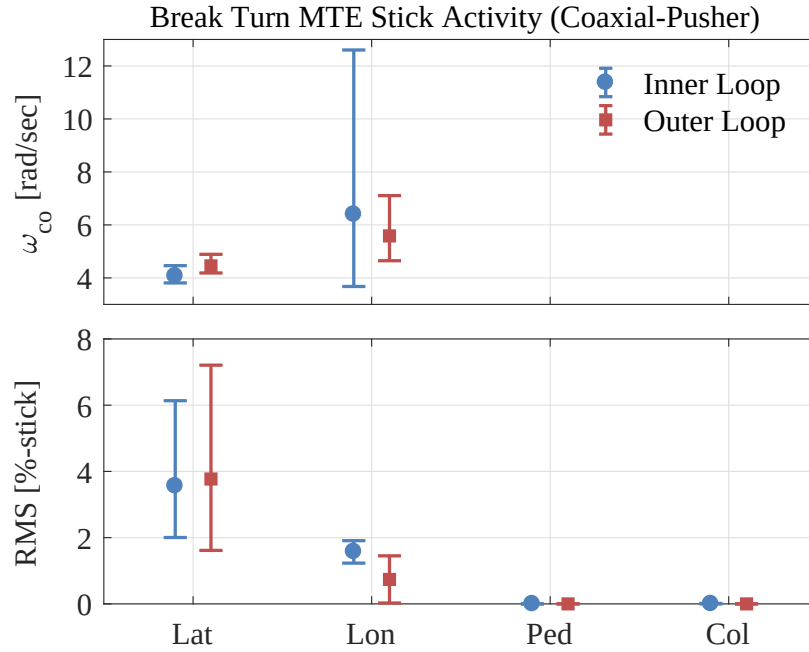


Figure 6.11. Break Turn stick activity (coaxial-pusher, 180 kts).

6.2.2 High-Speed Acceleration/Deceleration

Figure 6.12(a) shows the High-Speed Acceleration MTE performance of the coaxial-pusher inner-loop control laws for all data runs for both pilots, plotted against the desired and adequate bounds listed in Table A.3. Both pilots were able to meet desired performance.

Figure 6.12(b) shows the High-Speed Acceleration MTE performance of the coaxial-pusher outer-loop control laws for all data runs for both pilots. Both pilots were also able to meet desired performance with the outer-loops engaged, and do so with faster times to complete the maneuver. This is because the outer-loop control system supplements the pusher propeller commands with pitch attitude θ to accelerate at lower speeds, as seen by comparing the last subplot in Figures 6.12(a) and (b).

Pilot A rated the High-Speed Acceleration MTE with the coaxial-pusher inner-loop control laws an HQR 2 (note, since this is a low-workload tasks, Bedford Workload Ratings were not collected). Pilot A commented that it was “effortless to meet desired [performance]” for this task. He did not have to make any longitudinal inputs to stay within the ± 100 ft altitude tolerance bounds, but noted that he could “because there [was] nothing else to do.”

Pilot A also rated the High-Speed Acceleration MTE with the coaxial-pusher outer-loop control laws an HQR 2. He noted again that it was easy to meet desired performance, but that it took slightly longer to null out the acceleration with the outer-loops than with the inner-loops.

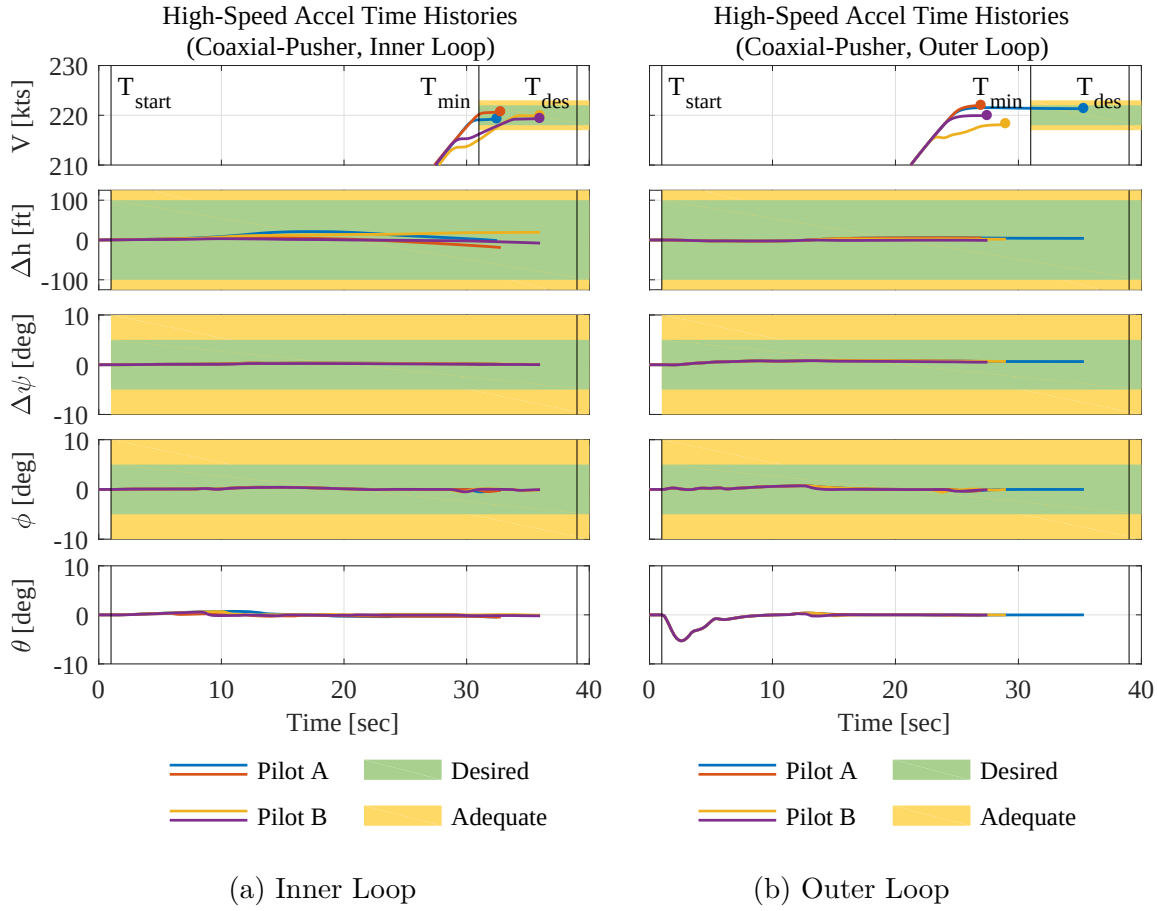


Figure 6.12. High-Speed Acceleration task performance for (a) inner-loop control laws and (b) outer-loop control laws (coaxial-pusher, 180 kts).

Pilot B rated this task an HQR 2 with the inner-loop control laws and an HQR 1 with the outer-loop control laws. Pilot B similarly commented that it was “fairly easy to meet desired [performance]” with both inner- and outer-loop control laws.

Figure 6.13(a) shows the High-Speed Deceleration MTE performance of the coaxial-pusher inner-loop control laws for all data runs for both pilots, plotted against the desired and adequate bounds listed in Table A.3. Both pilots were able to meet desired performance.

Figure 6.13(b) shows the High-Speed Deceleration MTE performance of the coaxial-pusher outer-loop control laws for all data runs for both pilots. Both pilots were also able to meet desired performance with the outer loops engaged, although tended to have more overshoot on the airspeed capture at the end of the maneuver.

Pilot A rated the High-Speed Deceleration MTE with the coaxial-pusher inner-loop control laws an HQR 3. He performed the entire maneuver with two buttons presses, first the Set Zero Thrust button, and then the Couple Thrust to Airspeed button, and noted that no sidestick, pedals, or collective inputs were necessary. The critical phase

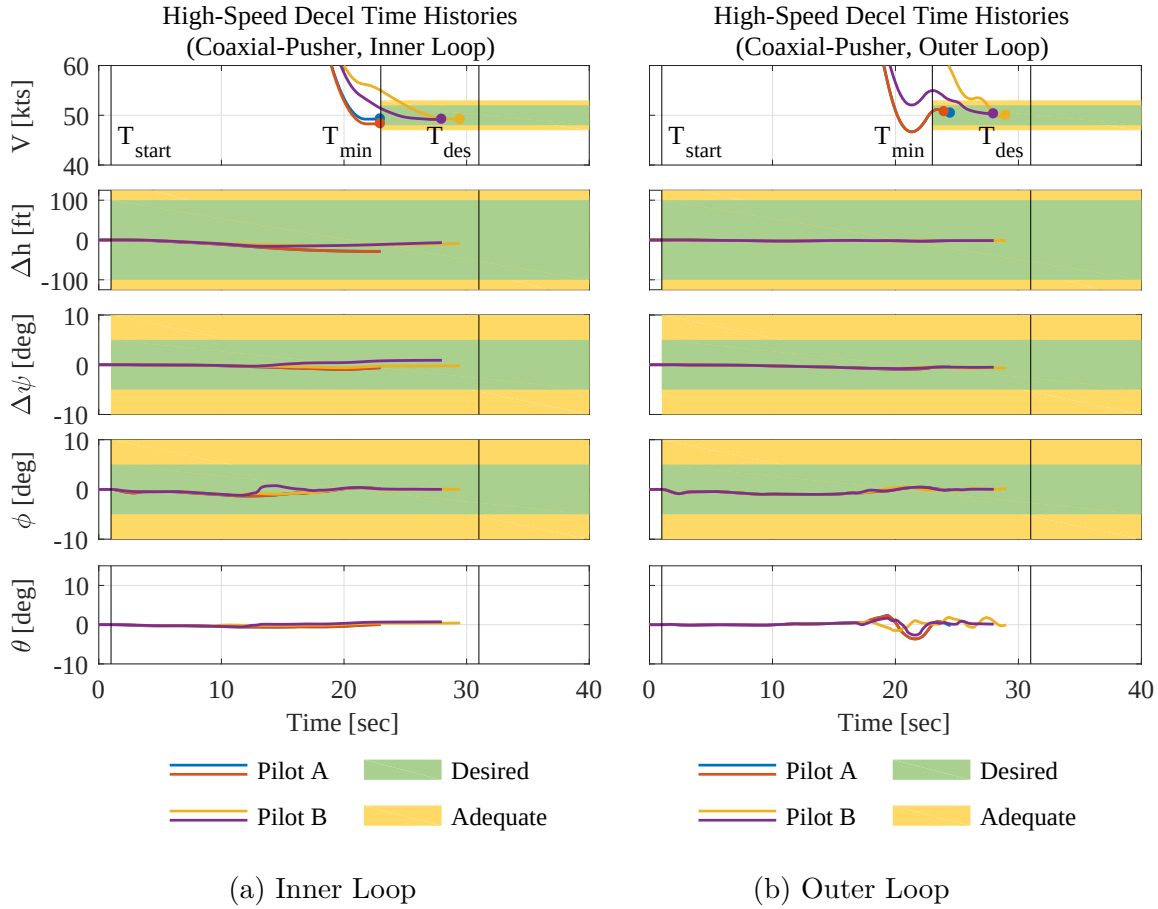


Figure 6.13. High-Speed Deceleration task performance for (a) inner-loop control laws and (b) outer-loop control laws (coaxial-pusher, 180 kts).

of the task was timing nulling the deceleration with the Couple Thrust to Airspeed button press, which could be done precisely (within the ± 2 kts desired bounds).

With the outer-loop control laws, Pilot A similarly rated the High-Speed Deceleration MTE an HQR 3. Pilot A noted that he could be more precise using the collective for deceleration control rather than the thumb hat switch (since the thumb hat switch can only command maximum deceleration). Therefore, with the thumb hat switch he could be aggressive, but some guess work was required on the timing aspect which resulted in more over-/under-shoots on speed captures.

Pilot B rated High-Speed Deceleration MTE an HQR 3 for both the inner- and outer-loop control laws. He noted that with the inner-loop control laws, there was an inability to accurately stop airspeed. However, he also noted that predictability of the outer-loop control laws was not as good on the deceleration as it was on the acceleration.

Stick activity was not analyzed for either the High-Speed Acceleration or Deceleration MTEs for the coaxial-pusher, since the pilots did not need to use the sidestick or

collective for altitude control with either the inner- or outer-loop control laws.

Overall, the coaxial-pusher inner-loop control laws with direct control of the pusher propeller thrust, provide Level 1 handling qualities for the High-Speed Acceleration/Deceleration tasks. In this case, the outer-loop control laws provided the same handling qualities Level as the inner-loop control laws.

6.2.3 Pitch Angle Capture and Hold

Figure 6.14 shows two example record runs for the Pitch Angle Capture and Hold MTE, one with the inner-loop control law and one with the outer-loop control laws. The two runs shown are representative of the additional record runs collected. Pilots were able to meet desired performance with both inner- and outer-loop control laws.

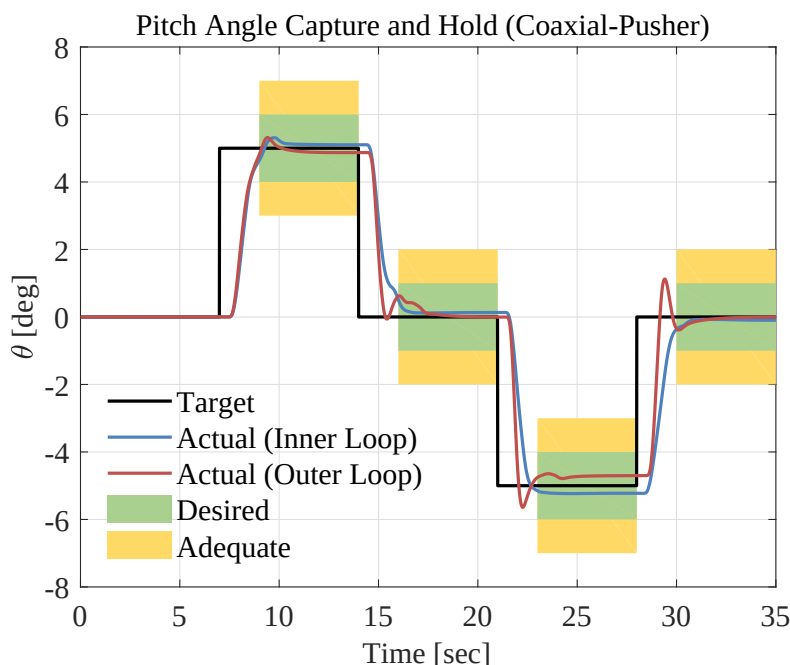


Figure 6.14. Pitch Angle Capture and Hold example time history (coaxial-pusher, 180 kts).

Pilot A rated the Pitch Angle Capture and Hold MTE for the coaxial-pusher inner-loop control laws an HQR 2, and noted that he could be aggressive and that precision was not a problem. With the outer-loop control laws, Pilot A rated the Pitch Angle Capture and Hold MTE an HQR 4. He commented that the initial response was faster than he expected, and noted the pitch attitude dropback characteristic that was not present with the inner-loop control laws. He was able to compensate for the dropback, but referred to it as “annoying” which led to the Level 2 HQR.

Pilot B rated the Pitch Angle Capture and Hold MTE an HQR 3 for both the inner- and outer-loop control laws. He noted that it was “easy to meet desired” with

the inner-loop control laws and that the response was “very predictable.” With the outer-loop control laws, Pilot B noted that it required more control inputs to get same effect as the inner-loop control laws since he was compensating for the dropback characteristic.

Figure 6.15 shows the pitch stick activity for both inner- and outer-loop control laws during the Pitch Angle Capture and Hold MTE. The stick RMS is slightly lower for the outer-loop control laws, although the pilot cutoff frequency is higher, corresponding to the additional compensation the pilots were required to do to account for the pitch attitude dropback. This correlates with the degraded handling qualities assigned to the outer-loop controls laws (HQR 3.5) as compared to the inner-loop control laws (HQR 2.5) for this task.

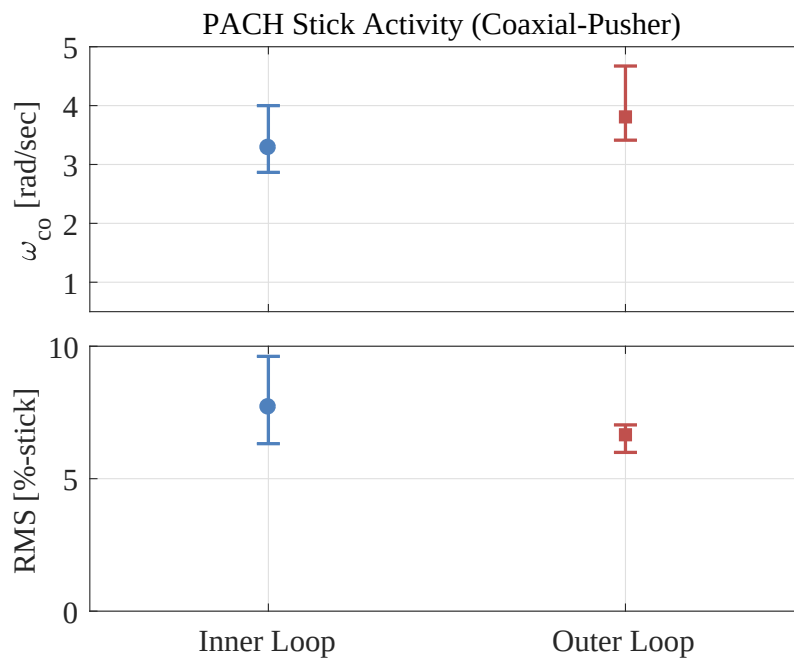


Figure 6.15. Pitch Angle Capture and Hold stick activity (coaxial-pusher, 180 kts).

6.2.4 Pitch Sum-of-Sines Tracking

Figure 6.16 shows the tracking performance for the Pitch Sum-of-Sines MTE for the coaxial-pusher. Only Pilot A flew this task for record, and was able to attain desired performance with both the inner- and outer-loop control laws.

Pilot A rated the inner-loop control laws an HQR 3 for this task and commented that the task was “not overly taxing.” He further noted that there was not much of a penalty for attempting to over-compensate in that he could stop the pitch rates generated and move back in other the direction quickly.

For the outer-loop control laws, Pilot A had similar task performance to the inner-loop control laws and also rated the task an HQR 3. However, for the outer loops he commented that although he was able to meet desired performance, it was “more difficult to be precise” and he was “working harder.” He further noted that the reversals were more difficult which led him to be less aggressive.

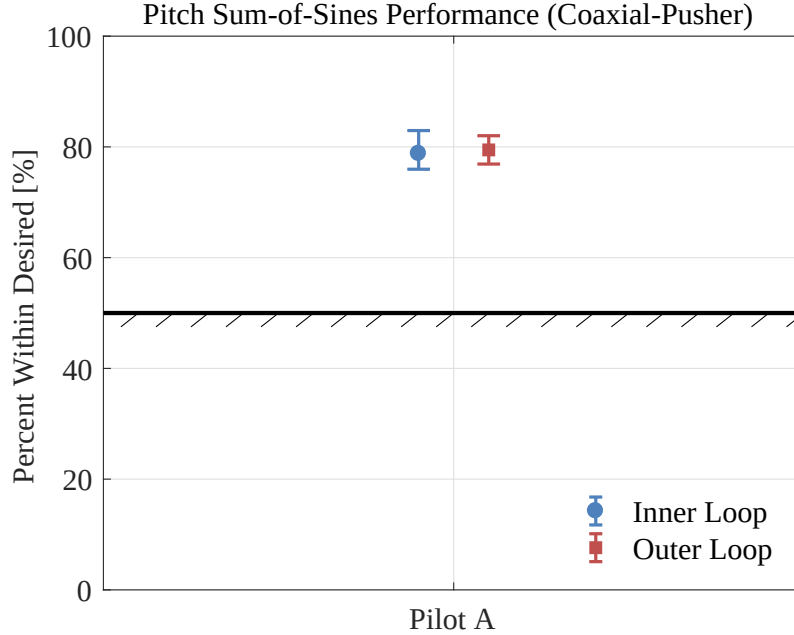


Figure 6.16. Pitch Sum-of-Sines Tracking performance (coaxial-pusher, 180 kts).

A quantitative comparison of the two control laws was done by investigating the characteristics of the pilot Y_p vehicle Y_c system (PVS), as described in Sec. 4.4.5. In the case of the Sum-of-Sines Tracking tasks, the forcing function of the PVS is known, and so all elements of the PVS can be identified and compared between the two control laws.

Figure 6.17 shows the aircraft frequency response $Y_c = \theta/\delta_{lon_s}$ extracted from the time history data using CIPHER[®], with discrete sum-of-sines frequencies highlighted. As discussed in Sec. 5.6, the outer-loop control laws have some additional pitch attitude dropback which can be seen as the hump in the magnitude curve between $\omega = 2 - 10$ rad/sec.

Figure 6.18 shows the PVS broken-loop frequency response ($Y_p Y_c$) extracted from the time history data at the discrete frequencies used in the tracking signal using CIPHER[®]. The figure also shows a crossover model [108] fit to the each response, given by:

$$Y_p Y_c = \frac{\omega_c}{s} e^{-\tau s} \quad (6.5)$$

where:

Y_p denotes the pilot response ($\delta_{\text{lon}_s}/\theta_{\text{error}}$),
 Y_c denotes the aircraft response ($\theta/\delta_{\text{lon}_s}$),
 ω_c is the crossover frequency, and
 τ is the equivalent time delay.

The parameters of the crossover model are given in Table 6.1 for both inner- and outer-loop control laws.

Finally, Figure 6.19 shows the pilot frequency response $Y_p = \delta_{\text{lon}_s}/\theta_{\text{error}}$.

For the coaxial-pusher, the pilot was able to compensate by lowering his gain at high frequency ($\omega = 2 - 10$ rad/sec in Figure 6.19) to maintain the same performance and overall PVS characteristics. This correlates with the lower cutoff frequency ω_{co} for the outer-loop control laws and the pilot's comment that he had to be less aggressive with the outer loops. This adjustment by the pilot to compensate for the difference in aircraft response led to similar PVS characteristics (such as crossover frequency ω_c and gain and phase margin) between the inner- and outer-loop control laws (Table 6.1).

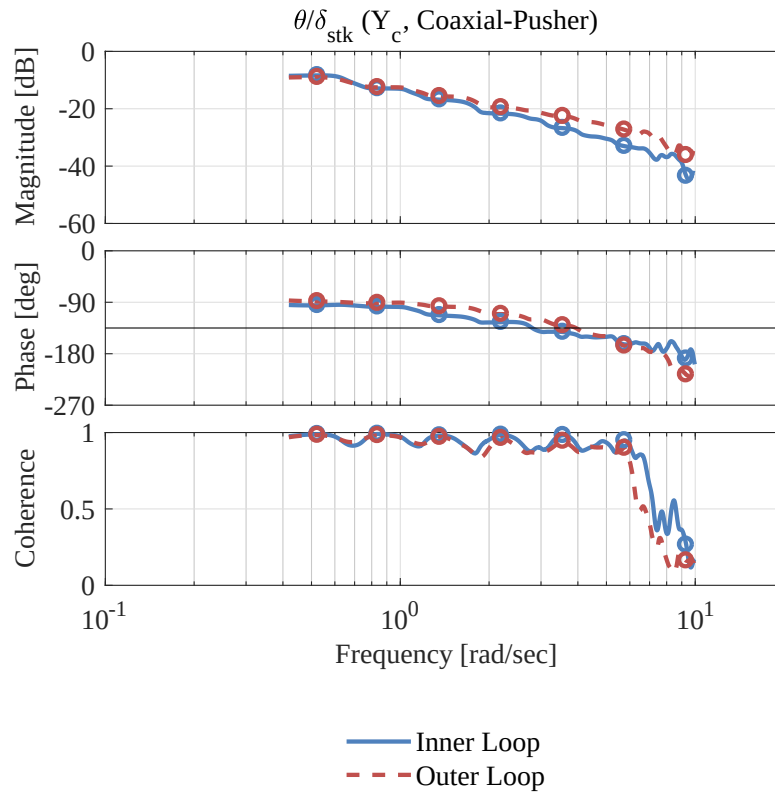


Figure 6.17. Vehicle frequency responses comparison for Pitch Sum-of-Sines Tracking task (coaxial-pusher, 180 kts).

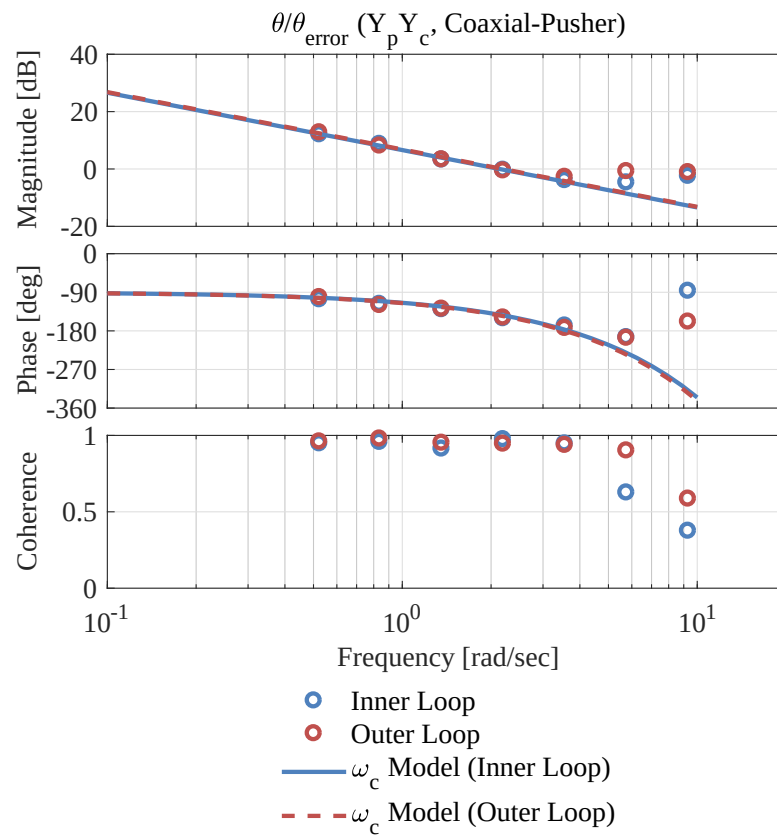


Figure 6.18. Pilot-Vehicle System broken-loop frequency responses comparison for Pitch Sum-of-Sines Tracking task (coaxial-pusher, 180 kts).

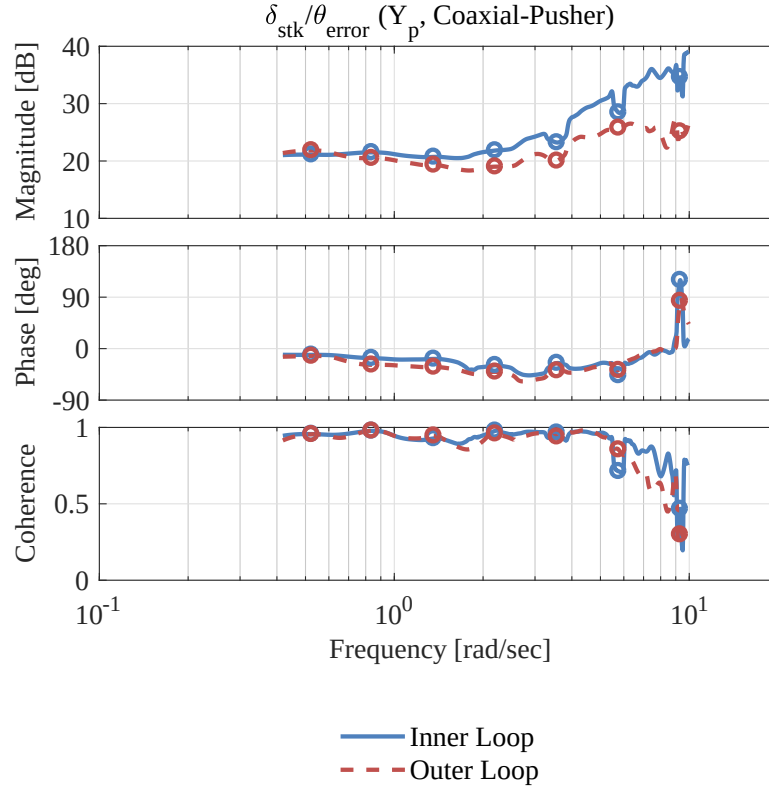


Figure 6.19. Pilot frequency responses comparison for Pitch Sum-of-Sines Tracking task (coaxial-pusher, 180 kts).

Table 6.1. Pilot-Vehicle Parameters for Pitch Sum-of-Sines Tracking Task (Coaxial-Pusher, 180 kts)

	Inner Loop	Outer Loop
<i>Pilot-Vehicle System Parameters ($Y_p Y_c$)</i>		
Pilot-Vehicle Crossover Frequency ω_c [rad/sec]	2.14	2.19
Gain Margin [dB]	4.68	4.22
Phase Margin [deg]	37.57	34.68
Pilot-Vehicle Equivalent Time Delay τ [sec]	0.43	0.44
<i>Pilot Parameters (Y_p)</i>		
Pilot Gain K_p	13.32	11.19
Time Delay τ_p [sec]	0.16	0.17
Cutoff Frequency ω_{co} [rad/sec]	1.37	1.11
Stick RMS [%]	12.89	11.31

6.2.5 Formation Flying

Figure 6.20 show the Formation Flying task time histories for two of Pilot B’s runs with the coaxial-pusher—one with the inner-loop control laws and one with the outer-loop control laws. The figure shows the position error between the coaxial-pusher and the formation position as described in Sec. 6.1.3, as well as the airspeed difference and climb rate traces for both control laws. Although Pilot B was able to move into the formation position equally well with both control laws, the climb rate trace in Figure 6.20 (bottom subplot) is less oscillatory with the outer-loop control laws.

Table 6.2 lists the lateral and longitudinal stick activity during this task. In the lateral axis, there was a 24% reduction in stick RMS for the outer-loop control laws as compared to the inner-loop control laws, while the cutoff frequency remained almost the same. In the longitudinal axis, where the outer-loop control laws are active, the stick RMS for the outer-loop control laws was 43% less than for the inner-loop control laws, while the cutoff frequency was about 25% less. These values demonstrate the lower workload of moving into the formation position with the outer-loop control laws than inner-loop control laws, and matched the pilot comments.

Table 6.2. Formation Flying Stick Activity (Coaxial-Pusher)

	Inner Loop	Outer Loop
<i>Lateral Stick</i>		
Cutoff Frequency ω_{co} [rad/sec]	2.44	2.29
Stick RMS [%]	3.71	2.82
<i>Longitudinal Stick</i>		
Cutoff Frequency ω_{co} [rad/sec]	1.81	1.36
Stick RMS [%]	4.53	2.56

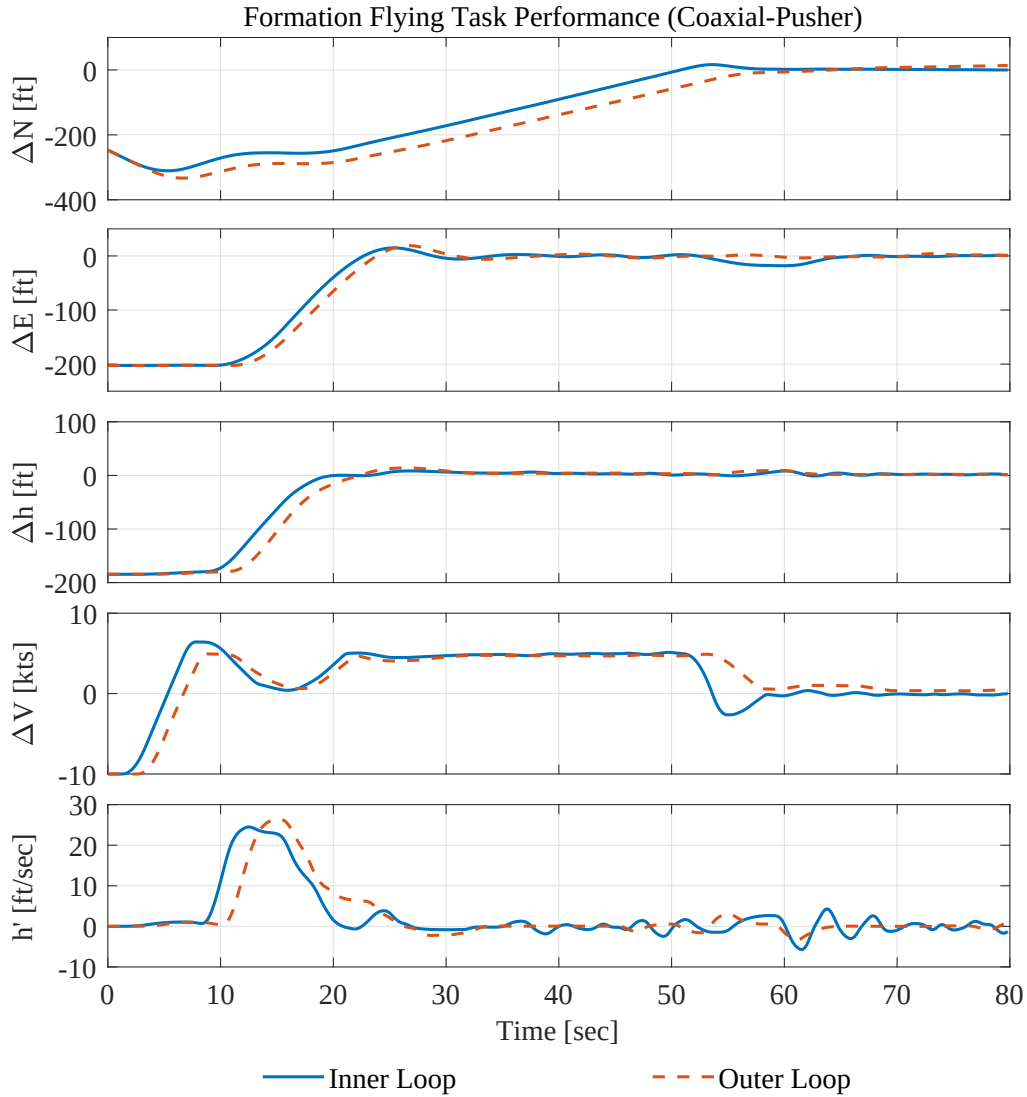


Figure 6.20. Formation flying task performance (coaxial-pusher).

6.3 Tiltrotor Handling Qualities Results

Figure 6.21 shows the Handling Qualities Ratings (HQRs) collected for the tiltrotor aircraft for the tasks flow in the PSU simulator. Results are shown for the inner-loop control laws (red squares) and outer-loop control laws (green triangles). In addition, the relevant results from the VMS simulation (discussed in Chapter 4) are shown for comparison (blue circles). The error bars represent average, maximum, and minimum ratings collected.

As with the coaxial-pusher results, average HQRs for the inner loop from the

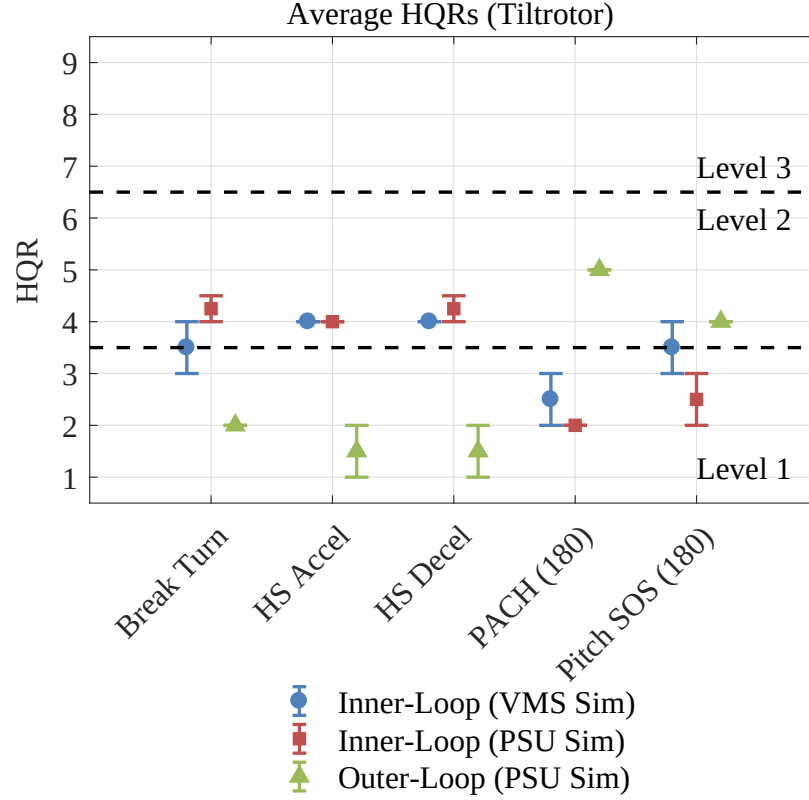


Figure 6.21. Handling qualities rating summary (tiltrotor).

VMS experiment and PSU simulator experiment are within 1 HQR for all of the tasks, suggesting that the PSU simulator provided comparable fidelity to the VMS for conducting high-speed handling qualities evaluations.

Comparing the PSU simulator results for the inner- and outer-loop control laws, the outer-loop control laws improve the Break Turn ratings from Level 2 (average HQR 4.25) to Level 1 (average HQR 2). A significant improvement in handling qualities with the outer-loop control laws is also seen for the High-Speed Acceleration and Deceleration MTEs, with ratings going from Level 2 (average HQR 4 for High-Speed Acceleration and average HQR 4.25 for High-Speed Deceleration) for the inner-loop control laws to Level 1 (average HQR 1.5 for both High-Speed Acceleration and Deceleration) for the outer-loop control laws.

In the case of the Pitch Attitude Capture and Hold and Pitch Sum-of-Sines Tracking MTEs, the results degraded for the outer-loop control laws, going from Level 1 ratings for the inner-loop control laws to Level 2 for the outer-loop control laws. The following sections will discuss the results for each MTE in more detail.

6.3.1 Break Turn

Figure 6.22(a) shows the Break Turn MTE performance of the tiltrotor inner-loop control laws for all data runs for both pilots, plotted against the desired and adequate bounds listed in Table A.1. Pilot A was able to meet desired performance, while Pilot B met desired performance on all parameters except time to complete the maneuver (which was inside the adequate bounds).

Figure 6.22(b) shows the Break Turn MTE performance of the tiltrotor outer-loop control laws for all data runs for both pilots. With the outer-loops engages, times to complete for both pilots were shorter and both pilots met desired performance on all of their runs. In addition, there was less variability in airspeed V and altitude h during the maneuver, and the bank angle ϕ traces were qualitatively smoother.

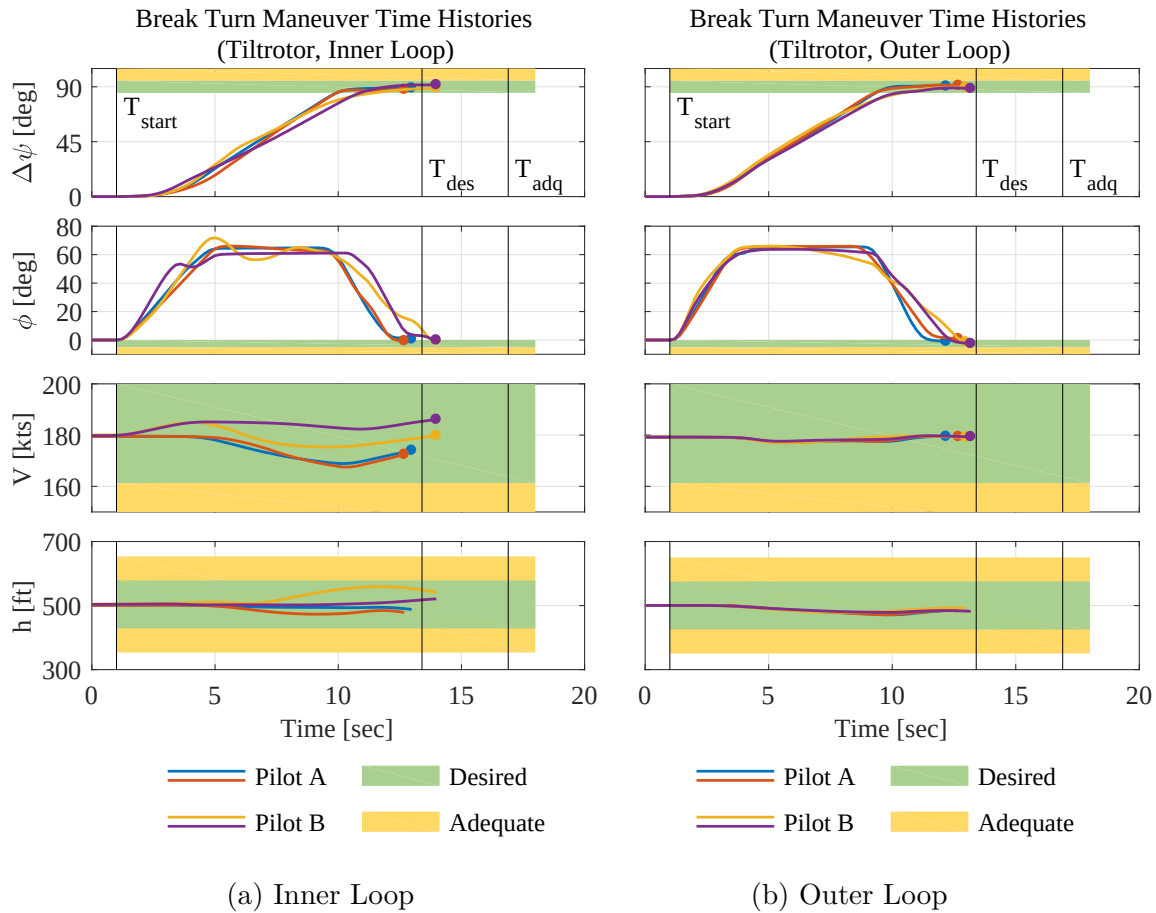


Figure 6.22. Break Turn task performance for (a) inner-loop control laws and (b) outer-loop control laws (tiltrotor, 180 kts).

Pilot A rated the Break Turn MTE with the inner-loop control laws an HQR 4 and BWLR 7. He commented that he was “not guaranteed to make desired” when flying this task with the inner-loop control laws. He noted that he could be aggressive

in roll and pitch, but that being precise was hard. Pilot A commented that the critical axis during this maneuver was pitch (trying to control vertical rate during the turn) even though the roll axis was also very active. In general, the bank angle was “a lot messier” with the inner-loop control laws because the focus was so much on pitch.

With the outer-loop control laws, Pilot A rated the Break Turn MTE an HQR 2 and a BWLR 3. He commented that the task was “much easier” and essentially became a single axis task where only rolling into and out of the turn had to be commanded by the pilot.

Pilot B rated the Break Turn MTE with the inner-loop control laws an HQR 4.5 (due to his times to complete being beyond the desired performance criteria). He commented that the aircraft response was predictable, but that when the aircraft was banked (during the steady portion of the turn), it was very pitch sensitivity.

With the outer-loop control laws, Pilot B rated the Break Turn MTE with the tiltrotor an HQR 2. He commented that it became “easy to meet desired [performance]” and that he was “monitoring the pitch axis rather than interacting with it.”

Figure 6.23 shows the pilot stick activity (cutoff frequency ω_{co} and RMS) for all axes during the Break Turn maneuver for both inner- and outer-loop control laws. The error bars represent the average, maximum, and minimum values for all of the runs. For axes with RMS = 0, ω_{co} was not calculated. Neither pilot had to use the pedals during the turn with either set of control laws. Pilot B did slightly increase the collective before the turn when flying with the inner-loop control laws to help maintain airspeed, although no collective input was necessary when flying with the outer-loop control laws. Pilots had comparable ω_{co} and stick RMS values for the lateral stick for both control laws. The main difference is for the longitudinal stick, which had comparable values to the lateral stick RMS for the inner-loop control laws but was nearly 0 for the outer-loop control laws. These results demonstrate the excellent airspeed and altitude hold capabilities of the outer-loop control laws.

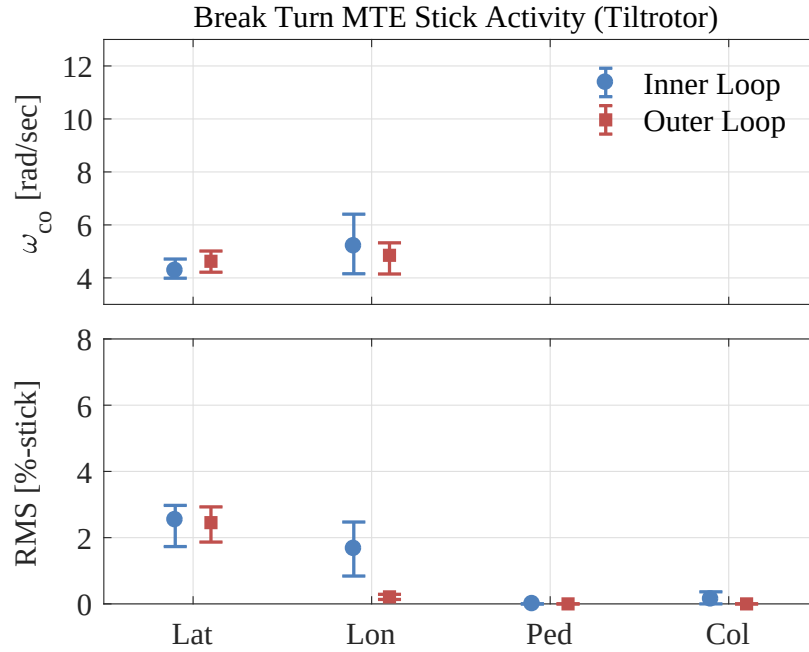


Figure 6.23. Break Turn stick activity (tiltrotor, 180 kts).

6.3.2 High-Speed Acceleration/Deceleration

Figure 6.24(a) shows the High-Speed Acceleration MTE performance of the tiltrotor inner-loop control laws for all data runs for both pilots, plotted against the desired and adequate bounds listed in Table A.3. Both pilots were able to meet desired performance.

Figure 6.24(b) shows the High-Speed Acceleration MTE performance of the tiltrotor outer-loop control laws for all data runs for both pilots. Both pilots were also able to meet desired performance with the outer-loops engaged, with significantly less variability in altitude and pitch attitude.

Pilot A rated the High-Speed Acceleration MTE with the tiltrotor inner-loop control laws an HQR 4 and BWLR 7. He commented that altitude was the most difficult parameter to control during this task. This was because at the beginning of the task, the tiltrotor is controlled like a conventional helicopter, with pitch attitude being used to command acceleration [see negative pitch attitudes during first part of maneuver, Figure 6.24(a), bottom subplot]. Eventually, as airspeed increases and the wing becomes effective, pitch attitude has to be increased in order to have a positive angle of attack on the wing. Having to manage pitch attitude, collective, and nacelle angle during this task led to a high workload rating. Pilot A did note that near the end of the maneuver, in airplane mode, altitude was well behaved and he was only concentrating on using collective to manage airspeed.

Pilot A rated the High-Speed Acceleration MTE with the tiltrotor outer-loop

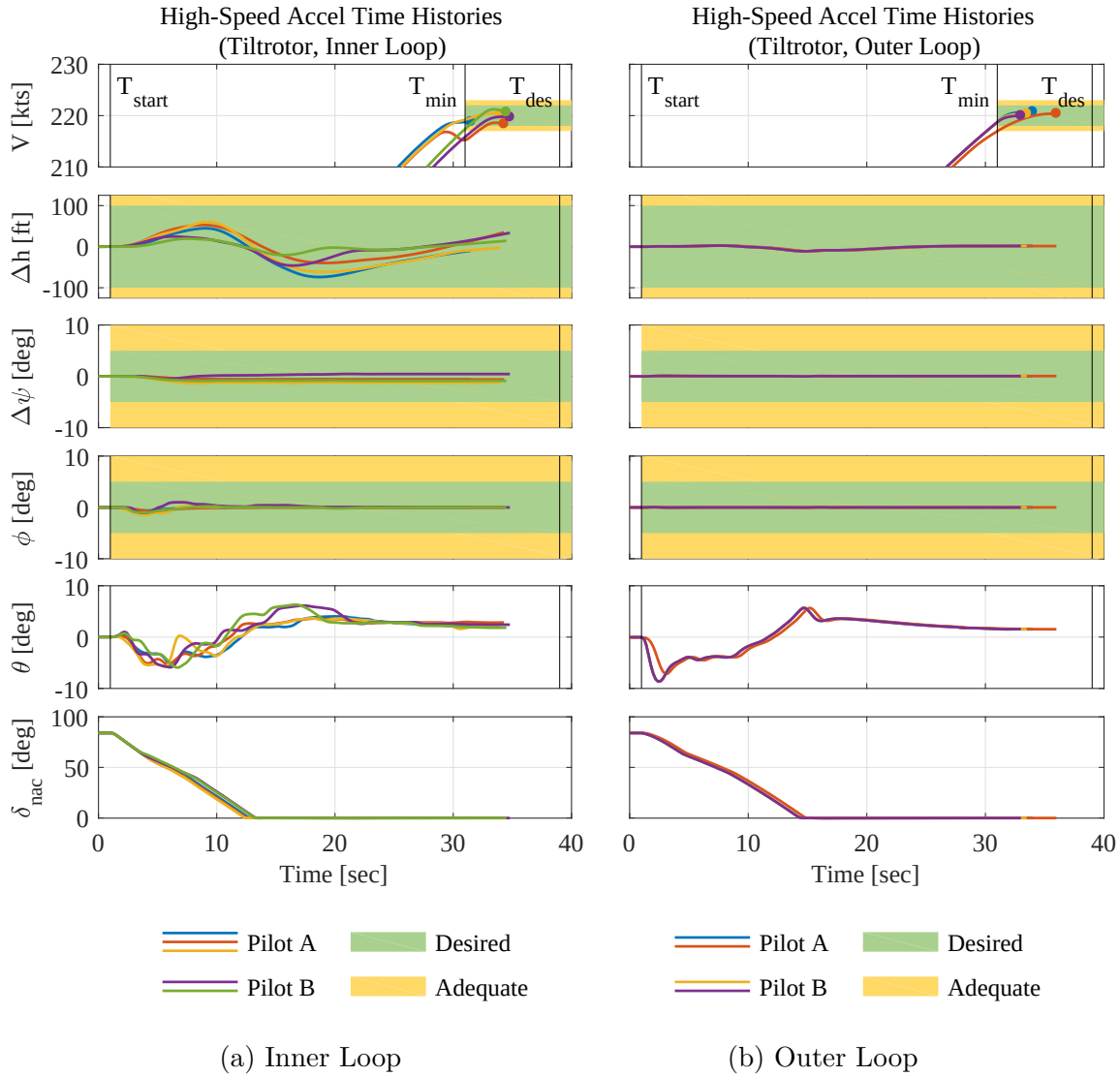


Figure 6.24. High-Speed Acceleration task performance for (a) inner-loop control laws and (b) outer-loop control laws (right) (tiltrotor, 180 kts).

control laws an HQR 2 and BWLR 3. He attempted the task using both the collective stick and thumb hat switch to control acceleration and was able to easily meet desired with both. Pilot A commented that precision was “great” and that the task did not require any control compensation. The only reason Pilot A rated the task an HQR 2 instead of HQR 1 was because he had to hold the thumb hat switch/collective up the entire run (as opposed to being able to set a final airspeed and having the outer-loop automatically go there).

Pilot B rated the High-Speed Acceleration MTE with the tiltrotor inner-loop control laws an HQR 4. He noted that to meet desired performance, he had to be

very aggressive with power and nacelle and so the highest workload became managing altitude with the pitch axis. With the outer-loop control laws, Pilot B rated the task an HQR 1. He said the task was “easy” and only required holding the thumb hat switch down, with no control inputs required in pitch.

Figure 6.25 shows the pilot stick activity (cutoff frequency ω_{co} and RMS) for longitudinal stick and collective inputs during the High-Speed Acceleration MTE for both inner- and outer-loop control laws. Both pilots had similar collective RMS and cutoff frequency ω_{co} between the inner- and outer-loop control laws, however neither pilot had to touch the longitudinal stick with the outer-loop control laws (RMS = 0), while still maintaining minimal altitude deviations throughout the task. This demonstrated the good altitude hold capabilities of the outer-loop control laws.

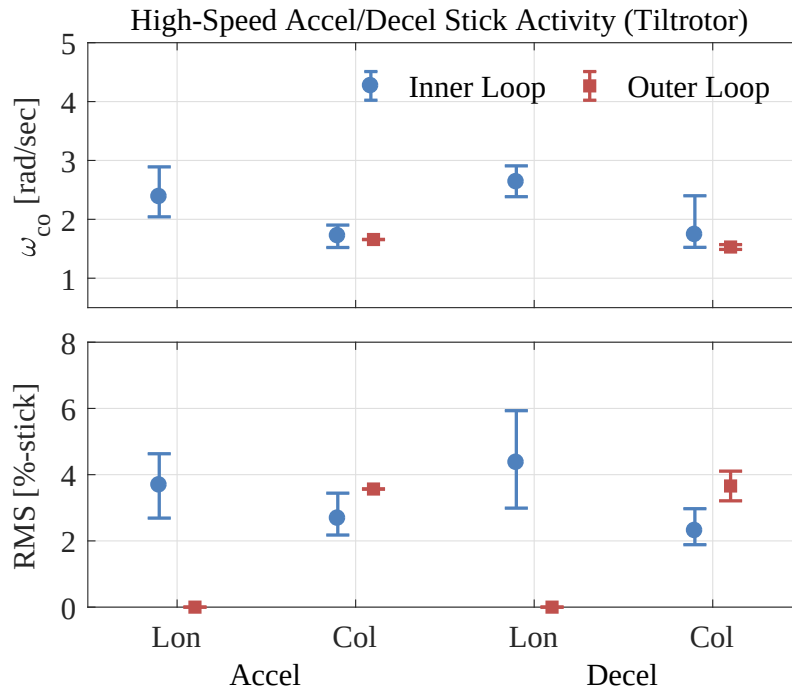


Figure 6.25. High-Speed Acceleration/Deceleration stick activity (tiltrotor).

Figure 6.26(a) shows the High-Speed Deceleration MTE performance of the tiltrotor inner-loop control laws for all data runs for both pilots, plotted against the desired and adequate bounds listed in Table A.3. Both pilots were able to meet desired performance, with Pilot B having one run that was completed faster than the analytically determined minimum time to complete T_{min} , due to him going to a higher pitch attitude during the maneuver.

Figure 6.26(b) shows the High-Speed Deceleration MTE performance of the tiltrotor outer-loop control laws for all data runs for both pilots. Both pilots were able to meet desired performance with the outer-loops engaged, with significantly less variability in altitude and pitch attitude.

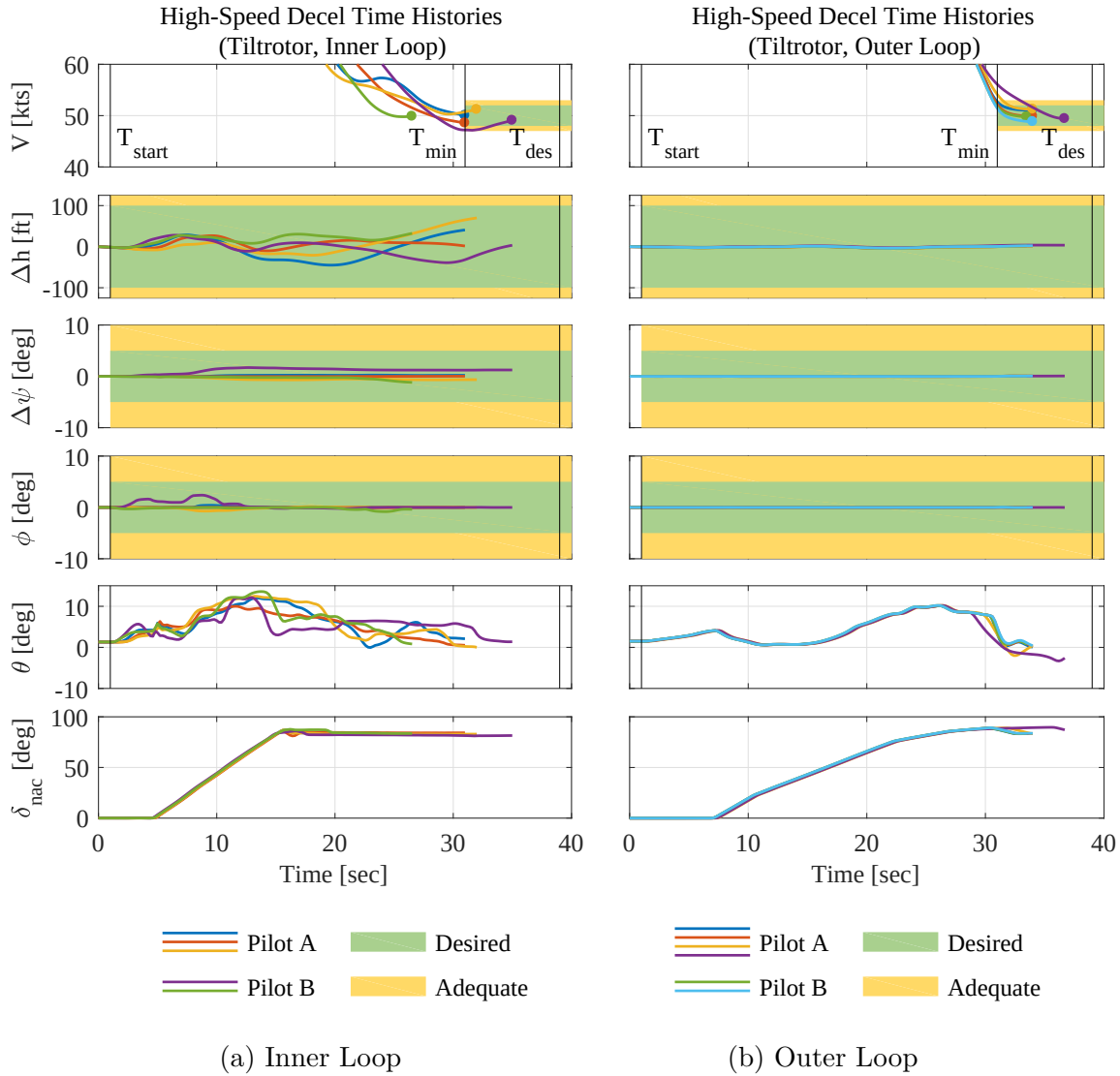


Figure 6.26. High-Speed Deceleration task performance for (a) inner-loop control laws and (b) outer-loop control laws (tiltrotor, 180 kts).

Pilot A rated the High-Speed Deceleration MTE with the tiltrotor inner-loop control laws an HQR 4 and BWLR 7. He commented that out of all of the maneuvers that he flew in this simulation experiment, this was the “busiest” one and that he could not individually isolate the controls. He had to use longitudinal stick, collective, and nacelle controls all at the same time. Although Pilot A noted that he was able to make desired performance and that there was “plenty of time” to complete the maneuver, he rated it an HQR 4 because of the difficult altitude maintenance.

With the outer-loop control laws, Pilot A rated the High-Speed Deceleration MTE an HQR 2 and BWLR 3. Pilot A attempted the deceleration with both collective and

thumb hat switch and noted that it was easy to make desired performance with both. He did note that precision with the thumb switch was more difficult (because it only commands maximum deceleration) and timing the release took a few practice runs. However, the collective made it easier to be precise.

Pilot B rated the High-Speed Deceleration MTE with the tiltrotor inner-loop control laws an HQR 4.5, noting the high workload in the pitch axis to maintain altitude during the task. Pilot B rated the MTE an HQR 1 with the outer-loop control laws, and commented that altitude control was “very tight” and that the task became just a “single button press.”

The stick activity for the High-Speed Deceleration MTE is shown in Figure 6.25 and is similar to the High-Speed Acceleration MTE. Both pilots had similar collective RMS and cutoff frequency ω_{co} between the inner- and outer-loop control laws, however neither pilot had to touch the longitudinal stick with the outer-loop control laws (RMS = 0), while still maintaining minimal altitude deviations throughout the task. Again, this demonstrated the good altitude hold capabilities of the outer-loop control laws.

6.3.3 Pitch Angle Capture and Hold

Figure 6.27 shows two example record runs for the Pitch Angle Capture and Hold MTE, one with the inner-loop control law and one with the outer-loop control laws. The two runs shown are representative of the additional record runs collected. Pilots were able to meet desired performance with the inner-loop control laws, but had some excursions into adequate performance with the outer-loop control laws, which drove the Level 2 HQRs for the outer loops.

Pilot A rated the inner-loop control laws an HQR 2 for the Pitch Angle Capture and Hold MTE. He commented that the response was “heavily/too damped” and that the aircraft felt like it had a lot of inertia. Pilot A noted that the response was predictable and easy to stop, although the slow initial acceleration caused him to lead his input to get a faster response.

For the outer-loop control laws, Pilot A rated the Pitch Angle Capture and Hold MTE an HQR 5. He commented that it was “hard to be precise” with this configuration since the magnitude of the overshoot was too big. This corresponds to the large value for pitch attitude dropback for the outer-loop control laws (Figure 5.37, last row, third subplot) as compared to the inner-loop control laws (Figure 3.27, last row, fourth subplot).

Pilot B only tested the inner-loop control laws and gave an HQR 2. He commented that the speed of the responses was good, and that the response was “predictable” and “well behaved.”

Figure 6.28 shows the pilot longitudinal stick activity for the Pitch Angle Capture and Hold MTE for both inner- and outer-loop control laws. With the outer-loop control laws, the pilot cutoff frequency ω_{co} was significantly increased as the pilots had to put in additional inputs to account for the large pitch attitude dropback

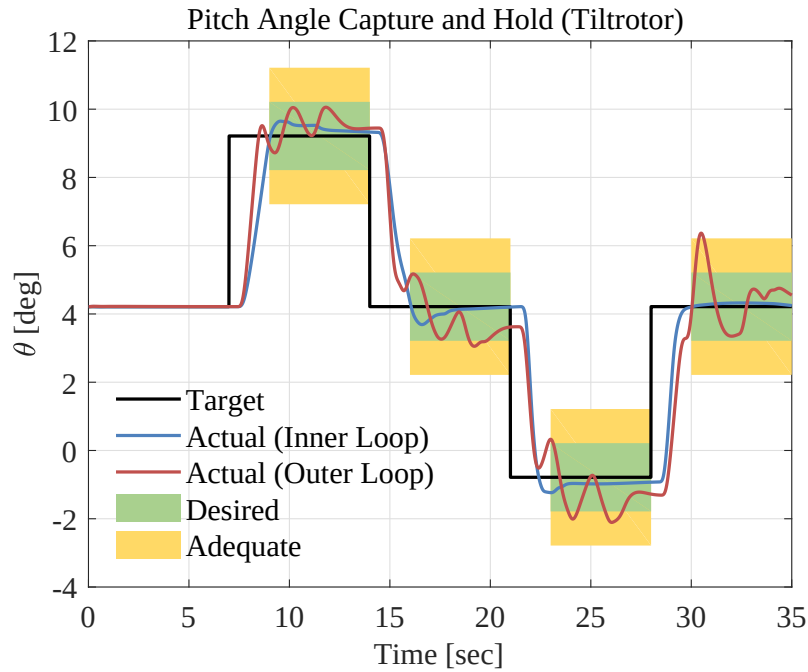


Figure 6.27. Pitch Angle Capture and Hold example time histories (tiltrotor, 180 kts).

characteristic. This correlates with the degradation to Level 2 HQRs for the outer loops (Figure 6.21). The stick RMS is slightly less for the outer-loop control laws due to the larger initial pitch acceleration.

These results suggest that the pitch attitude dropback value for this configuration, which is in Level 1, is too high and that the specification boundary may need to be reduced.

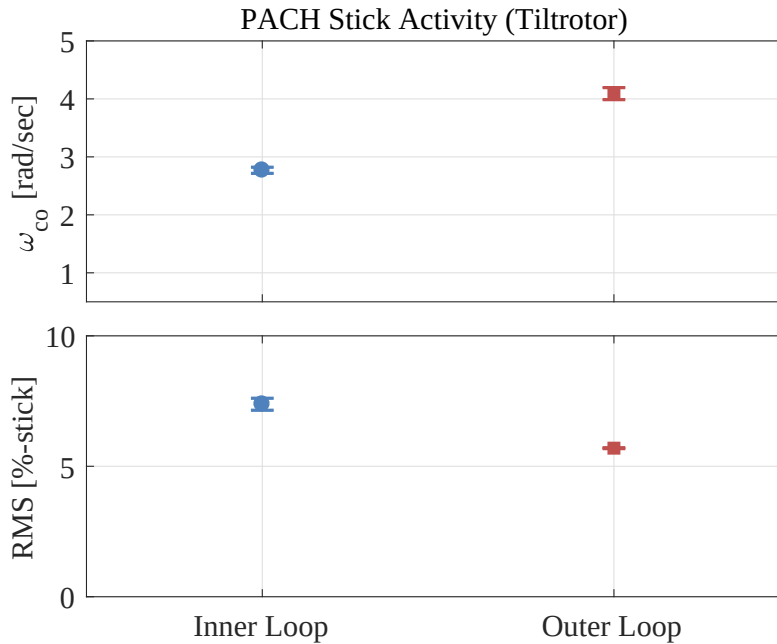


Figure 6.28. Pitch Angle Capture and Hold stick activity (tiltrotor, 180 kts).

6.3.4 Pitch Sum-of-Sines Tracking

Figure 6.29 shows the performance for the Pitch Sum-of-Sines Tracking MTE for the tiltrotor. Both pilots were able to meet desired performance with both sets of control laws, although performance for the inner-loop control laws was significantly better for both pilots.

Pilot A rated the Pitch Sum-of-Sines Tracking MTE an HQR 2 for the inner-loop control laws. He commented that he had “no problem with precision,” was “not having to compensate,” and “definitely liked this one.” He also noted, though, that for the inner loops, although small pitch motions were “easy/nice,” large pitch motions were “hard.” This correlates well with Pilot A’s comments for the Pitch Attitude Capture and Hold task that the inner-loop control laws were too damped.

For the outer-loop control laws, Pilot A rated this task an HQR 4. He commented that he “could still get desired [performance],” but was required to “use a counter-move to get aircraft to stop.” The pitch attitude dropback was seen as uncommanded pitch motion by the pilot, which he referred to as an “annoying deficiency.”

Pilot B rated the Pitch Sum-of-Sines Tracking MTE an HQR 3 for the inner-loop control laws. He commented that he could “easily meet desired [performance]” and was “able to be aggressive.” Pilot B also noted that although the response was well damped, he “did not want quite as much [damping].”

For the outer-loop control laws, Pilot B rated this task an HQR 4. He noted that the responsiveness of the outer-loop control laws was better, but with the penalty of

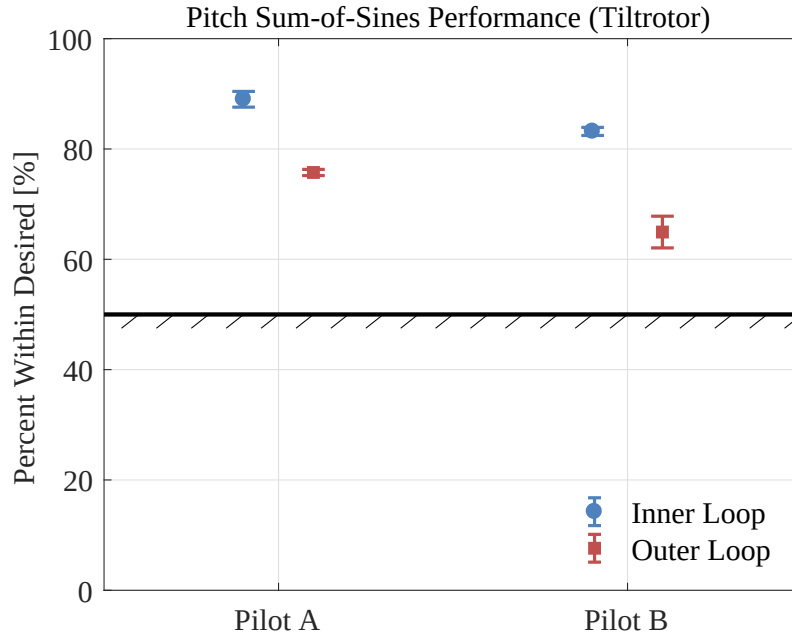


Figure 6.29. Pitch Sum-of-Sines Tracking performance (tiltrotor, 180 kts).

overshoots. He also commented that although he was able to meet desired performance, his stress level went up with the outer-loop control laws as he was constantly making inputs and always fixing the over-/under-shoot.

As with the coaxial-pusher, a quantitative comparison of the two control laws was done by investigating the characteristics of the pilot Y_p vehicle Y_c system (PVS). Figure 6.30 shows the aircraft frequency response $Y_c = \theta/\delta_{\text{lon}}$ extracted from the time history data using CIPHER[®], with discrete sum-of-sines frequencies highlighted. Here, the outer-loop control laws have significantly more pitch attitude dropback than the inner-loop control laws, as discussed in Sec. 5.6 and can be seen as the large hump in the magnitude curve between $\omega = 1 - 10$ rad/sec.

Figure 6.31 shows the PVS broken-loop frequency response ($Y_p Y_c$) extracted from the time history data at the discrete frequencies used in the tracking signal using CIPHER[®], as well as a crossover model [108] fit to the each response. The parameters of the crossover model are given in Table 6.3 for both inner- and outer-loop control laws.

Unlike the coaxial-pusher, where the differences in aircraft response between the inner- and outer-loop control laws were smaller, and the pilot was able to adjust to maintain performance and PVS characteristics, for the tiltrotor, the pilot had to decrease his overall gain and cutoff frequency. This led to a reduction in task performance with the outer-loop control laws as well as a reduction in the PVS crossover frequency ω_c and degradation in HQR (Figure 6.21).

Figure 6.32 shows the pilot frequency response $Y_p = \delta_{\text{lon}}/\theta_{\text{error}}$, and shows the

overall reduced gain the pilot had to use with the outer-loop control laws.

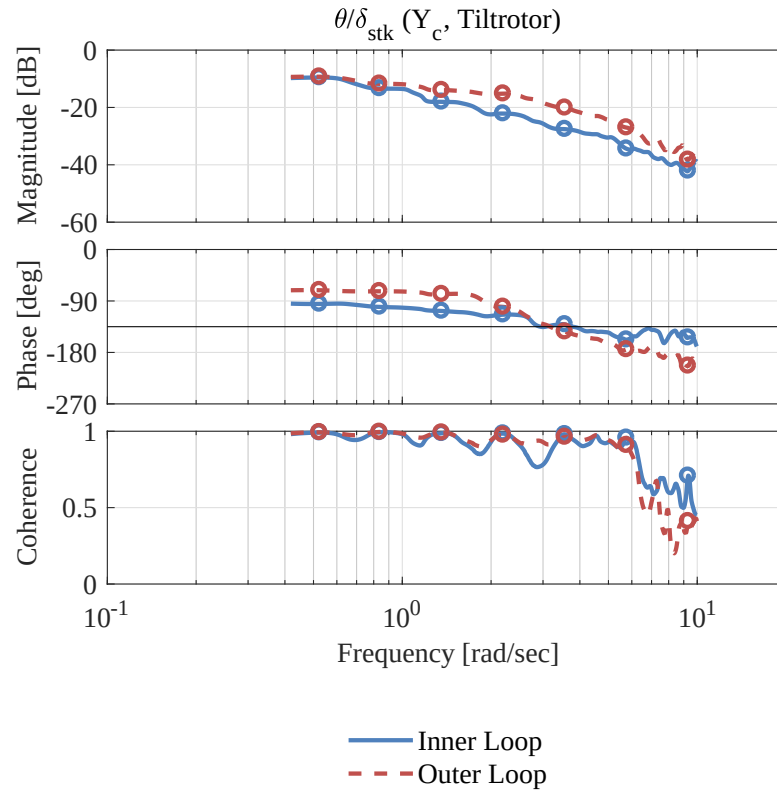


Figure 6.30. Vehicle frequency responses comparison for Pitch Sum-of-Sines Tracking task (tiltrotor, 180 kts).

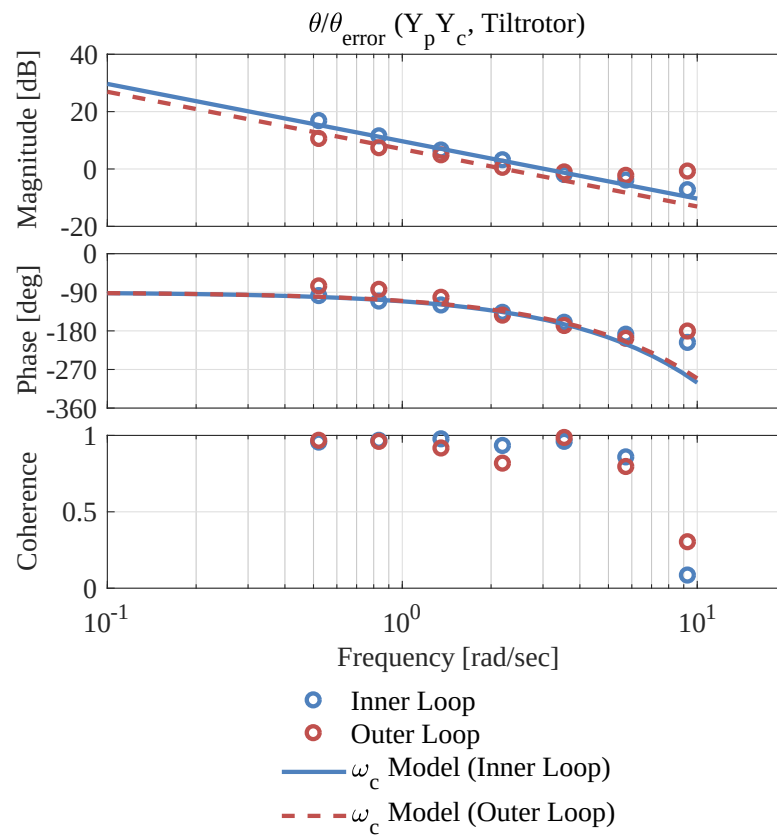


Figure 6.31. Pilot-Vehicle System broken-loop frequency responses comparison for Pitch Sum-of-Sines Tracking task (tiltrotor, 180 kts).

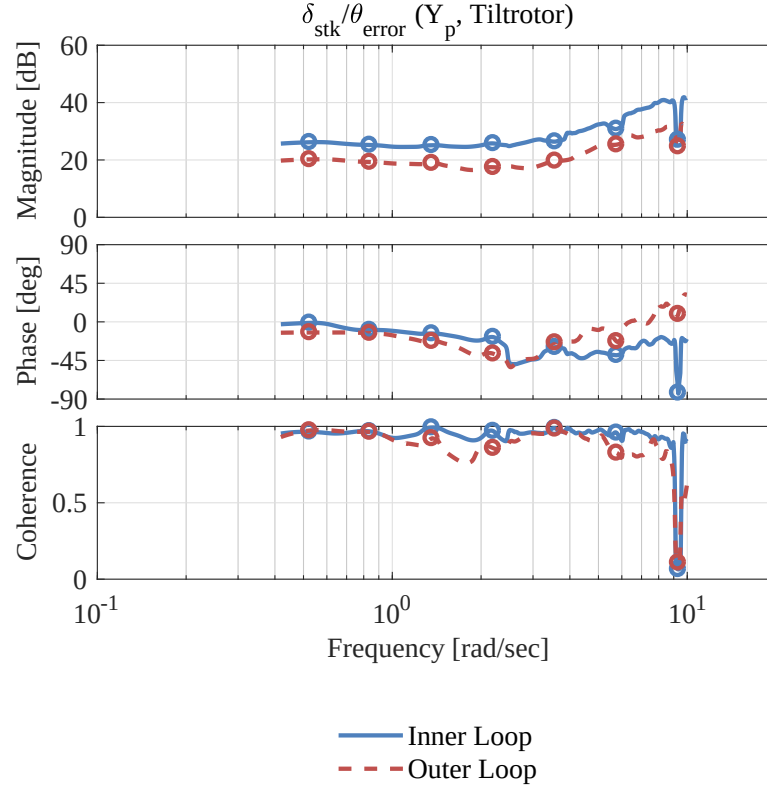


Figure 6.32. Pilot frequency responses comparison for Pitch Sum-of-Sines Tracking task (tiltrotor, 180 kts).

Table 6.3. Pilot-Vehicle Parameters for Pitch Sum-of-Sines Tracking Task (Tiltrotor, 180 kts)

	Inner Loop	Outer Loop
<i>Pilot-Vehicle System Parameters ($Y_p Y_c$)</i>		
Pilot-Vehicle Crossover Frequency ω_c [rad/sec]	3.04	2.22
Gain Margin [dB]	2.93	6.11
Phase Margin [deg]	25.83	45.43
Pilot-Vehicle Equivalent Time Delay τ [sec]	0.37	0.35
<i>Pilot Parameters (Y_p)</i>		
Pilot Gain K_p	21.29	10.65
Time Delay τ_p [sec]	0.13	0.12
Cutoff Frequency ω_{co} [rad/sec]	1.45	1.10
Stick RMS [%]	13.76	9.45

6.3.5 Formation Flying

Figure 6.33 shows the Formation Flying task time histories for two of Pilot A’s runs with the tiltrotor—one with the inner-loop control laws and one with the outer-loop control laws. Pilot A was able to get into the formation position more quickly with the outer-loop control laws. Both airspeed and climb rate traces are much smoother and less oscillatory with the outer-loop control laws than with the inner-loop control laws. Pilot A commented that the task was “a lot easier” with the outer-loop control laws as compared to the inner-loop control laws.

Table 6.4 lists the lateral and longitudinal stick activity during this task. In both axes, the stick RMS was reduced for the outer-loop control laws and the cutoff frequency was higher. This is because with the outer-loop control laws, the pilot tended to make fewer, small inputs, both in terms of magnitude and duration. This correlates well with Pilot A’s comment that flying this task with the outer-loops was easier.

Table 6.4. Formation Flying Stick Activity (Tiltrotor)

	Inner Loop	Outer Loop
<i>Lateral Stick</i>		
Cutoff Frequency ω_{co} [rad/sec]	1.17	3.75
Stick RMS [%]	3.64	2.58
<i>Longitudinal Stick</i>		
Cutoff Frequency ω_{co} [rad/sec]	0.815	3.05
Stick RMS [%]	3.51	2.56

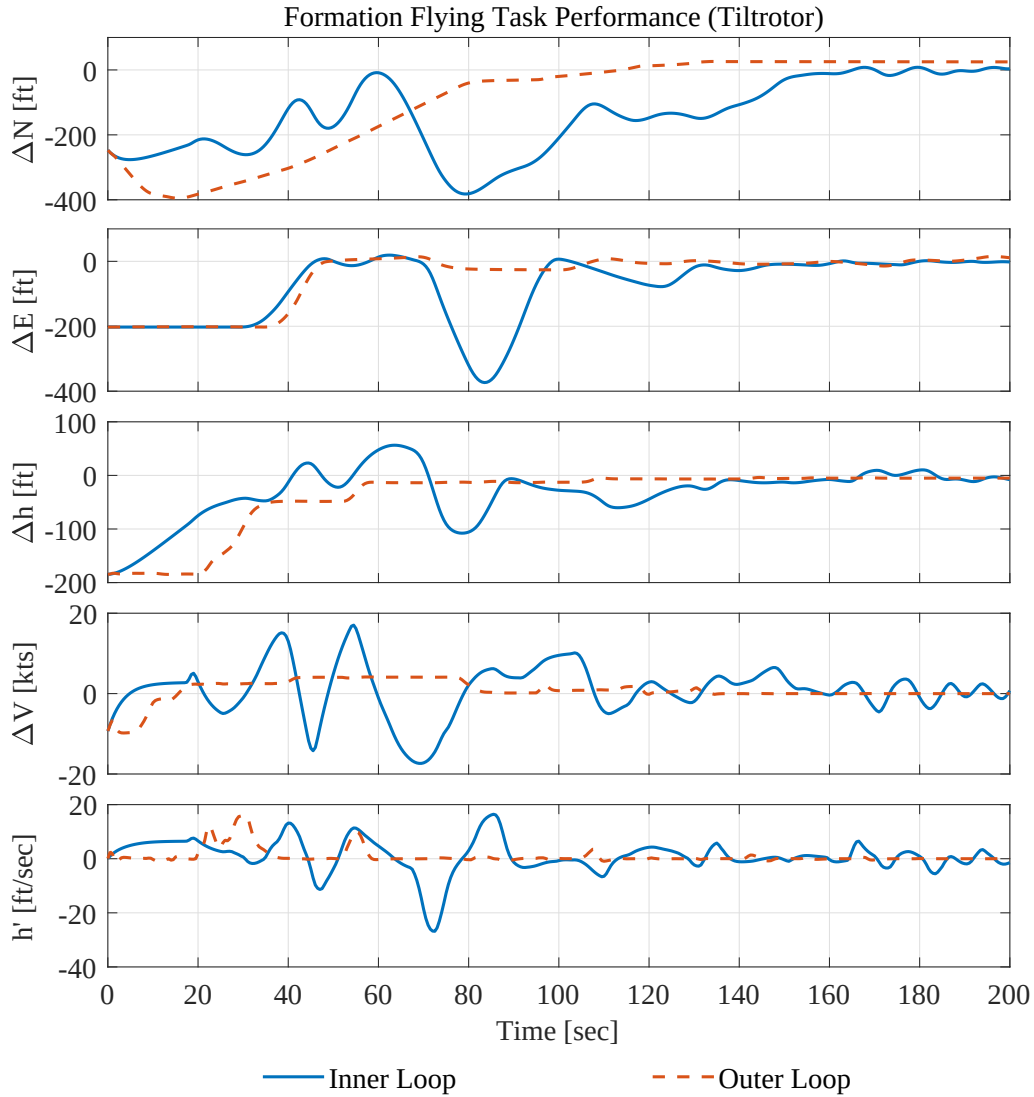


Figure 6.33. Formation flying task performance (tiltrotor).

6.4 Discussion

Overall, differences in Handling Qualities Ratings (both positive and negative) between the inner- and outer-loop control laws were larger for the tiltrotor than the coaxial-pusher for two main reasons. First, the coaxial-pusher has an independent thrust control (the pusher propeller) and the inner-loop control laws contain a collective trim map. This makes the High-Speed Acceleration/Deceleration MTEs with the inner-loop control laws a single, decoupled axis task with Level 1 HQRs, and the outer-loop control laws were not a significant improvement over this. The coaxial-

pusher inner-loop control laws also received average HQR 3.5 ratings for the Break Turn MTE. The outer-loop control laws improved to a solid Level 1 rating of average HQR 3 for the Break Turn MTE.

However, in the case of the tiltrotor with inner-loop control laws, the High-Speed Acceleration/Deceleration MTEs were a very dynamic and coupled task, where the pilot had to manage pitch attitude, collective, and nacelle angle manually. This resulted in high workload (BWL 7, which correlates to workload not tolerable for the task) and Level 2 HQRs for the inner-loop control laws. In contrast, the outer-loop control laws made the High-Speed Acceleration/Deceleration MTEs a single, decoupled axis task, where now the pilot only had to command acceleration or deceleration directly, and the control system managed pitch attitude, collective, and nacelle angle. This resulted in a significant improvement in HQRs (average HQR 1.5) and in workload (BWL 3, which correlates to workload satisfactory without reduction). In addition, the tiltrotor had a bigger improvement in HQRs for the Break Turn MTE, going from solid Level 2 handling qualities for the inner-loop control laws (average HQR 4.25) to solid Level 1 (average HQR 2).

The second reason for a larger difference in HQRs between the inner- and outer-loop control laws for the tiltrotor is the larger difference in pitch attitude dropback between the two control laws for the tiltrotor (as discussed in Sec. 5.6). This resulted in the handling qualities of the tiltrotor degrading more than the coaxial-pusher for the Pitch Sum-of-Sines Tracking and Pitch Attitude Capture and Hold MTEs. In the case of the coaxial-pusher, there was a half point degradation in handling qualities for the Pitch Sum-of-Sines Tracking MTE and a full point degradation for the Pitch Attitude Capture and Hold MTEs with the outer-loop control laws. For the tiltrotor, with the its large value of pitch attitude dropback for the outer-loop control laws, there was 1.5 point degradation in handling qualities for the Pitch Sum-of-Sines Tracking MTE and a three point degradation for the Pitch Attitude Capture and Hold MTE with the outer-loop control laws.

The outer-loop control laws for both coaxial-pusher and tiltrotor have Level 1 pitch attitude dropback at the flight condition tested ($V_{CAS} = 180$ kts), however the pitch tracking tasks received borderline Level 1/Level 2 ratings for the coaxial-pusher and solid Level 2 ratings for the tiltrotor. In addition, most of the pilot comments about these configurations were about the dropback characteristics, confirming that dropback drove the HQRs. This suggests that the dropback requirement should be included as a Tier 1 specification in the control law design and enforced with a more stringent boundary (as discussed further in Chapter 7). Initial results suggest the Level 1/Level 2 boundary should correspond to the coaxial-pusher level of dropback (since it received average HQR 3.5), but a more thorough analysis will be done in the next chapter.

6.5 Summary

This chapter described the handling qualities assessment of outer-loop flight control systems for both the generic lift offset coaxial and tiltrotor rotorcraft. A piloted simulation experiment was conducted in the PSU Flight Simulator facility with participation from two Army experimental test pilots. Five high-speed handling-qualities demonstration maneuvers were used: Pitch Attitude Capture and Hold, Pitch Sum-of-Sines Tracking, Break Turn, and High-Speed Acceleration and Deceleration. Overall, the outer-loop control laws for both coaxial-pusher and tiltrotor aircraft were assigned Level 1 handling qualities for the Break Turn and High-Speed Acceleration/Deceleration tasks, and improved pilot workload over the inner-loop control laws. The outer-loop control laws also improved performance and reduced pilot workload in a formation flying task developed for this experiment. The coaxial-pusher outer loops received borderline Level 1/Level 2 ratings for the Pitch Attitude Capture and Hold and Sum-of-Sines Tracking tasks, while the tiltrotor outer loops (with their increased value of pitch attitude dropback) received Level 2 ratings.

Chapter 7 |

Pitch Axis Response Type Investigation

This chapter describes the results of a piloted handling qualities simulation experiment that was carried out at the Pennsylvania State University (PSU) Flight Simulator facility to test different variations of the pitch-axis command model for the inner-loop control laws described in Chapter 3. The evaluation consisted of two handling qualities demonstration maneuvers (Pitch Sum-of-Sines Tracking and Pitch Attitude Capture and Hold) flown by two Army experimental test pilots.

The chapter begins with a brief review of high-speed forward flight pitch-axis dynamics. Then the variations in the pitch-axis command model will be described. Results for the simulation evaluation will be provided for the two tasks. Finally, a discussion of the results will be provided.

7.1 Review of Forward Flight Pitch Dynamics

At high-speed forward flight, the dynamics of both coaxial-pusher and tiltrotor are similar to those of a typical fixed-wing aircraft (Chapter 2). In the mid-frequency range (short term response), the bare-airframe pitch rate q response to longitudinal input δ_{lon} is well represented by a first-over-second-order transfer function and time delay:

$$\frac{q}{\delta_{\text{lon}}} = \frac{M_{\delta_{\text{lon}}}(s + 1/T_{\theta_2})e^{-\tau_{\theta}s}}{s^2 + 2\zeta_{\text{sp}}\omega_{\text{sp}}s + \omega_{\text{sp}}^2} \quad (7.1)$$

while the bare-airframe normal acceleration n_z response to longitudinal input δ_{lon} is well represented by a zeroth-over-second-order transfer function and time delay:

$$\frac{n_z}{\delta_{\text{lon}}} = \frac{n_{z\delta_{\text{lon}}}e^{-\tau_{n_z}s}}{s^2 + 2\zeta_{\text{sp}}\omega_{\text{sp}}s + \omega_{\text{sp}}^2} \quad (7.2)$$

Note that the typical nonminimum phase zero in the normal acceleration transfer

function (due to the initial loss of lift when the elevator is deflected trailing edge up) is dropped here because $Z_{\delta_{\text{lon}}} \approx 0$ due to the control allocation. Therefore the instantaneous center of rotation (ICR) is located close to the CG.

From Eqs. 7.1 and 7.2, it is clear that the normal acceleration response lags behind the pitch rate response by:

$$\frac{n_z}{q} = \frac{n/\alpha}{(s + 1/T_{\theta_2})} \quad (7.3)$$

where the gain n/α is given by [95]:

$$\frac{n}{\alpha} = \frac{V}{gT_{\theta_2}} \quad (7.4)$$

The dynamic relationship in Eq. 7.3 also describes the flight path response to pitch attitude:

$$\frac{\gamma}{\theta} = \frac{1/T_{\theta_2}}{(s + 1/T_{\theta_2})} \quad (7.5)$$

and hence T_{θ_2} is referred to as the flight path-attitude lag.

Figure 7.1 shows an example pitch attitude step response for the dynamics in Eqs. 7.1 and 7.2. The flight path response γ can be seen to lag behind the pitch attitude response θ by the flight path-attitude lag T_{θ_2} . The flight path lag T_γ shown on the figure is given by:

$$T_\gamma = \frac{2\zeta_{\text{sp}}}{\omega_{\text{sp}}} \quad (7.6)$$

The pitch rate and attitude responses can be seen to have some overshoot before settling to their respective steady-state values, due to the zero ($1/T_{\theta_2}$) in the numerator of the pitch rate transfer function (Eq. 7.1). The definition of the classical fixed-wing parameter pitch attitude dropback DB_θ [113] is shown in the figure, and its value relative to the steady state pitch rate q_{ss} is given by [17]

$$\frac{\text{DB}_\theta}{q_{\text{ss}}} = \left(T_{\theta_2} - \frac{2\zeta_{\text{sp}}}{\omega_{\text{sp}}} \right) \quad (7.7)$$

Note that in Eq. 7.7, the dropback is referenced to the pitch attitude when the stick δ_{lon} input is removed, and not the peak pitch attitude value, as is typically used in the dropback specification [17].

Another classical fixed-wing handling qualities parameter is the Control Anticipation Parameter (CAP), which is defined as the ratio of the initial pitch acceleration to steady state normal acceleration:

$$\text{CAP} = \frac{\ddot{\theta}(0)}{n_{z_{\text{ss}}}} \quad (7.8)$$

From Eqs. 7.1–7.4, it can be seen that:

$$\text{CAP} = \frac{M_{\delta_{\text{lon}}}}{n_{z\delta_{\text{lon}}}/\omega_{\text{sp}}^2} = \frac{\omega_{\text{sp}}^2}{\frac{V}{g} \frac{1}{T_{\theta_2}}} \quad (7.9)$$

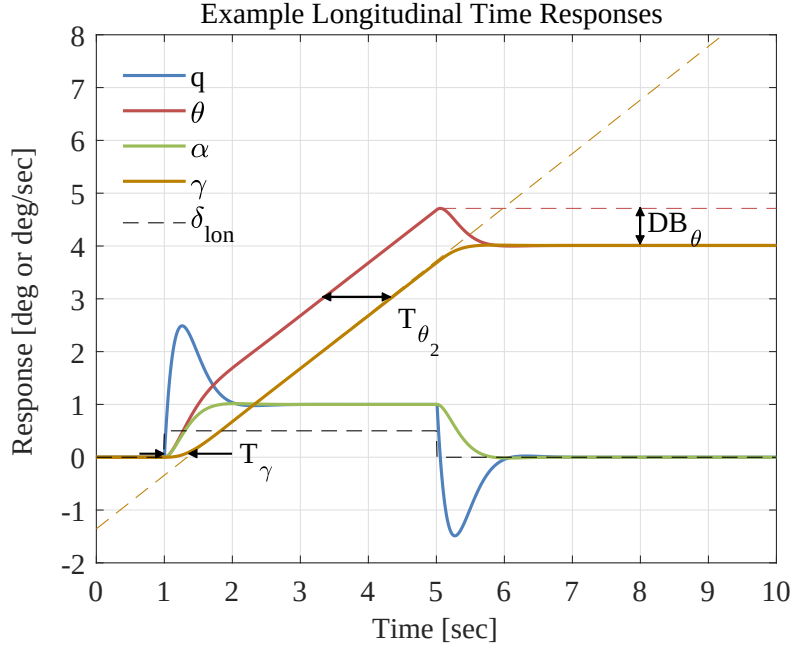


Figure 7.1. Example pitch attitude step response time history showing definition of flight path-attitude lag T_{θ_2} , flight path lag T_γ , and pitch attitude dropback DB_θ .

Recall from Sec. 5.6 that T_{θ_2} is a function of the bare-airframe and cannot be changed with feedback to a single control input. Therefore, given a value of T_{θ_2} , any attempt to reduce flight path lag T_γ (or equivalently increase flight path bandwidth $\omega_{\text{BW}_\gamma}$) will result in increased pitch attitude dropback.

7.2 Pitch Axis Command Model

As discussed in Sec. 3.1.2, the nominal inner-loop pitch axis command model for the Rate-Command/Attitude-Hold (RCAH) response type is a first-order system given by:

$$\frac{q_{\text{cm}}}{\delta_{\text{lon s}}} = \frac{K_{\text{lon cm}}}{\tau_{\text{lon cm}} s + 1} \quad (7.10)$$

Assuming good model following, the closed-loop aircraft pitch rate and normal

acceleration responses are given by:

$$\begin{aligned}\frac{q}{\delta_{\text{lon}_s}} &= \frac{M_{\delta_{\text{lon}}}(s + 1/T_{\theta_2})e^{-\tau_{\theta}s}}{(s + 1/\tau_{\text{lon}_{\text{cm}}})(s + 1/T_{\theta_2})} \\ \frac{n_z}{\delta_{\text{lon}_s}} &= \frac{Z_{\delta_{\text{lon}}}e^{-\tau_{n_z}s}}{(s + 1/\tau_{\text{lon}_{\text{cm}}})(s + 1/T_{\theta_2})}\end{aligned}\quad (7.11)$$

In this case, the flight path lag is given by:

$$T_{\gamma} = \tau_{\text{lon}_{\text{cm}}} + T_{\theta_2} \quad (7.12)$$

and the pitch attitude dropback is given by combining Eqs. 7.6, 7.7, and 7.12 as:

$$\frac{\text{DB}_{\theta}}{q_{ss}} = -\tau_{\text{lon}_{\text{cm}}} \quad (7.13)$$

Again, note that dropback here is referenced to the pitch attitude when the stick input δ_{lon_s} is removed, and not the peak pitch attitude value, which results in negative values of dropback being possible (as shown in Figure 7.2). These negative values of dropback are associated with the pitch attitude continuing to drift after the stick input is removed. Gibson [113] notes that using this definition of dropback, negative values are associated with sluggish responses in flight path control and tracking. This is consistent with comments made by pilots that the RCAH responses were “too damped” (see Secs. 4.2.3, 4.3.3, 6.3.3, 6.3.4).

The amount of negative dropback can be reduced (by decreasing the command model time constant $\tau_{\text{lon}_{\text{cm}}}$) but not eliminated with a first-order command model. Therefore, an investigation was done using a higher-order pitch rate command model, given by:

$$\frac{q_{\text{cm}}}{\delta_{\text{lon}_s}} = \frac{K_{\text{lon}_{\text{cm}}}(s + 1/T_{\theta_{\text{cm}}})}{s^2 + 2\zeta_{\text{lon}_{\text{cm}}}\omega_{\text{lon}_{\text{cm}}}s + \omega_{\text{lon}_{\text{cm}}}^2} \quad (7.14)$$

Assuming good model following, the closed-loop aircraft pitch rate and normal acceleration responses using the pitch rate command model in Eq. 7.14 are given by:

$$\begin{aligned}\frac{q}{\delta_{\text{lon}}} &= \frac{M_{\delta_{\text{lon}}}(s + 1/T_{\theta_{\text{cm}}})e^{-\tau_{\theta}s}}{s^2 + 2\zeta_{\text{lon}_{\text{cm}}}\omega_{\text{lon}_{\text{cm}}}s + \omega_{\text{lon}_{\text{cm}}}^2} \\ \frac{n_z}{\delta_{\text{lon}}} &= \frac{Z_{\delta_{\text{lon}}}(s + 1/T_{\theta_{\text{cm}}})e^{-\tau_{n_z}s}}{(s^2 + 2\zeta_{\text{lon}_{\text{cm}}}\omega_{\text{lon}_{\text{cm}}}s + \omega_{\text{lon}_{\text{cm}}}^2)(s + 1/T_{\theta_2})}\end{aligned}\quad (7.15)$$

Note that here, the normal acceleration response is first-over-third-order, which can result in undesirable “g creep” [114] if $T_{\theta_{\text{cm}}}$ is set too far away from the bare-airframe T_{θ_2} .

Two pitch RCAH families were developed to be tested. The first uses a first-order pitch command model (Eq. 7.10) while the second used a first-over-second order

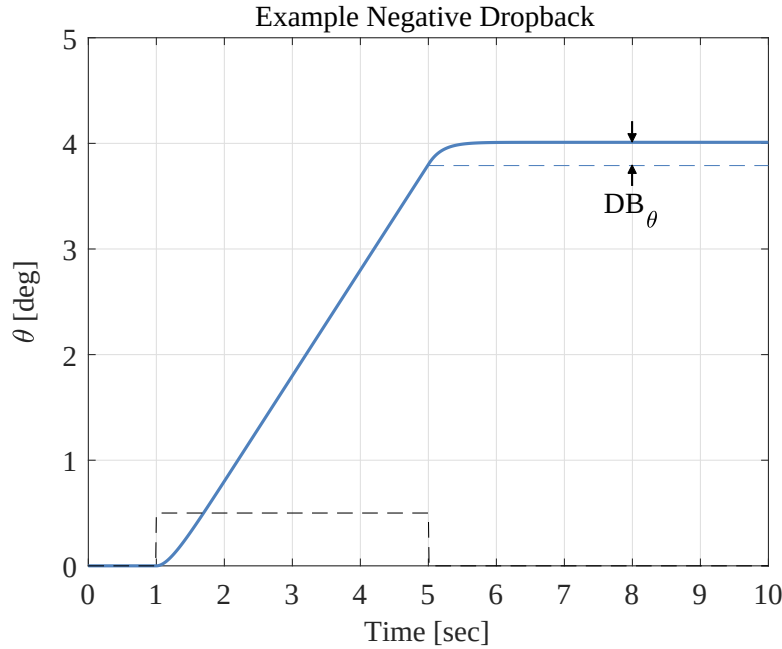


Figure 7.2. Example pitch attitude step response time history showing definition of negative pitch attitude dropback DB_θ .

command model (Eq. 7.14). Within each family of designs, the parameters of the command model were varied to span the handling qualities criteria. The following sections describe the configurations for the coaxial-pusher and tiltrotor.

7.2.1 Coaxial-Pusher Configurations

The parameters of the coaxial-pusher command models investigated (Families 1 and 2) are shown in Table 7.1. In addition, the parameters of the normal acceleration command model used above $V_{CAS} = 200$ kts (Family 3, tested in Chapter 4) and outer-loop flight path rate command model (Family 4, tested in Chapter 6) are given in Table 7.1. Figures 7.3 through 7.6 show where the designs lie on several of the handling qualities specifications.

Family 1 was designed to span a range of pitch attitude bandwidth values (Figure 7.3), with Configuration 1-C matching the baseline inner-loop pitch rate command model (evaluated in Chapter 4). The designs in Family 1 also span a range of flight path bandwidth values as seen in Figure 7.4. Since Family 1 uses a first-order pitch rate command model, all of the designs have negative dropback using the definition in Eq. 7.7, and zero dropback using the MIL-STD-1797B definition and shown in Figure 7.5. Figure 7.6 shows the Control Anticipation Parameter (CAP) versus equivalent short period frequency ζ_{sp} . As seen in Eq. 7.15, the first-order pitch rate command model used in Family 1 results in equivalent $\zeta_{sp} > 1$.

Family 2 was designed using the first-over-second order pitch rate command model

(Eq. 7.14). The parameters of the designs in Family 2 were chosen to have the same value of pitch attitude phase bandwidth as Configuration C in Family 1, but varying levels of flight path bandwidth (and therefore pitch attitude dropback). The way the parameters were calculated is given in Appendix C. Figure 7.3 shows that the Family 2 designs all have the same pitch attitude bandwidth as Configuration 1-C, as designed. Figure 7.4 shows the increased flight path bandwidth of Configuration 2-B as compared to 2-A, and Figure 7.5 shows the associated increased dropback of Configuration 2-B.

Table 7.1 also lists the pitch command model gain K_{loncm} . Note that for the normal acceleration command design (Family 3), simulator evaluations were performed at an airspeed of $V_{\text{CAS}} = 220$ kts, and so the equivalent pitch rate per stick displacement was lower than the other designs (to maintain the same 2.5 g command limit).

Table 7.1. Pitch Command Model Configurations (Coaxial-Pusher)

Family	Response Type	Config.	$1/\tau_{\text{loncm}}$ [1/sec]	ω_{loncm} [rad/sec]	ζ_{loncm} [-]	$1/T_{\theta\text{cm}}$ [1/sec]	K_{loncm} [deg/sec/%]
1	q RCAH	A	1.5	—	—	—	0.36
		B	3	—	—	—	0.36
		C	5	—	—	—	0.36
		D	10	—	—	—	0.36
2	q RCAH	A	—	2.43	1	2	0.36
		B	—	3.16	1	1	0.36
3	n_z -cmd	-	—	3	1	—	0.25*
4	$\dot{\gamma}$ -cmd	-	1.8	—	—	—	0.36

* Equivalent pitch rate command per stick displacement based on normal acceleration command per stick displacement evaluated at $V_{\text{CAS}} = 200$ kts

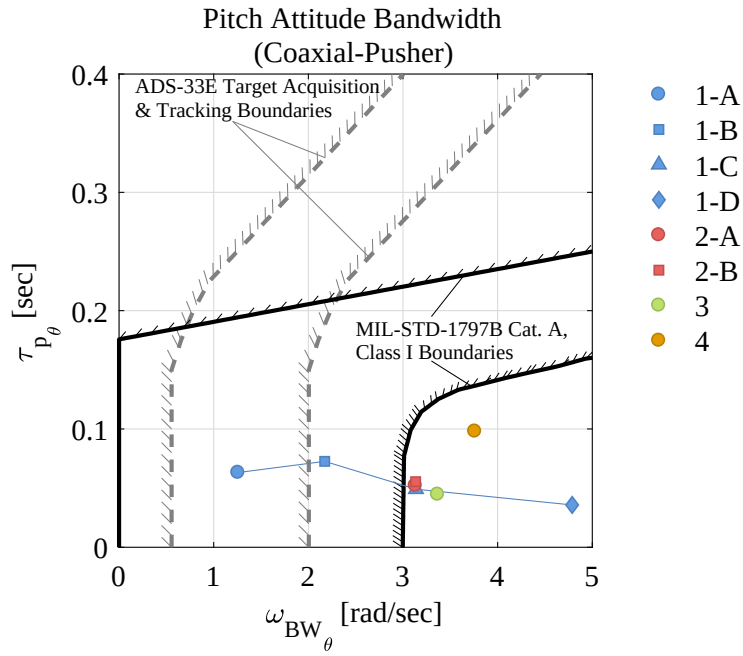


Figure 7.3. Pitch attitude bandwidth values for command model variations (coaxial-pusher).

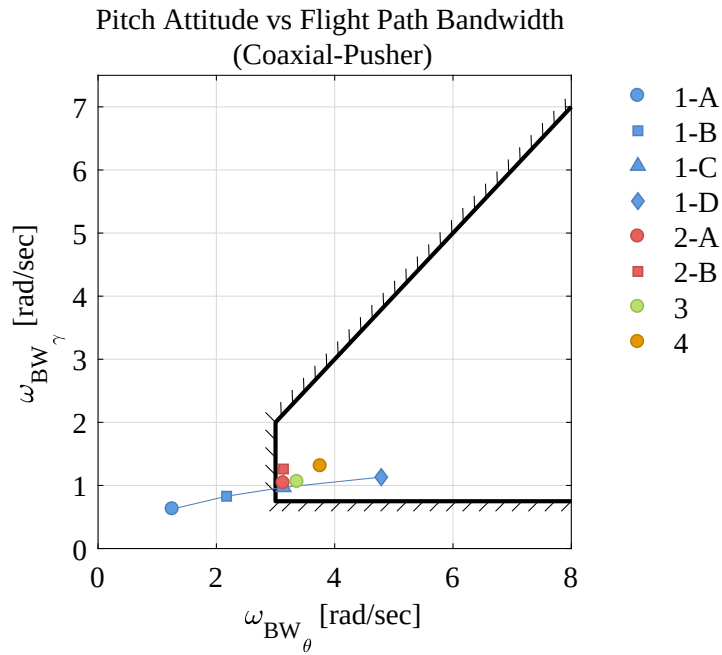


Figure 7.4. Flight path versus pitch attitude bandwidth values for command model variations (coaxial-pusher).

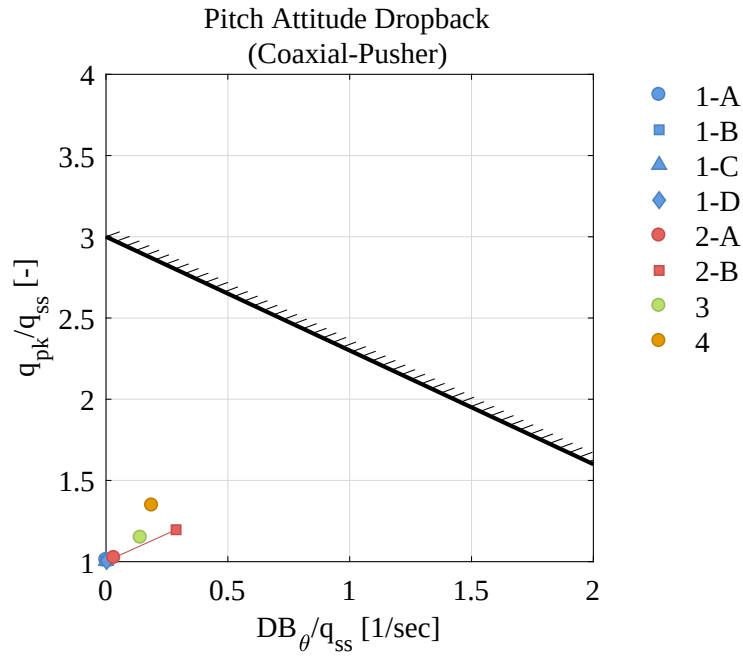


Figure 7.5. Pitch attitude dropback values for command model variations (coaxial-pusher).

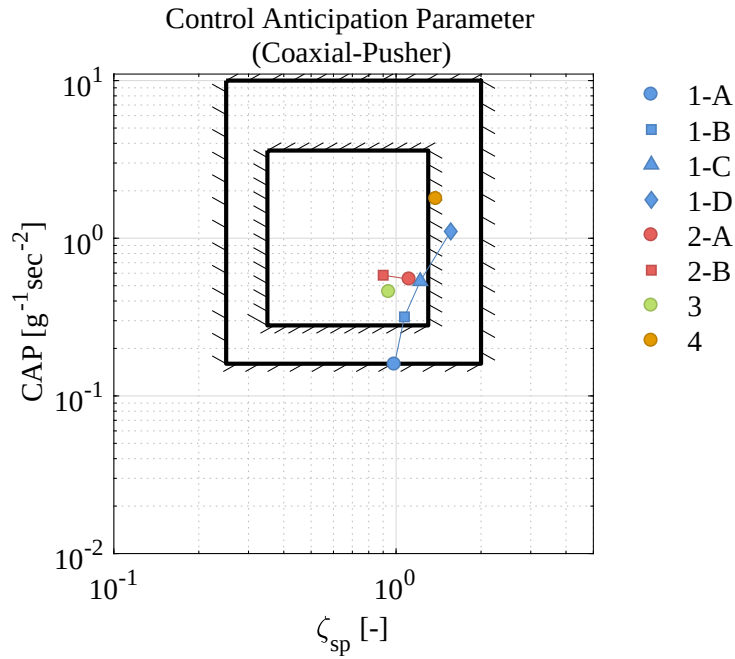


Figure 7.6. Control anticipation parameters values for command model variations (coaxial-pusher).

7.2.2 Tiltrotor Configurations

The parameters of the tiltrotor command model investigated (Families 1 and 2) are shown in Table 7.2. In addition, the parameters of the normal acceleration command model used above $V_{CAS} = 200$ kts (Family 3, tested in Chapter 4) and outer-loop flight path rate command model (Family 4, tested in Chapter 6) are given in Table 7.1. Figures 7.7 through 7.10 show where the designs lie on several of the handling qualities specifications.

The tiltrotor command models were tuned similarly to the coaxial-pusher, as described above. For similar values of flight path bandwidth ω_{BW_γ} , the tiltrotor designs have significantly higher values of pitch attitude dropback. This is due to the tiltrotor's larger inherent flight path-attitude lag T_{θ_2} as discussed in Sec. 5.6.

Table 7.2 also lists the pitch command model gain $K_{lon_{cm}}$. Note that the tiltrotor command model gains are slightly lower than the coaxial-pusher values (Table 7.1). In addition, as with the coaxial-pusher, the normal acceleration command design (Family 3) was evaluated at an airspeed of $V_{CAS} = 220$ kts, and so the equivalent pitch rate per stick displacement was lower than the other designs.

Table 7.2. Pitch Command Model Configurations (Tiltrotor)

Family	Response Type	Config.	$1/\tau_{lon_{cm}}$ [1/sec]	$\omega_{lon_{cm}}$ [rad/sec]	$\zeta_{lon_{cm}}$ [—]	$1/T_{\theta_{cm}}$ [1/sec]	$K_{lon_{cm}}$ [deg/sec/%]
1	q RCAH	A	1.5	—	—	—	0.3
		B	3	—	—	—	0.3
		C	5	—	—	—	0.3
		D	10	—	—	—	0.3
2	q RCAH	A	—	3.21	1	2	0.3
		B	—	2.53	1	1	0.3
		C	—	2.33	1	0.7	0.3
3	n_z -cmd	—	—	3	1	—	0.25*
4	$\dot{\gamma}$ -cmd	—	3.9	—	—	—	0.3

* Equivalent pitch rate command per stick displacement based on normal acceleration command per stick displacement evaluated at $V_{CAS} = 200$ kts

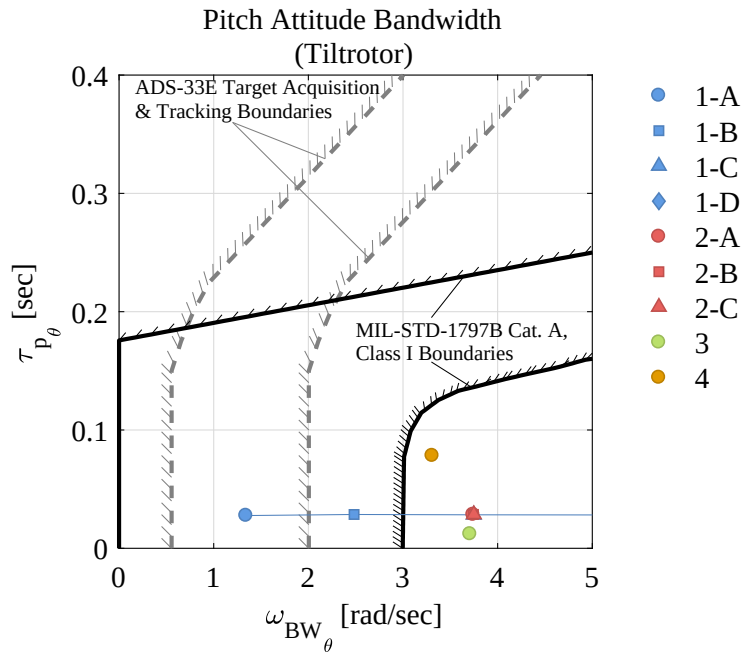


Figure 7.7. Pitch attitude bandwidth values for command model variations (tiltrotor).

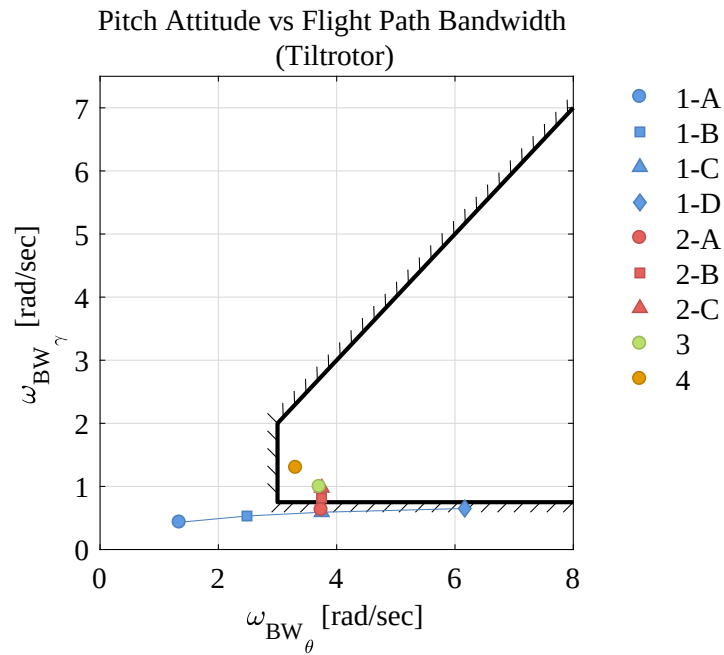


Figure 7.8. Flight path versus pitch attitude bandwidth values for command model variations (tiltrotor).

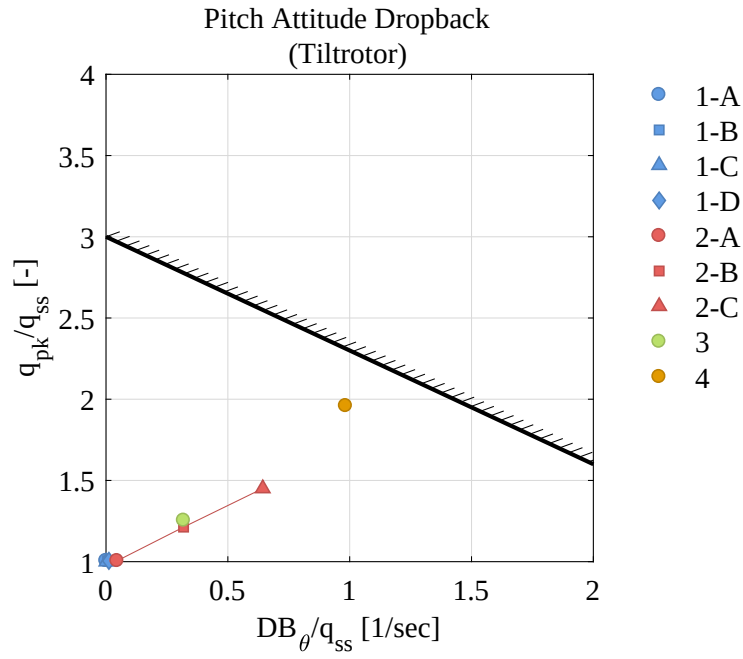


Figure 7.9. Pitch attitude dropback values for command model variations (tiltrotor).

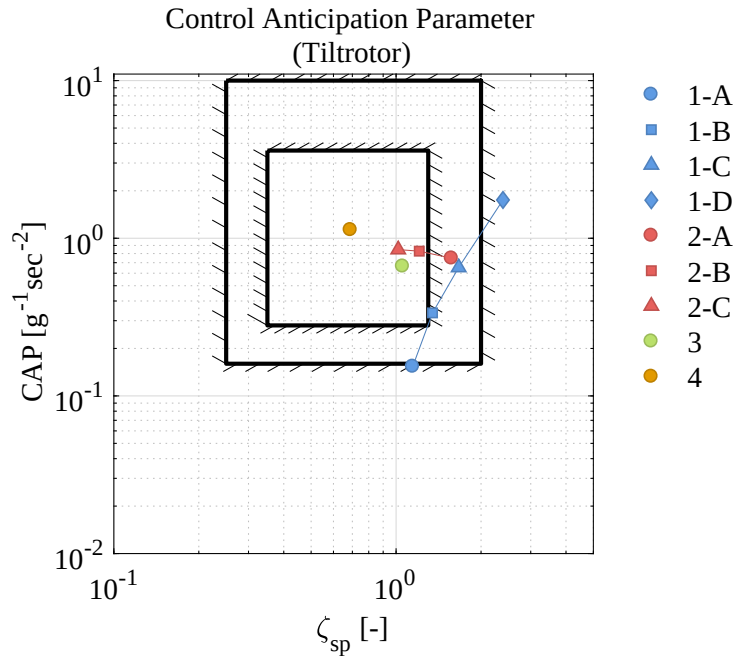


Figure 7.10. Control anticipation parameters values for command model variations (tiltrotor).

7.3 Experimental Setup

The designs described above were tested in the Pennsylvania State University (PSU) Simulator facility. Evaluations were conducted by two U.S. Army experimental test pilots using two high-speed handling qualities demonstration maneuvers—Pitch Sum-of-Sines Tracking and Pitch Attitude Capture and Hold. The simulator, tasks, and conduct of the test are described in more detail in Sec. 6.1.

7.4 Handling Qualities Results

Table 7.3 shows the average Cooper-Harper Handling Qualities Ratings (HQRs) for the coaxial-pusher and tiltrotor configurations tested. The following sections show the results on several of the key specifications, first for the Pitch Sum-of-Sines Tracking task and then for the Pitch Attitude Capture and Hold task.

Table 7.3. Pitch Configurations Handling Qualities Ratings

Family	Response Type	Configuration	Pitch SOS	Pitch ACH
<i>Coaxial-Pusher</i>				
1	q RCAH	A	4	—
		B	3	—
		C	3.25	2.63
		D	3	2
2	q RCAH	A	—	4
		B	3.5	4
3	n_z command	—	3	3
4	$\dot{\gamma}$ command	-	3.5	3.5
<i>Tiltrotor</i>				
1	q RCAH	A	4.25	4
		B	3	3
		C	3	2.25
		D	—	2
2	q RCAH	A	—	2
		B	2	4
		C	3	4.5
3	n_z command	—	2.5	3
4	$\dot{\gamma}$ command	-	4	5

7.4.1 Pitch Sum-of-Sines Tracking

Figures 7.11 through 7.14 show the Pitch Sum-of-Sines Tracking task average HQRs on the pitch attitude bandwidth, flight path bandwidth, pitch attitude dropback, and CAP specifications. As expected and can be seen in Figure 7.11, cases with low pitch attitude bandwidth ω_{BW_θ} have Level 2 HQRs. The boundary between Level 1 and Level 2 ratings appears to match well with the ADS-33E specification boundary plotted in Figure 7.11 (dashed gray line). In addition, meeting the bandwidth requirement does not guarantee Level 1 ratings. Cases that meet the pitch attitude bandwidth, flight path bandwidth, and CAP requirements but still have Level 2 ratings can be seen to have a large amount of pitch attitude dropback (Figure 7.13).

Figure 7.15 shows the average HQRs for the Pitch Sum-of-Sines Tracking task plotted versus the pitch attitude dropback DB_θ/q_{ss} . Here, only the cases that meet the flight path bandwidth requirement in Figure 7.12 are plotted (i.e., cases where $\omega_{BW_\theta} \geq 3$ rad/sec and $\omega_{BW_\gamma} \geq 0.75$ rad/sec). Trend lines for the results of each aircraft are also plotted on the figure. The trend lines show that for the coaxial-pusher, $DB_\theta/q_{ss} \approx 0.3$ sec⁻¹ corresponds to the Level 1/Level 2 boundary, while for the tiltrotor, pilots were willing to tolerate additional dropback and $DB_\theta/q_{ss} \approx 0.8$ sec⁻¹ corresponds to the Level 1/Level 2 boundary. This may be due to the tiltrotor's smaller stick gain $K_{lon_{cm}} = 0.3$ deg/sec/% as compared to the coaxial-pusher's $K_{lon_{cm}} = 0.36$ deg/sec/%.

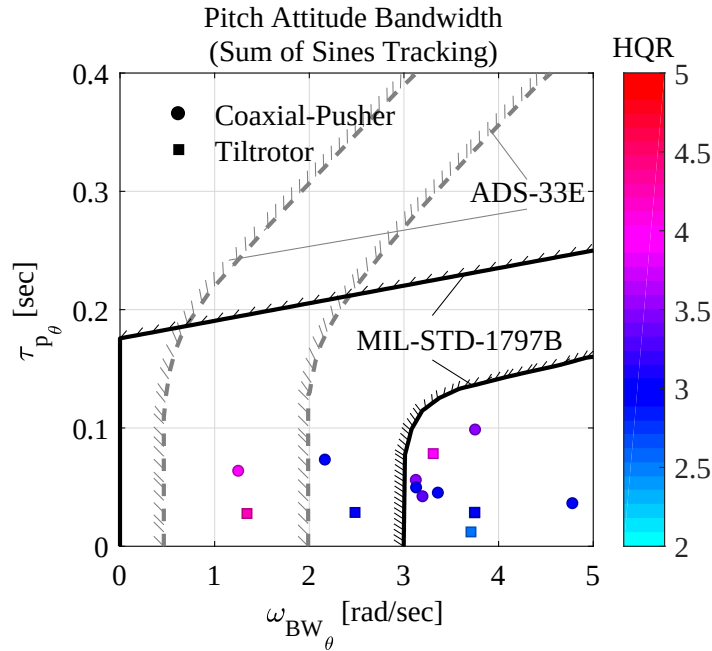


Figure 7.11. Pitch Sum-of-Sines Tracking handling qualities ratings plotted on the pitch attitude bandwidth requirement (coaxial-pusher and tiltrotor).

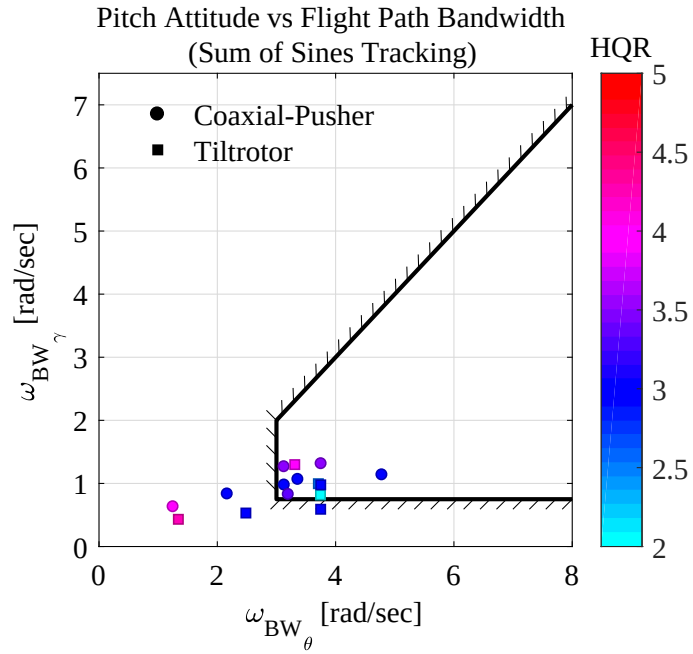


Figure 7.12. Pitch Sum-of-Sines Tracking handling qualities ratings plotted on the pitch attitude versus flight path bandwidth requirement (coaxial-pusher and tiltrotor).

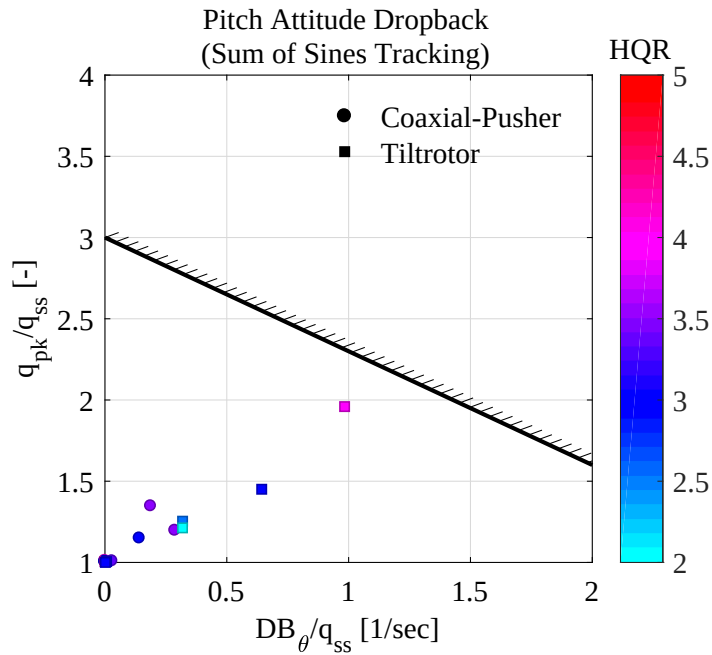


Figure 7.13. Pitch Sum-of-Sines Tracking handling qualities ratings plotted on the pitch attitude dropback requirement (coaxial-pusher and tiltrotor).

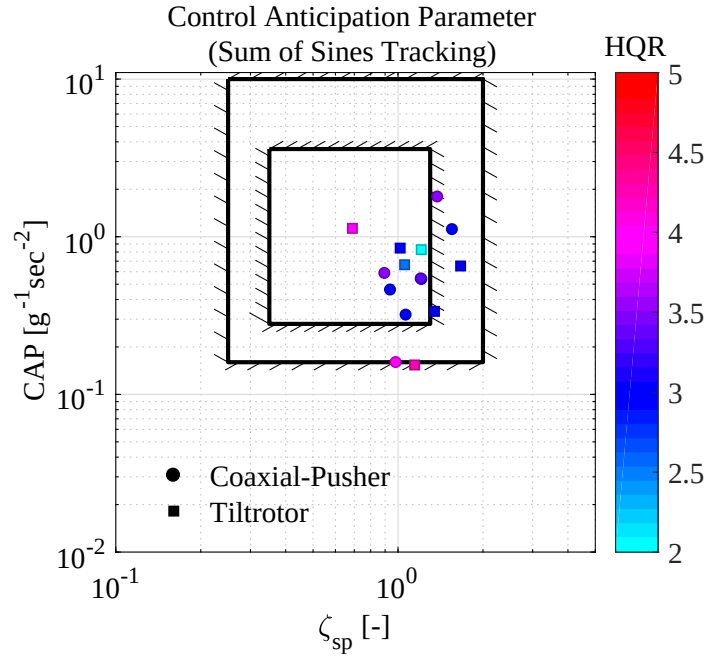


Figure 7.14. Pitch Sum-of-Sines Tracking handling qualities ratings plotted on the Control Anticipation Parameter (CAP) requirement (coaxial-pusher and tiltrotor).

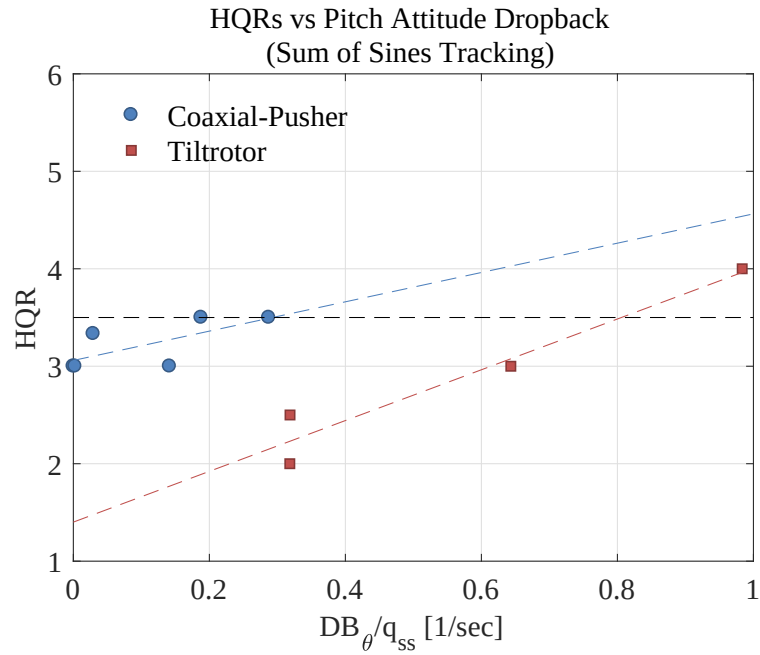


Figure 7.15. Pitch Sum-of-Sines Tracking handling qualities ratings plotted as a function of pitch attitude dropback (coaxial-pusher and tiltrotor)

Pilot comments for coaxial-pusher Configuration 1-A (first order command model, low pitch attitude bandwidth) included “bad,” “closest to PIO that I have gotten,” “tendency to overshoot,” and “very heavily damped.” For the tiltrotor Configuration 1-A pilot comments were similar and included “very sluggish,” “seems to coast, which causes overshoot,” and “need to lead input.”

For the coaxial-pusher Configuration 1-B, pilots commented that although it was not bad, “the most objectionable part [was] the amount of coast down after taking input out” (i.e., large negative dropback). For the tiltrotor Configuration 1-B, pilots commented that the response was sluggish and there was a potential to overshoot.

Comments about tiltrotor Configuration 1-C (baseline first order command model configuration) include “not a lot of overshoots” and “response well damped, but did not want quite as much [damping].”

Comments about coaxial-pusher Configuration 1-D (first order command model, high pitch attitude bandwidth) include “minimal overshoots, settles quickly,” “compensation quantitatively decreased,” and “by far the best one.”

Comments about coaxial-pusher Configuration 2-B include “allows to be more aggressive since I can get faster rates, but needs more inputs for counter correction,” “more overshooting”, and “working harder.”

7.4.2 Pitch Attitude Capture and Hold

Figures 7.16 through 7.19 show the Pitch Attitude Capture and Hold task average HQRs on the pitch attitude bandwidth, flight path bandwidth, pitch attitude dropback, and CAP specifications. As expected and can be seen in Figure 7.16, the case with low pitch attitude bandwidth ($\omega_{BW_\theta} < 2$ rad/sec) received Level 2 HQRs. The boundary between Level 1 and Level 2 ratings appears to match well with the ADS-33E specification boundary plotted in Figure 7.16 (dashed gray line). In addition, meeting the bandwidth requirement does not guarantee Level 1 ratings. Cases that meet the pitch attitude bandwidth, flight path bandwidth, and CAP requirements but still have Level 2 ratings can be seen to have a large amount of pitch attitude dropback (Figure 7.18).

Figure 7.20 shows the average HQRs for the Pitch Attitude Capture and Hold task plotted versus the pitch attitude dropback DB_θ/q_{ss} . Here, only the cases that meet the flight path bandwidth requirement in Figure 7.17 are plotted (i.e., cases where $\omega_{BW_\theta} \geq 3$ rad/sec and $\omega_{BW_\gamma} \geq 0.75$ rad/sec). A trend line for the results is also plotted on the figure. Unlike the Pitch Sum-of-Sines Tracking tasks where the coaxial-pusher and tiltrotor followed different trend lines, here the results for both aircraft line up. The trend line shows that $DB_\theta/q_{ss} \approx 0.3 \text{ sec}^{-1}$ corresponds to the Level 1/Level 2 boundary.

Pilot comments for coaxial-pusher Configuration 1-A (first order command model, low pitch attitude bandwidth) suggested that their precision suffered and that they had to lead their inputs in order to make desired performance. Tiltrotor Configuration

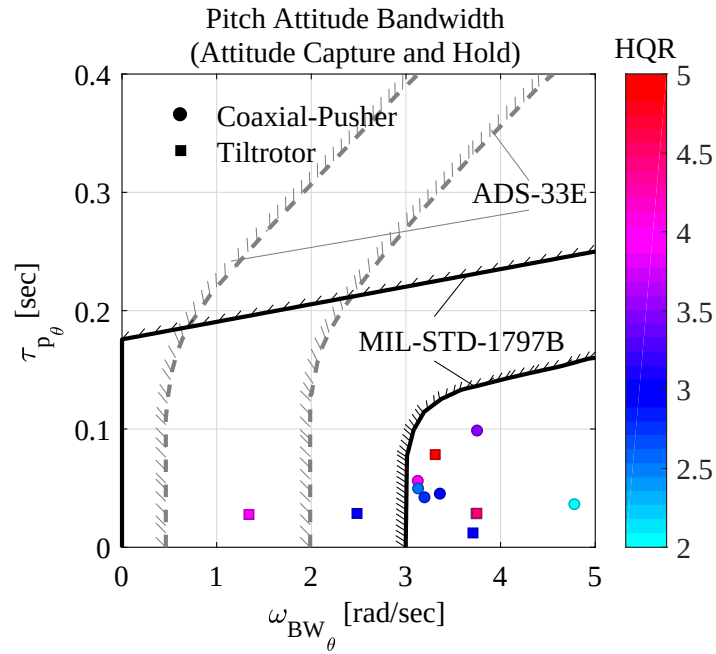


Figure 7.16. Pitch Attitude Capture and Hold handling qualities ratings plotted on the pitch attitude bandwidth requirement (coaxial-pusher and tiltrotor).

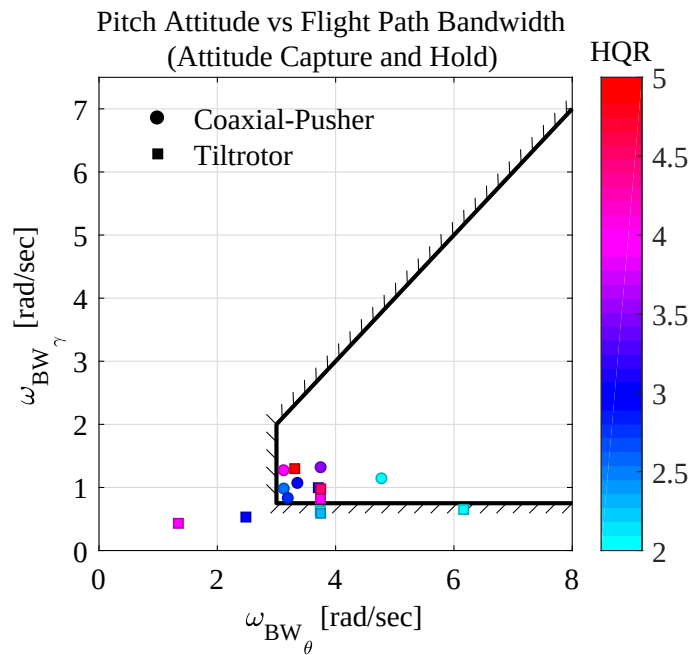


Figure 7.17. Pitch Attitude Capture and Hold handling qualities ratings plotted on the pitch attitude versus flight path bandwidth requirement (coaxial-pusher and tiltrotor).

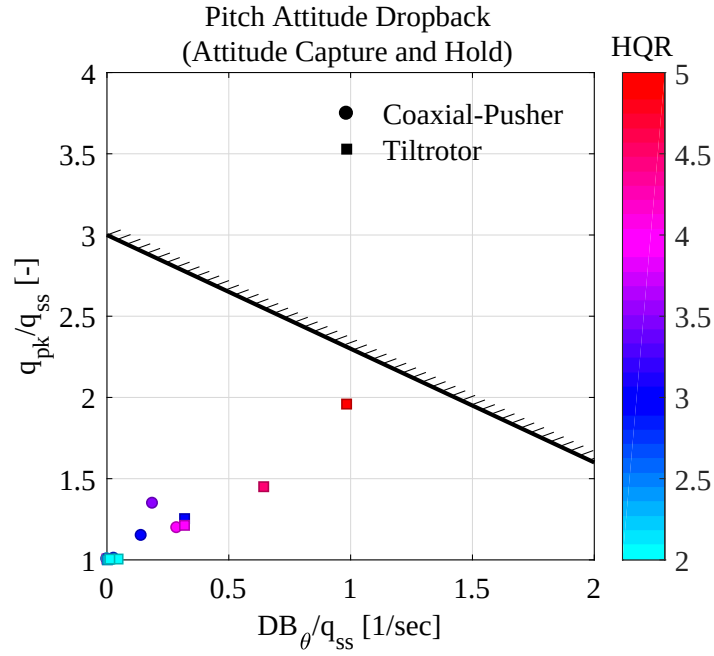


Figure 7.18. Pitch Attitude Capture and Hold handling qualities ratings plotted on the pitch attitude dropback requirement (coaxial-pusher and tiltrotor).

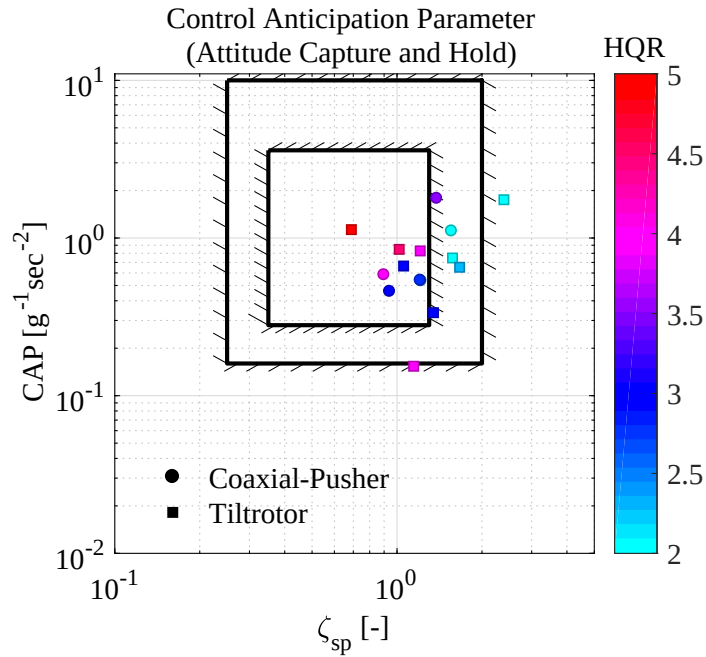


Figure 7.19. Pitch Attitude Capture and Hold handling qualities ratings plotted on the Control Anticipation Parameter (CAP) requirement (coaxial-pusher and tiltrotor).

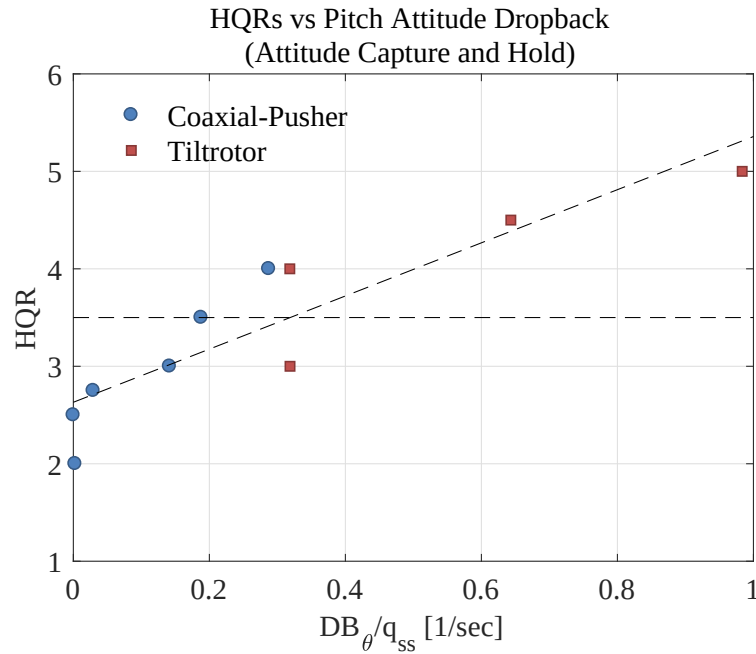


Figure 7.20. Pitch Attitude Capture and Hold handling qualities ratings plotted as a function of pitch attitude dropback (coaxial-pusher and tiltrotor)

1-B was described as “well damped” but not as good as the baseline configuration (Configuration 1-C).

Configuration 1-C was described as predictable for both aircraft, but pilots also commented that it was “heavily/too damped” (tiltrotor). Configuration 1-D was described as “more predictable,” “stops almost immediately,” and “not having to lead input.”

Configuration 2-A (tiltrotor) was overall liked by the pilots, although they did notice “ever so slight residual motion” (i.e., pitch attitude dropback). It was also compared to Configuration 1-D, which is the highest bandwidth configuration.

Pilots noted that they could get very high pitch rates with Configuration 2-B (coaxial-pusher), but noted that the dropback characteristic was “annoying.” Although they were still able to make desired performance, they noted they were “working too hard.” Similar comments were made for the tiltrotor Configuration 2-B. For the tiltrotor Configuration 2-C, the response was described as “very aggressive, but not well damped.”

7.5 Discussion

For the Pitch Sum-of-Sines Tracking task, pilots were willing to tolerate more pitch attitude dropback to get a faster pitch rate response. However, the current boundary for the pitch attitude dropback specification appears to be too relaxed, with a configuration

in the Level 1 region receiving Level 2 ratings. The pitch attitude dropback Level 1/Level 2 boundary for this task appears to be different between the two aircraft (see Figure 7.15). This may be due to the tiltrotor having a lower command model gain ($K_{\text{lon}_{\text{cm}}} = 0.3 \text{ deg/sec/\%}$ as compared to $K_{\text{lon}_{\text{cm}}} = 0.36 \text{ deg/sec/\%}$ for the coaxial-pusher), suggesting pilots are more sensitive to the absolute value of pitch attitude dropback DB_{θ} , and not necessary the normalized value $\text{DB}_{\theta}/q_{\text{ss}}$. It may also be due to the fact that the tiltrotor designs had a larger value of pitch attitude bandwidth ($\omega_{\text{BW}_{\theta}} = 3.73 \text{ rad/sec}$ as compared to $\omega_{\text{BW}_{\theta}} = 3.13 \text{ rad/sec}$ for the coaxial-pusher). Furthermore, pilots also notice when the response is over-damped (first-order pitch rate command model), especially when it is accompanied by low bandwidth.

For the Pitch Attitude Capture and Hold task, pilots were less tolerant of pitch attitude dropback, with the trend being the same for both aircraft (see Figure 7.20). In addition, pilots preferred configurations with higher levels of equivalent short period damping, with several configurations having Levels 2 and 3 values of equivalent short period damping ζ_{sp} on the CAP requirement receiving Level 1 ratings.

Overall, configurations with Level 2 or 3 values of CAP received Level 2 ratings, although these same configurations were in the Level 2 of the Pitch Attitude Bandwidth and Flight Path versus Pitch Attitude Bandwidth specifications. In addition, as mentioned above, the equivalent damping ratio value of the CAP specification did not correlate well with pilot ratings. Combined with the limited applicability of the CAP requirement to only conventional response types, it is recommended to keep this specification as a Tier 2 (check only) requirement and not enforce it as part of control system optimization.

Designs with pitch attitude bandwidth less than $\omega_{\text{BW}_{\theta}} = 2 \text{ rad/sec}$ received Level 2 ratings. This corresponds to the boundary of the ADS-33E Pitch Attitude Bandwidth specification for Target Acquisition & Tracking. Note though that the non-aggressive version of the Pitch Sum-of-Sines Tracking MTE was used, suggesting that $\omega_{\text{BW}_{\theta}} = 2 \text{ rad/sec}$ Level 1/Level 2 boundary may need to be higher for the aggressive version of the MTE (which should correspond better to Target Acquisition & Tracking). The MIL-STD-1797B Pitch Attitude Bandwidth specification for Category A flight phase (non-terminal flight phases that require rapid maneuvering, precision tracking, or precise flight path control) has a $\omega_{\text{BW}_{\theta}} = 3 \text{ rad/sec}$ Level 1/Level 2 boundary, which may be more appropriate for Target Acquisition & Tracking. Further testing with the aggressive version of the Pitch Sum-of-Sines Tracking MTE should be done to assess the correct boundary location.

Several designs with flight path bandwidth less than $\omega_{\text{BW}_{\theta}} = 0.75 \text{ rad/sec}$ (corresponding to the Level 1/Level 2 boundary of the MIL-STD-1797B Flight Path versus Pitch Attitude Bandwidth specification) received Level 1 ratings. This may be due to that fact that both MTEs tested are pitch tracking tasks, and not flight path tracking tasks (e.g., formation flying). Further testing should be done with a separate formation flying task to assess the location of the Level 1/Level 2 flight path bandwidth location.

Finally, several designs that met the Pitch Attitude Bandwidth, Flight Path versus

Pitch Attitude Bandwidth, and CAP specifications received Level 2 ratings. These all correspond to designs with values of pitch attitude dropback above $DB_{\theta}/q_{ss} \approx 0.3 \text{ sec}^{-1}$ for the Pitch Attitude Capture and Hold Task and $DB_{\theta}/q_{ss} \approx 0.8 \text{ sec}^{-1}$ for the Pitch Sum-of-Sines Tracking task. This suggests that the dropback specification should be included as a Tier 1 requirement to supplement the bandwidth specification in control law design. In addition, the boundary on the dropback specification may need to be tightened, such that designs receiving Level 2 rating are in the Level 2 region. Figure 7.21 shows a proposed updated boundary for the pitch attitude dropback specification that correlates with the evaluations conducted here.

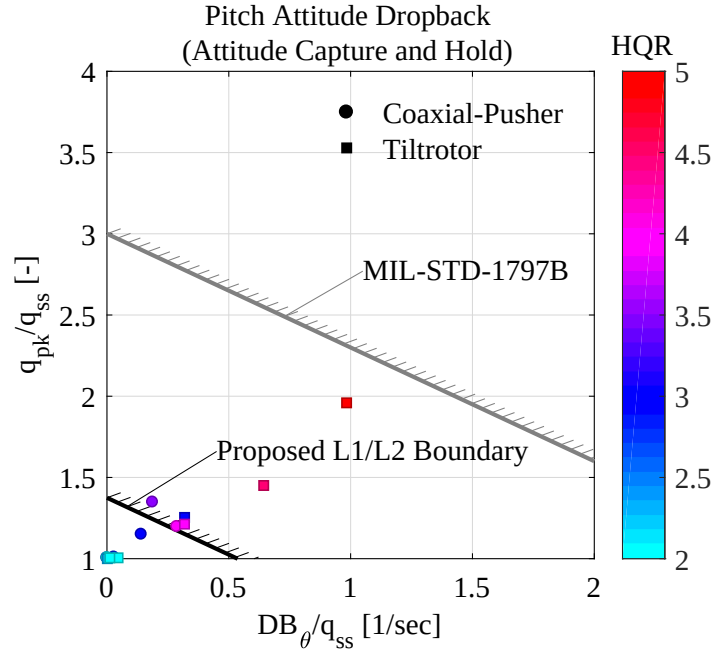


Figure 7.21. Pitch Attitude Capture and Hold handling qualities ratings plotted on the pitch attitude dropback requirement with proposed Level 1/Level 2 boundary (coaxial-pusher and tiltrotor).

7.6 Summary

Two pitch command model forms were investigated in a piloted handling qualities simulation using two pitch tracking tasks. The first command model form considered was based on a first-order transfer function, typically used for rotorcraft RCAH response types. The second command model form was based on the conventional airplane short-term pitch rate transfer function form of first-over-second order. Designs of varying levels of pitch attitude bandwidth, flight path bandwidth, CAP, and pitch attitude dropback were evaluated for both coaxial-pusher and tiltrotor. The results

suggest that there is a lower limit on pitch attitude bandwidth and an upper limit on pitch attitude dropback, and both requirements need to be met for Level 1 handling qualities, as expected. The current boundary for pitch attitude dropback appears to be too loose and may need to be tightened to match better with pilot ratings, as proposed in Figure 7.21.

Chapter 8 | Inner- and Outer-Loop Control Law Robustness Analysis

Chapters 3 and 5 described the design of inner- and outer-loop control laws for the nominal coaxial-pusher and tiltrotor aircraft dynamics. In those chapters, predicted control system performance and quantitative handling qualities results were shown for these nominal aircraft models with no uncertainty. Typically though, the aircraft models used for control system design are not known perfectly, and a robustness analysis of the control system performance to variations in the model parameters is performed (i.e., parametric model uncertainty robustness analysis [50]). The traditional approach to parametric model uncertainty robustness analysis is to perform a Monte Carlo analysis, running a large number of trial cases where the parameters of the model are varied by their known statistics and the stability and performance of the control system are assessed.

When the model is identified from test data, the identification typically provides a mean value for each model parameter as well as a covariance matrix which quantifies the accuracy of the identified parameters. When a simulation model is used, as in this case, parameter variances can be prescribed (e.g., 10%) to generate a covariance matrix. The covariance matrix is then used to determine a normal distribution for each parameter when performing the Monte Carlo analysis. To perform this process accurately, a large number of random perturbations and control system analyses must be performed, which may be very computationally expensive and time consuming. In the case of the coaxial-pusher and tiltrotor models, this is exacerbated by two factors. First, the bare-airframe models of both aircraft (described in Chapter 2) contain a large number of states (48 for the coaxial-pusher and 51 for the tiltrotor) and inputs (10 for both aircraft). This results in over 500 nonzero stability and control derivatives (excluding kinematic relationships) that can be perturbed during the robustness analysis which would require a large number of trials to perform an accurate analysis. The second factor making the standard robustness analysis not practical in this case is the large number of specifications used in the control law design [48 specification for the inner loop (Chapter 3) and 23 specification for the

outer loop (Chapter 5)].

Therefore, a new, more efficient approach is proposed here to assess the robustness of the coaxial-pusher and tiltrotor control laws to bare-airframe uncertainty. This approach reduces the number of perturbations and control system evaluations that must be made in two ways. First, instead of perturbing all of the stability and control derivatives of the full-order models, perturbations are made only to the fuselage derivatives based on their equivalent quasi-steady six degrees-of-freedom (6 DOF) rigid body values (see Sec. 2.1.2.3). Secondly, instead of performing a standard Monte Carlo analysis an alternate approach to assess control system robustness by using an Unscented Transform (UT) [115] of the model parameter statistics is employed. This UT robustness analysis technique was first used in Ref. 116 to assess the robustness of a UAV quadrotor control system.

The remainder of this chapter will describe the bare-airframe parameter perturbations and show their effects on an example key on-axis frequency response. Next, a brief overview of the Unscented Transform will be provided followed by a simple example of its application to assessing control system robustness. Results will then be shown for both the coaxial-pusher and tiltrotor inner- and outer-loop control laws, followed by a discussion of the results.

8.1 Bare-Airframe Perturbations

The bare-airframe models of both aircraft contain a large number of states (48 for the coaxial-pusher and 51 for the tiltrotor) and inputs (10 for both aircraft). This results in over 500 nonzero stability and control derivatives that can be perturbed during the robustness analysis which would require a large number of trials to get accurate results.

To limit the number of variations in the analysis, only the primary fuselage stability and control derivatives are perturbed. These are the primary derivatives that affect the aircraft responses in the frequency range where the control system is active (around the crossover frequency $\omega_c \approx 1 - 6$ rad/sec).

However, the fuselage derivatives in the higher-order models used for the coaxial-pusher and tiltrotor have small, near-zero magnitudes, since the response is dominated by the rotor. Therefore, the quasi-steady 6 DOF stability and control derivatives are first extracted from the higher-order models by reducing out the rotor, inflow, and nacelle angle states (as described in Sec. 2.1.2.3), leaving only the nine rigid body states. Perturbations to the fuselage derivatives in the full-order model are then made based on their quasi-steady 6 DOF values from the reduced order model.

Figures 8.1 and 8.2 show the coaxial-pusher roll rate frequency response to symmetric lateral cyclic input p/θ'_{1c} at a flight condition of $V = 180$ kts. The figures show the response for the full-order nominal model (solid red line) as well as the reduced-order 6 DOF nominal model (solid blue line). The figures also show perturbed

frequency responses (dashed lines) by varying the primary roll stability derivative L_p (Figure 8.1) and primary roll control derivative $L_{\theta'_{1c}}$ (Figure 8.2). The perturbed responses are shown for the 6 DOF model by perturbing the 6 DOF derivatives by $\pm 20\%$ (dashed blue lines), which results in a perturbations in the frequency response magnitude of about ± 2 dB. When perturbing the full-order response by modifying the L_p and $L_{\theta'_{1c}}$ derivatives by $\pm 20\%$ of their full-order values (dashed green lines), the frequency response magnitude is perturbed by only about ± 0.2 dB, which would not accurately assess the robustness of the control system. Finally, the figures show the full-order model with L_p and $L_{\theta'_{1c}}$ perturbed by $\pm 20\%$ of their respective 6 DOF values (dashed red lines), which results in a perturbations in the frequency responses of the same magnitude as perturbing the 6 DOF model.

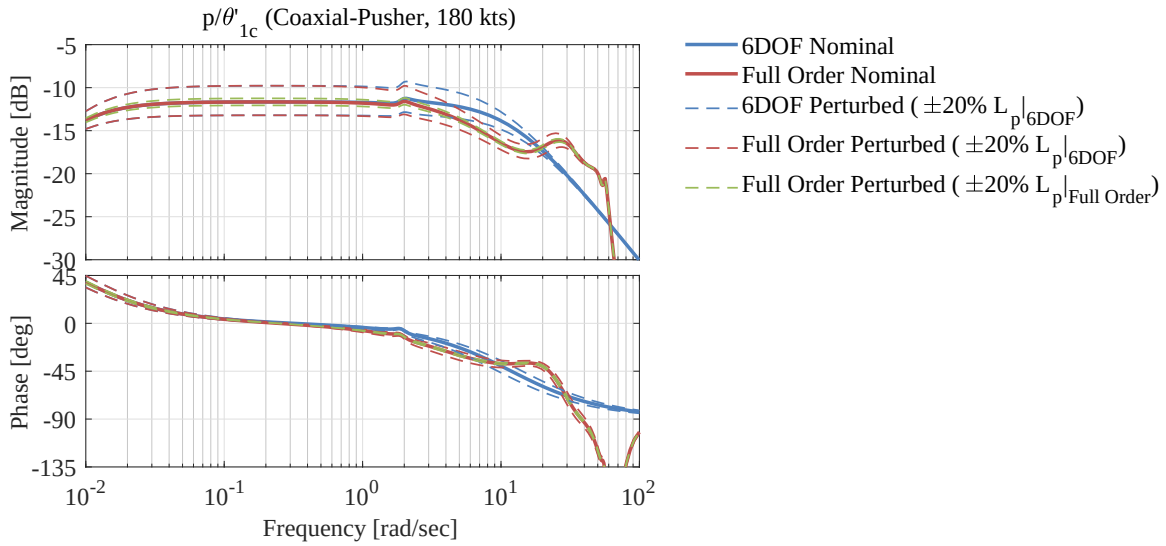


Figure 8.1. Variations in roll rate to symmetric lateral cyclic p/θ'_{1c} frequency responses due to $\pm 20\%$ perturbation in rigid-body L_p stability derivative (coaxial-pusher, 180 kts).

Thus, to get an accurate representation of the perturbed full-order model frequency responses by perturbing only the fuselage derivatives, the derivatives should be perturbed by a percentage of their quasi-steady 6 DOF values.

8.2 Unscented Transform

8.2.1 Overview

The Unscented Transform (UT) [115] is a technique for propagating uncertainty through a nonlinear transformation, and was developed as an alternate method to apply the Kalman Filter to nonlinear systems. Instead of performing a large number of random trials, such as a Monte Carlo simulation, the UT works by constructing a

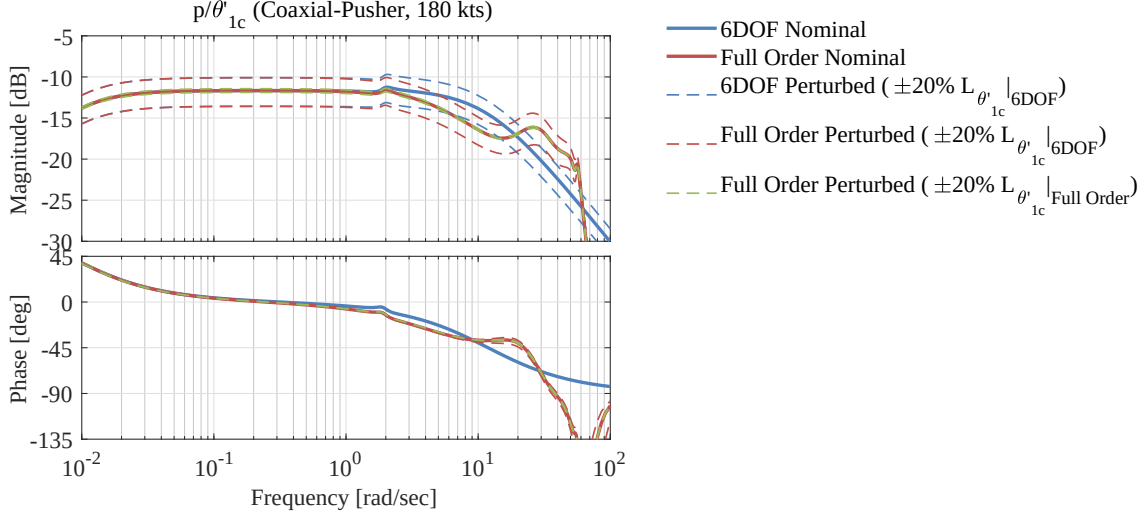


Figure 8.2. Variations in roll rate to symmetric lateral cyclic p/θ'_{1c} frequency responses due to $\pm 20\%$ perturbation in rigid-body $L_{\theta'_{1c}}$ control derivative (coaxial-pusher, 180 kts).

set of specific points, referred to as *sigma points*, which have the same known statistics (mean and covariance) as the input data set. These sigma points are then passed through the nonlinear transformation, and the *unscented* estimate can be obtained by computing the statistics of the transformed set.

Consider an input data set $\mathbf{x} \in \mathbb{R}^n$ with known mean $\bar{\mathbf{x}}$ and covariance $\mathbf{P}$, and an output data set $\mathbf{y} \in \mathbb{R}^m$ based on the nonlinear transformation h :

$$\mathbf{y} = h(\mathbf{x}) \quad (8.1)$$

The mean and covariance of output $\mathbf{y}$ can be determined by performing a Monte Carlo analysis, passing many values of $\mathbf{x}$ through the nonlinear transformation h , and analyzing the resulting values of $\mathbf{y}$, or by linearizing the nonlinear function h (as done for example in the Extended Kalman Filter). Another approach using the UT can be applied by first determining the set of $2n$ sigma points $\mathbf{x}^{(i)}$:

$$\mathbf{x}^{(i)} = \bar{\mathbf{x}} + \tilde{\mathbf{x}}^{(i)}, \quad i = 1, \dots, 2n \quad (8.2)$$

where:

$$\begin{aligned} \tilde{\mathbf{x}}^{(i)} &= +(\sqrt{n\mathbf{P}})_i^T, \quad i = 1, \dots, n \\ \tilde{\mathbf{x}}^{(i+n)} &= -(\sqrt{n\mathbf{P}})_i^T, \quad i = 1, \dots, n \end{aligned} \quad (8.3)$$

and $(\sqrt{n\mathbf{P}})_i$ is the i -th row of matrix $\sqrt{n\mathbf{P}}$.

The sigma points are then passed through the nonlinear function to get:

$$\mathbf{y}^{(i)} = h(\mathbf{x}^{(i)}) \quad (8.4)$$

Finally, the unscented mean $\bar{\mathbf{y}}_u$ and covariance $\mathbf{P}_u$ of $\mathbf{y}$ are determined as:

$$\bar{\mathbf{y}}_u = \frac{1}{2n} \sum_{i=1}^{2n} \mathbf{y}^{(i)} \quad (8.5)$$

$$\mathbf{P}_u = \frac{1}{2n} \sum_{i=1}^{2n} (\mathbf{y}^{(i)} - \bar{\mathbf{y}}) (\mathbf{y}^{(i)} - \bar{\mathbf{y}})^T \quad (8.6)$$

The following section will describe the application of the UT to control system robustness analysis.

8.2.2 Application to Control System Robustness Analysis

When assessing control system robustness to parametric model uncertainties, the parameters of the bare-airframe are the input data set $\mathbf{x}$ and the variations in the parameters are captured in the covariance matrix $\mathbf{P}$. When models are determined from system identification, $\mathbf{P}$ is typically estimated along with the mean parameter values $\bar{\mathbf{x}}$. When the models are simulation based, the covariance matrix $\mathbf{P}$ can be constructed based on a chosen variation in the parameter values (i.e., 10% variation). The nonlinear transformation h is then the conversion of the model parameters to the performance of the control system, as captures by the quantitative requirements (e.g., stability margins, piloted bandwidth and phase delay, disturbance rejection bandwidth, etc.).

To demonstrate the applicability of UT to control system robustness analysis and compare the UT results with a Monte Carlo analysis, a simple example is presented here based on a first-order approximation of a UH-60 roll rate response in hover [50, 101]:

$$\frac{p}{\delta_{\text{lat}}} = \frac{L_{\delta_{\text{lat}}}}{s - L_p} e^{-\tau_{\phi} s} \quad (8.7)$$

where:

$$L_{\delta_{\text{lat}}} = 1.33 \text{ deg/sec}^2/\text{in}$$

$$L_p = -3.34 \text{ 1/sec, and}$$

$$\tau_{\phi} = 0.16 \text{ sec}$$

The control system in this simple example is comprised of a roll rate feedback gain K_p and a roll attitude feedback gain K_{ϕ} , and was optimized in CONDUIT® [50] to meet the specifications in Table 8.1. The resulting optimized feedback gain values are:

$$K_p = 2.30 \text{ in/deg/sec} \quad (8.8)$$

$$K_\phi = 5.33 \text{ in/deg} \quad (8.9)$$

In this example, the UT input data set $\mathbf{x}$ is composed of the three parameters ($n = 3$) given in Eq. 8.7, with the mean value $\bar{\mathbf{x}}$ given by:

$$\bar{\mathbf{x}} = \begin{bmatrix} L_{\delta_{\text{lat}}} \\ L_p \\ \tau_\phi \end{bmatrix} = \begin{bmatrix} 1.33 \\ -3.34 \\ 0.16 \end{bmatrix} \quad (8.10)$$

Perturbations with standard deviations of $\sigma = 10\%$ are used to assess the control system robustness. Therefore, the covariance matrix $\mathbf{P}$ is given by:

$$\mathbf{P} = \begin{bmatrix} (0.1L_{\delta_{\text{lat}}})^2 & 0 & 0 \\ 0 & (0.1L_p)^2 & 0 \\ 0 & 0 & (0.1\tau_\phi)^2 \end{bmatrix} \quad (8.11)$$

Figure 8.3 shows a histogram of the perturbed roll damping derivative L_p for 200 trials made for the Monte Carlo analysis. The figure also shows the probability distribution for L_p from covariance matrix $\mathbf{P}$ (Eq. 8.11), with the 1σ , 2σ , and 3σ regions highlighted, corresponding to 10%, 20%, and 30% variation in L_p .

Figures 8.4 through 8.6 show the results of the Monte Carlo and UT robustness analyses for three specifications—stability margins, piloted bandwidth/phase delay, and disturbance rejection bandwidth/peak (DRB/DRP). On each figure the nominal (unperturbed model) results are plotted as a black plus sign. The results of the Monte Carlos analysis are shown as 200 blue dots, as well as the 3σ confidence ellipsoid calculated from the covariance of the Monte Carlos results (blue ellipse). The results of the UT sigma points $[\mathbf{y}^{(i)}]$ are shown as the six ($2n$) red x's on the figures. Finally, the 3σ confidence ellipsoids derived from the UT results ($\mathbf{P}_u$) are shown as the red ellipses.

There is an excellent agreement between the Monte Carlo and UT confidence ellipsoids for all three specifications shown in Figures 8.4 through 8.6, with the covariance matrices of the Monte Carlo results and UT analysis differing by less than 5%. Furthermore, in this example, the 3σ confidence ellipsoids remain within Level 1 for the stability margins (Figure 8.4) and piloted bandwidth (Figure 8.5) specifications demonstrating the robustness of the control system. However, the disturbance rejection 3σ confidence ellipsoids (Figure 8.6) does encroach into the Level 2 DRB region, suggesting that some additional margin should be used for this specification (i.e., design margin as discussed in Ref. 50).

These results demonstrate that the robustness of the control system to variations in the bare-airframe parameters can be accurately assessed by performing $2n$ (six in this example) evaluations, significantly less than required for a Monte Carlo analysis.

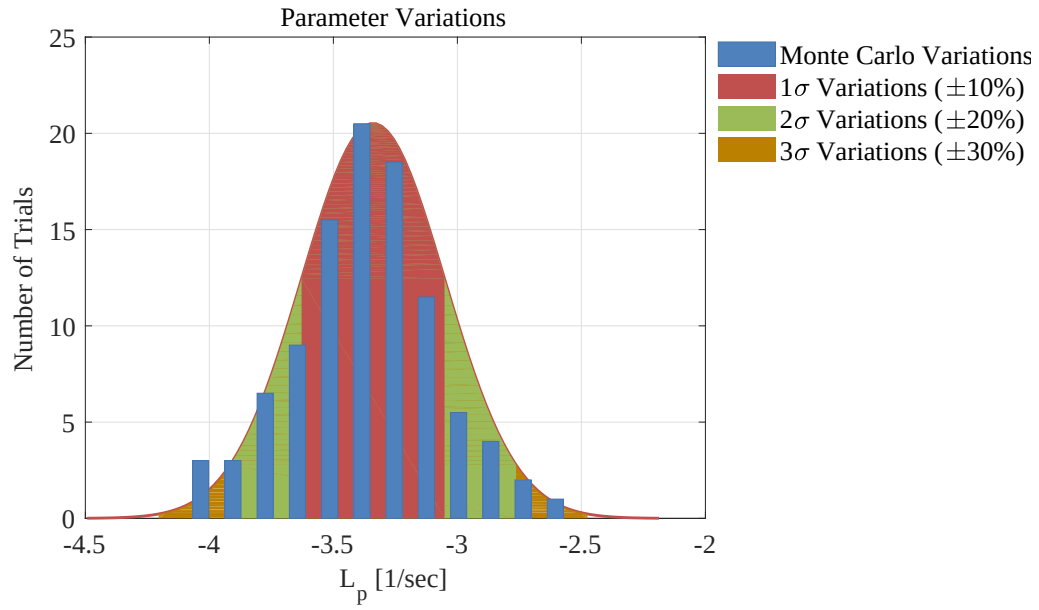


Figure 8.3. Variations in L_p parameter used for Monte Carlo analysis (UH-60 example, hover).

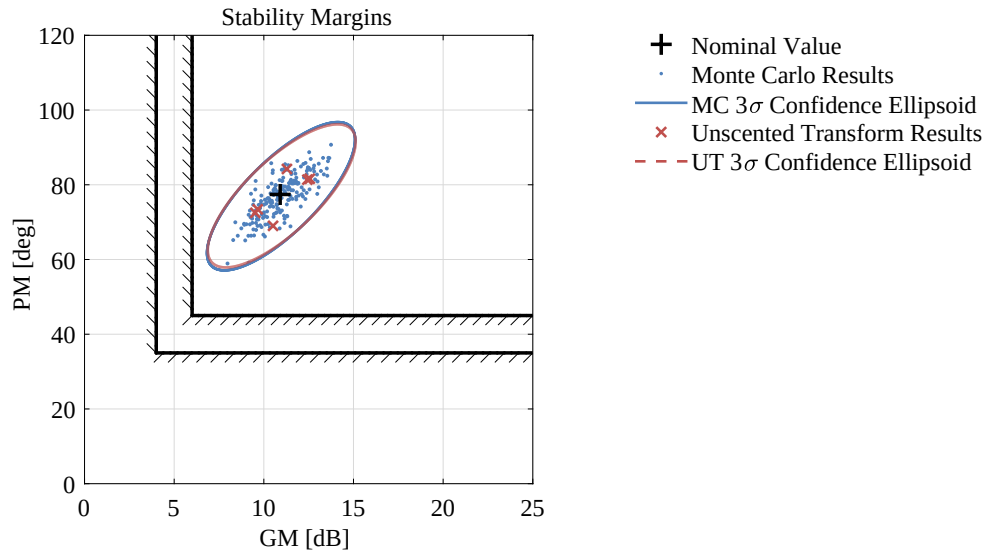
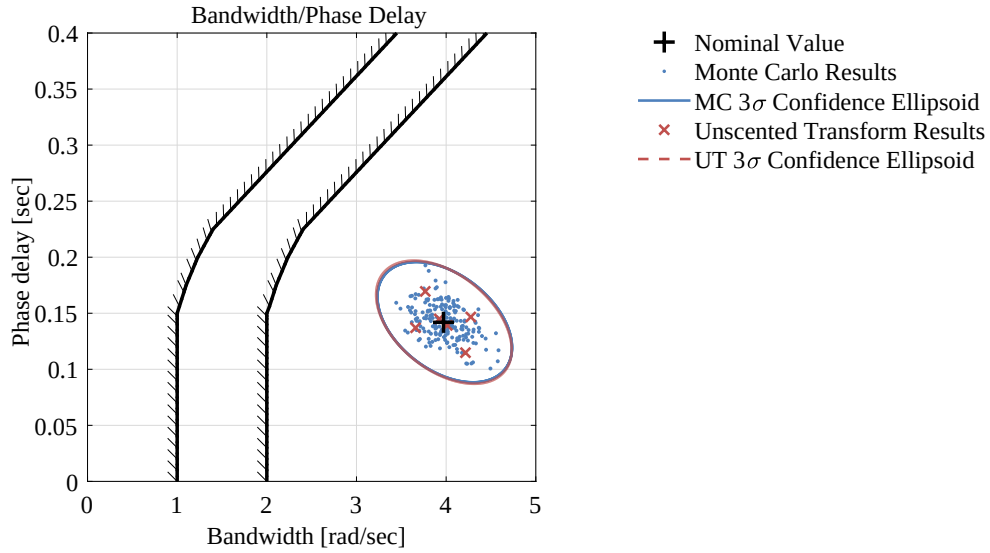


Figure 8.4. Comparison of Monte Carlo and Unscented Transform stability margin robustness analyses (UH-60 example, hover).

Table 8.1. Control System Specifications (UH-60 Example)

CONDUIT [®]		
Spec Name	Description (Motivation)	Source
<i>Hard Constraints (Stability Requirements)</i>		
EigLcG1	Eigenvalues in L.H.P. (Stability)	Generic
StbMgG1	Gain and phase margin (Stability)	AS94900
<i>Soft Constraints (Handling Qualities Requirements)</i>		
BnwAtH1	Roll attitude bandwidth and phase delay, hover (HQ)	ADS-33E
DstBwG1	Disturbance rejection bandwidth (Loads, Ride Quality)	Ref. 102
DstPkG1	Disturbance rejection peak (Loads, Ride Quality)	Ref. 102
EigDpG1	Eigenvalue damping (HQ, Loads)	Generic
CrsMnG2	Minimum $\omega_c \geq 2.5$ rad/sec (Robustness)	Generic
<i>Summed Objective (Performance Requirements)</i>		
CrsLnG1	Crossover frequency (Act. Activity)	Generic
RmsAcG1	Actuator RMS (Act. Activity)	Generic

**Figure 8.5.** Comparison of Monte Carlo and Unscented Transform piloted bandwidth and phase delay robustness analyses (UH-60 example, hover).

8.3 Coaxial-Pusher Robustness Analysis

This section shows the UT robustness analysis performed for the coaxial-pusher inner- and outer-loop control laws at the $V_{CAS} = 180$ kts flight condition. For this analysis, the vector of perturbed stability and control derivatives is given by:

$$\mathbf{x} = \begin{bmatrix} X_u & Y_v & Z_w & L_v & L_p & M_u & M_w & M_q & N_v & N_r & \dots \end{bmatrix}$$

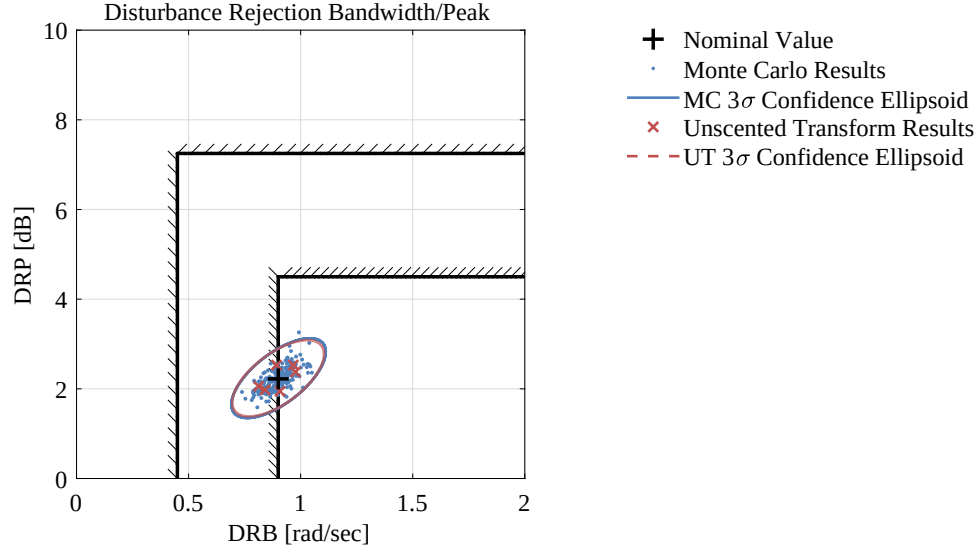


Figure 8.6. Comparison of Monte Carlo and Unscented Transform disturbance rejection bandwidth and peak robustness analyses (UH-60 example, hover).

$$\begin{bmatrix} Z_{\theta_0} & L_{\theta'_{1c}} & M_{\theta'_{1s}} & M_{\delta_e} & N_{\delta_r} & X_{\theta_{0PP}} \end{bmatrix}^T \quad (8.12)$$

The perturbations in each derivative were chosen to have a standard deviation of $\sigma = 10\%$ of each derivative's quasi-static 6 DOF value, as described in Sec. 8.1. Thus:

$$\mathbf{P} = \text{diag}\left((0.1\mathbf{x}_{|6\text{DOF}})^2\right) \quad (8.13)$$

Results are presented here for the inner-loop control laws described in Chapter 3 and the outer-loop control laws described in Chapter 5.

8.3.1 Inner-Loop Control Laws

For the inner loop, control system robustness was assessed using both Monte Carlo and Unscented Transform (UT) methods to further validate the UT approach. Two hundred random perturbations were run for the Monte Carlo analysis. Figure 8.7 shows the results for several of the key design specifications, including stability margins, disturbance rejection bandwidth/peak, model following cost, and piloted bandwidth/phase delay. For all specifications, the nominal value is shown as a blue dot while results from the Monte Carlo analysis are shown as smaller purple dots. In addition, the UT analysis 1σ , 2σ , and 3σ confidence ellipsoids are shown as the red, green, and orange ellipses, respectively.

For all of the specification shown in Figure 8.7, there is a good agreement between the results of the Monte Carlo and UT analyses. The robustness analysis results for the roll and yaw axis stability margins (Figure 8.7, plots 1 and 3) are all within Level

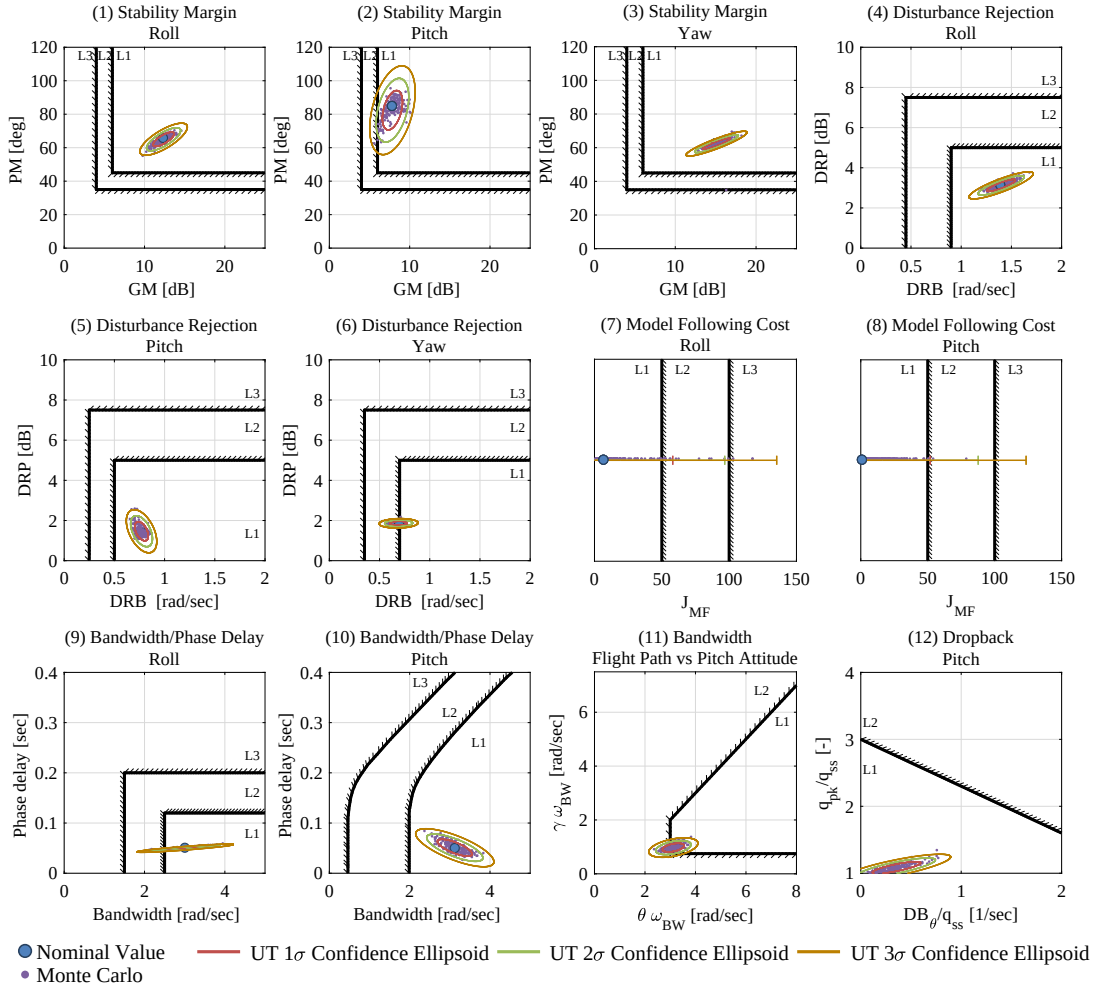


Figure 8.7. Monte Carlo (200 trials) and Unscented Transform robustness analysis results (coaxial-pusher, inner loop, 180 kts).

1, demonstrating the stability robustness of the inner-loop control laws. For the pitch axis stability margins (Figure 8.7, plots 2), the 2 σ confidence ellipsoid is in Level 1 while the 3 σ confidence ellipsoid encroaches into the Level 2 region with a minimum gain margin of $GM \approx 5$ dB, which shows the control system robustness for parameter variations on the order of $\pm 20\%$.

The robustness analysis results for disturbance rejection bandwidth and peak (DRB and DRP) for the roll and pitch axes are all within Level 1 (Figure 8.7, plots 4 and 5). This is a result of the design margin that was applied to these specifications during the inner-loop control system optimization, which required the designs to meet a higher DRB value than the Level 1/Level 2 boundary in the specification [102, 103]. A design margin was not applied to the DRB specification in the yaw axis (Figure 8.7,

plot 6), and as a result the robustness analysis results encroach into the Level 2 region of the specification. The 3σ confidence ellipsoid reaches $\text{DRB} = 0.5 \text{ rad/sec}$, which is below the $\text{DRB} = 0.7 \text{ rad/sec}$ Level 1/Level 2 boundary [102, 103]. It is therefore recommended that a design margin of at least 0.2 rad/sec be applied to this specification to ensure robust performance.

Model following performance robustness in both the roll and pitch axes is shown on the model following cost specifications (Figure 8.7, plots 7 and 8), which compare the closed-loop frequency response in each axis with the frequency response of the command model using a cost function J_{MF} based on the weighted difference in the magnitude and phase of the responses. For both axes, the 1σ confidence ellipsoid has a maximum model following cost less than $J_{\text{MF}} \approx 50$, which corresponds to the Level 1/Level 2 boundary used for nominal configuration [50]. In addition, the 2σ confidence ellipsoid has a maximum model following cost less than $J_{\text{MF}} \approx 100$, which corresponds to the Level 1/Level 2 boundary used for off-nominal configuration [50]. This demonstrated robust closed-loop performance of the control laws.

Closed-loop performance robustness is further demonstrated using the roll and pitch attitude piloted bandwidth/phase delay specifications (Figure 8.7, plots 9 and 10). In the roll axis, the 1σ confidence ellipsoid remains within Level 1 for the piloted bandwidth/phase delay specification, while the 2σ and 3σ confidence ellipsoids encroach into the Level 2 region. The worst case bandwidth remains above $\omega_{\text{BW}_\phi} = 1.8 \text{ rad/sec}$ for the 3σ confidence ellipsoid, and the phase delay remains well within Level 1. In the pitch axis, the 3σ confidence ellipsoid remains within Level 1, demonstrating good closed-loop robustness performance.

Finally, two MIL-STD-1797B specifications are shown in Figure 8.7—the flight path vs. pitch attitude bandwidth requirement (plot 11) and the pitch attitude dropback requirement (plot 12). For the flight path vs. pitch attitude bandwidth requirement, the 3σ confidence ellipsoid remains above $\omega_{\text{BW}_\gamma} = 0.6 \text{ rad/sec}$ which is similar to designs that received assigned Level 1 Handling Qualities Ratings (HQRs) as shown in Chapter 7 (Figures 7.12 and 7.17). For the pitch attitude dropback specification, all confidence ellipsoids remain within the Level 1 region.

This subset of specifications used for the inner-loop control system optimization and analysis demonstrate the good stability and performance robustness of the coaxial-pusher inner-loop control laws.

8.3.2 Outer-Loop Control Laws

Outer-loop robustness was assessed using the UT method and the results are shown in Figure 8.8 for several of the key design specifications. The nominal value for each specification is shown as a blue dot while the UT analysis 1σ , 2σ , and 3σ confidence ellipsoids are shown as the red, green, and orange ellipses, respectively.

Plots 1 and 2 in Figure 8.8 show the airspeed V and flight path γ loop stability margins, respectively. All confidence ellipsoids are within the Level 1 region, demon-

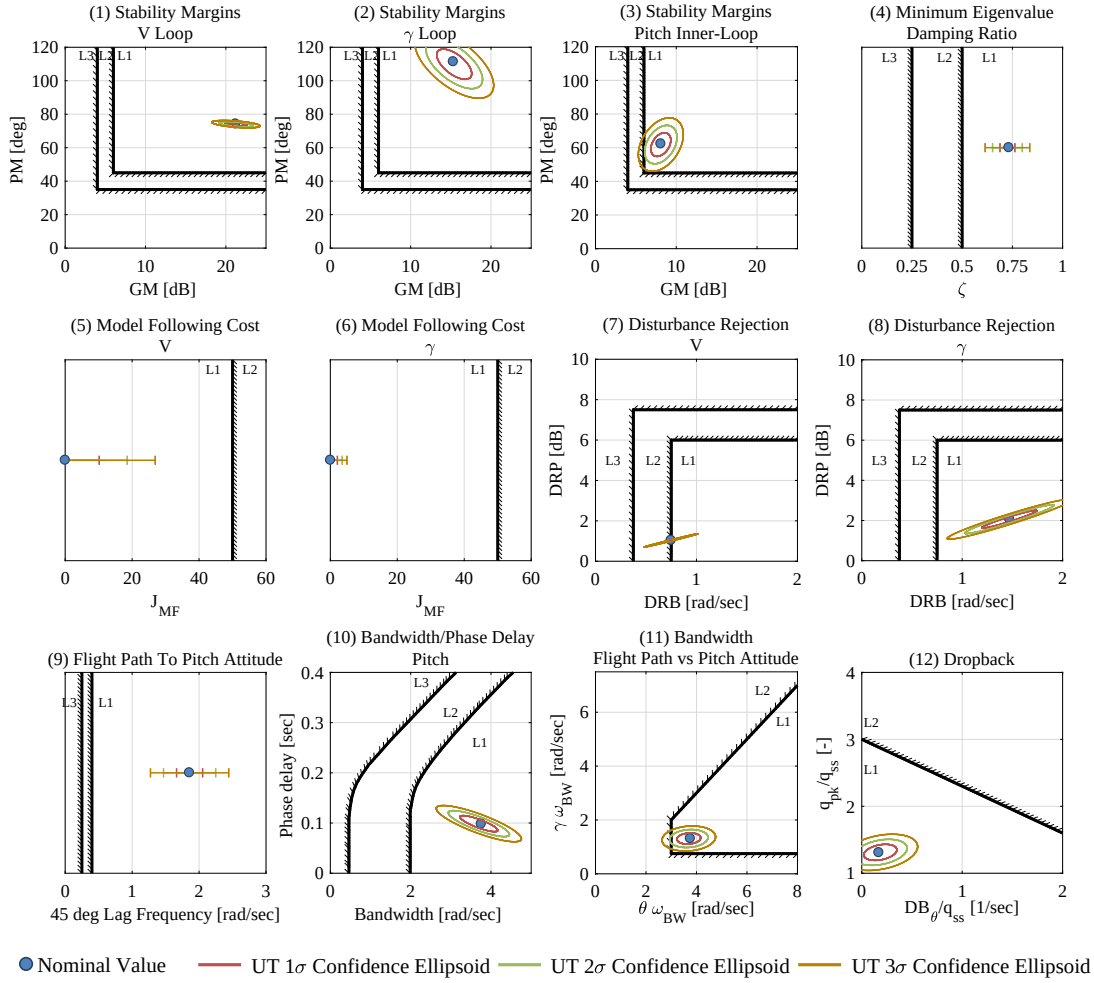


Figure 8.8. Unscented Transform robustness analysis results (coaxial-pusher, outer loop, 180 kts).

strating good stability robustness of the outer loops. Plot 3 in Figure 8.8 shows the inner pitch loop stability margins with the outer loops closed. For this specification, the 2σ confidence ellipsoid remains within Level 1, while the worst case 3σ confidence ellipsoid gain margin degrades into Level 2 (5.3 dB).

Plot 4 in Figure 8.8 shows the minimum eigenvalue damping ratio for the outer loops, with all confidence ellipsoids remaining within Level 1.

Plots 5 and 6 in Figure 8.8 show the airspeed and flight path closed-loop model following cost J_{MF} requirements. The worst case 3σ model following cost for the airspeed response is less than $J_{MF_V} = 30$, while for the flight path response the worst case 3σ model following cost is less than $J_{MF_\gamma} = 5$. This demonstrates the improved performance robustness of the outer-loop control system, as captured by the model

following specification, over that on the inner-loop control system.

Plots 7 and 8 in Figure 8.8 show the airspeed and flight path DRB/DRP specifications. For the airspeed loop, the worst case DRB degraded from the nominal value of $\text{DRB}_V = 0.75$ rad/sec to $\text{DRB}_V = 0.5$ rad/sec. Since there is currently no published high-speed airspeed DRB requirement, future studies will have to determine if degradation to $\text{DRB}_V = 0.5$ rad/sec results in Level 2 ratings. In the case of the flight path loop DRB/DRP specification, all confidence ellipsoids remaining within the design requirement of $\text{DRB}_\gamma \geq 0.75$ rad/sec and $\text{DRP}_\gamma \leq 6$ dB.

Plots 9–12 in Figure 8.8 show several of the outer-loop piloted response specifications. The ADS-33E flight path response to pitch attitude specification values (plot 9) and pitch attitude piloted bandwidth/phase delay (plot 10) all remain within Level 1. For the MIL-STD-1797B flight path vs. pitch attitude bandwidth requirement (plot 11), the 2σ confidence ellipsoid remains within Level 1, while the 3σ confidence ellipsoid degrades into Level 2 with $\omega_{\text{BW}_\theta} = 2.6$ rad/sec. Finally, all confidence ellipsoids remain within the Level 1 region for the pitch attitude dropback requirement (plot 12).

This subset of specifications used for the outer-loop control system optimization and analysis demonstrate the good stability and performance robustness of the coaxial-pusher outer-loop control laws.

8.4 Tiltrotor Robustness Analysis

This section shows the UT robustness analysis performed for the tiltrotor inner- and outer-loop control laws at the $V_{\text{CAS}} = 180$ kts, $\delta_{\text{nac}} = 0$ deg (airplane mode) flight condition. For this analysis, the vector of perturbed stability and control derivatives is given by:

$$\mathbf{x} = \begin{bmatrix} X_u & Y_v & Z_w & L_v & L_p & M_u & M_w & M_q & N_v & N_r & \dots \\ L_{\Delta\theta'_{1s}} & L_{\delta_a} & L_{\delta_r} & M_{\delta_e} & N_{\delta_r} & X_{\theta_0} \end{bmatrix}^T \quad (8.14)$$

The perturbations in each derivative were chosen to have a standard deviation of $\sigma = 10\%$ of each derivative's quasi-steady 6 DOF value, as described in Sec. 8.1. Thus:

$$\mathbf{P} = \text{diag}\left((0.1\mathbf{x}|_{6\text{DOF}})^2\right) \quad (8.15)$$

Results are presented here for the inner-loop control laws described in Chapter 3 and the outer-loop control laws described in Chapter 5.

8.4.1 Inner-Loop Control Laws

Inner-loop robustness was assessed using the UT method and the results are shown in Figure 8.9 for several of the key design specifications. The nominal value for each specification is shown as a blue dot while the UT analysis 1σ , 2σ , and 3σ confidence ellipsoids are shown as the red, green, and orange ellipses, respectively.

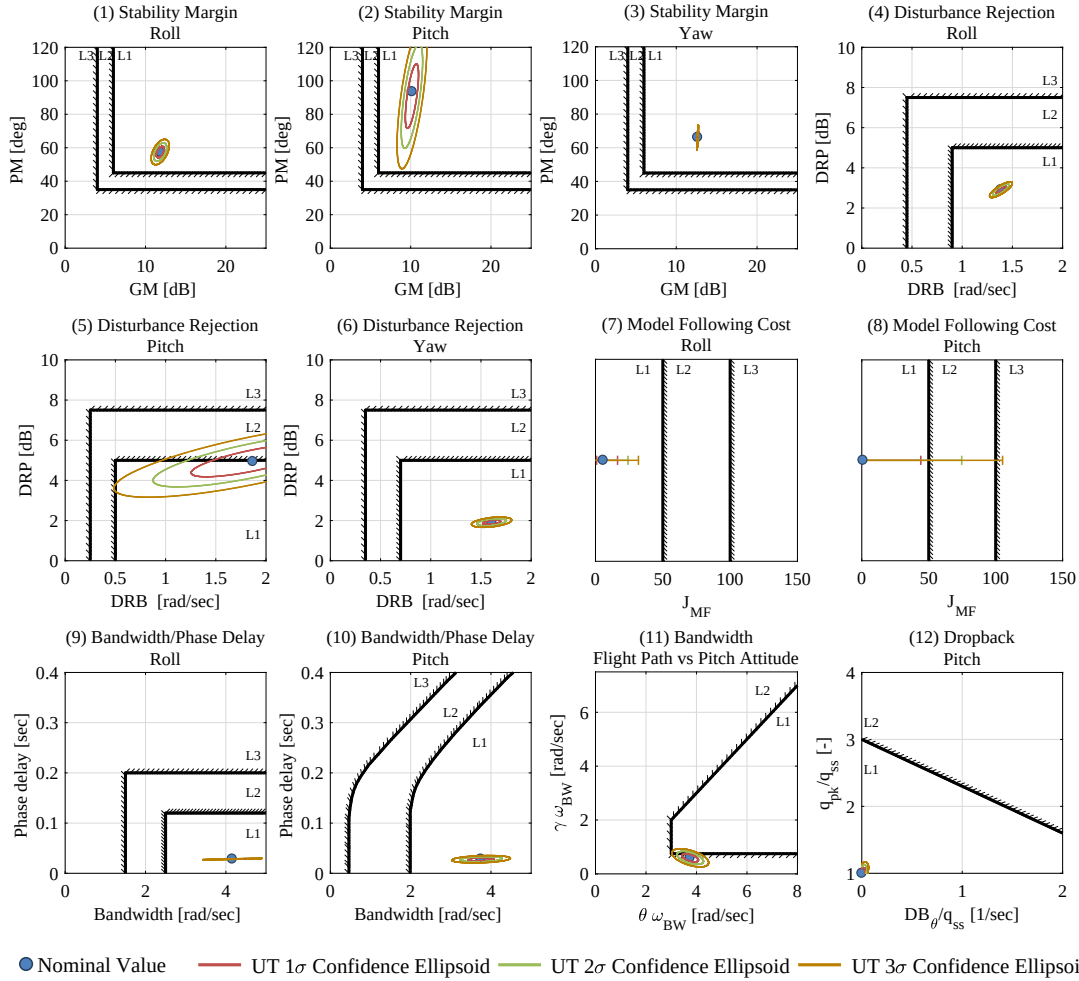


Figure 8.9. Unscented Transform robustness analysis results (tiltrotor, inner loop, 180 kts).

Overall, there is good stability and performance robustness for the tiltrotor inner-loop control laws. The variations in roll and yaw axis specification values are the smallest, while there are larger variations in the pitch axis specification. This is because the roll and yaw axis broken-loop frequency responses have the desired K/s (-20 dB/decade slope of the magnitude curve) shape around crossover frequency (Figures 3.16 and 3.18), which results in small variations in crossover frequency for

small variations in the bare-airframe response. Conversely, the pitch axis broken-loop frequency response (Figure 3.17) has a flat region around 0 dB from $\omega \approx 1 - 7$ rad/sec, which results in potentially large variations in crossover frequency for small variations in the bare-airframe response, and thus less stability and performance robustness. This characteristic of the pitch broken-loop response is a result of the large value of bare-airframe flight path-attitude lag T_{θ_2} , as described in Sec. 5.6.

All confidence ellipsoids for the roll, pitch, and yaw axis stability margins (Figure 8.9, plots 1–3) remain within Level 1, demonstrating the good stability robustness of the inner-loop control laws.

Robustness analysis results for roll and yaw DRB/DRP are all within Level 1 (Figure 8.9, plots 4 and 6). In the case of the pitch axis DRB/DRP (Figure 8.9, plot 5), the robustness analysis results encroach into the Level 2 DRP region, with the worst case 3σ confidence ellipsoid having a maximum value of $\text{DRP} = 6.8$ dB.

Plots 7 and 8 in Figure 8.9 show the roll and pitch axis model following cost results. In the roll axis, the maximum 3σ confidence ellipsoid model following cost is $J_{\text{MF}_p} \approx 30$, within Level 1. In the pitch axis, the maximum 3σ confidence ellipsoid model following cost is $J_{\text{MF}_q} \approx 100$, which corresponds to the Level 1/Level 2 boundary used for off-nominal configuration [50]. This demonstrates the robust closed-loop performance of the control laws.

All confidence ellipsoids remain within Level 1 for the roll and pitch attitude piloted bandwidth/phase delay specifications (Figure 8.9, plots 9 and 10), further demonstrating robust closed-loop performance.

Finally, two MIL-STD-1797B specifications are shown in Figure 8.9—the flight path vs. pitch attitude bandwidth requirement (plot 11) and the pitch attitude dropback requirement (plot 12). For the flight path vs. pitch attitude bandwidth requirement, the nominal value is in Level 2, and the 3σ confidence ellipsoid remains above $\omega_{\text{BW}_\gamma} = 0.2$ rad/sec. For the pitch attitude dropback specification, all confidence ellipsoids remain within the Level 1 region.

This subset of specifications used for the inner-loop control law optimization and analysis demonstrate the good stability and performance robustness of the tiltrotor inner-loop control laws.

8.4.2 Outer-Loop Control Laws

Outer-loop robustness was assessed using the UT method and the results are shown in Figure 8.10 for several of the key design specifications. The nominal value for each specification is shown as a blue dot while the UT analysis 1σ , 2σ , and 3σ confidence ellipsoids are shown as the red, green, and orange ellipses, respectively.

Plots 1 and 2 in Figure 8.10 show the airspeed V and flight path γ loop stability margins, respectively. For the airspeed loop, all confidence ellipsoids are within the Level 1. For the flight path loop, the 2σ confidence ellipsoid remains within Level 1 while the 3σ confidence ellipsoid degraded to Level 2 gain margin. Plot 3 in Figure 8.10

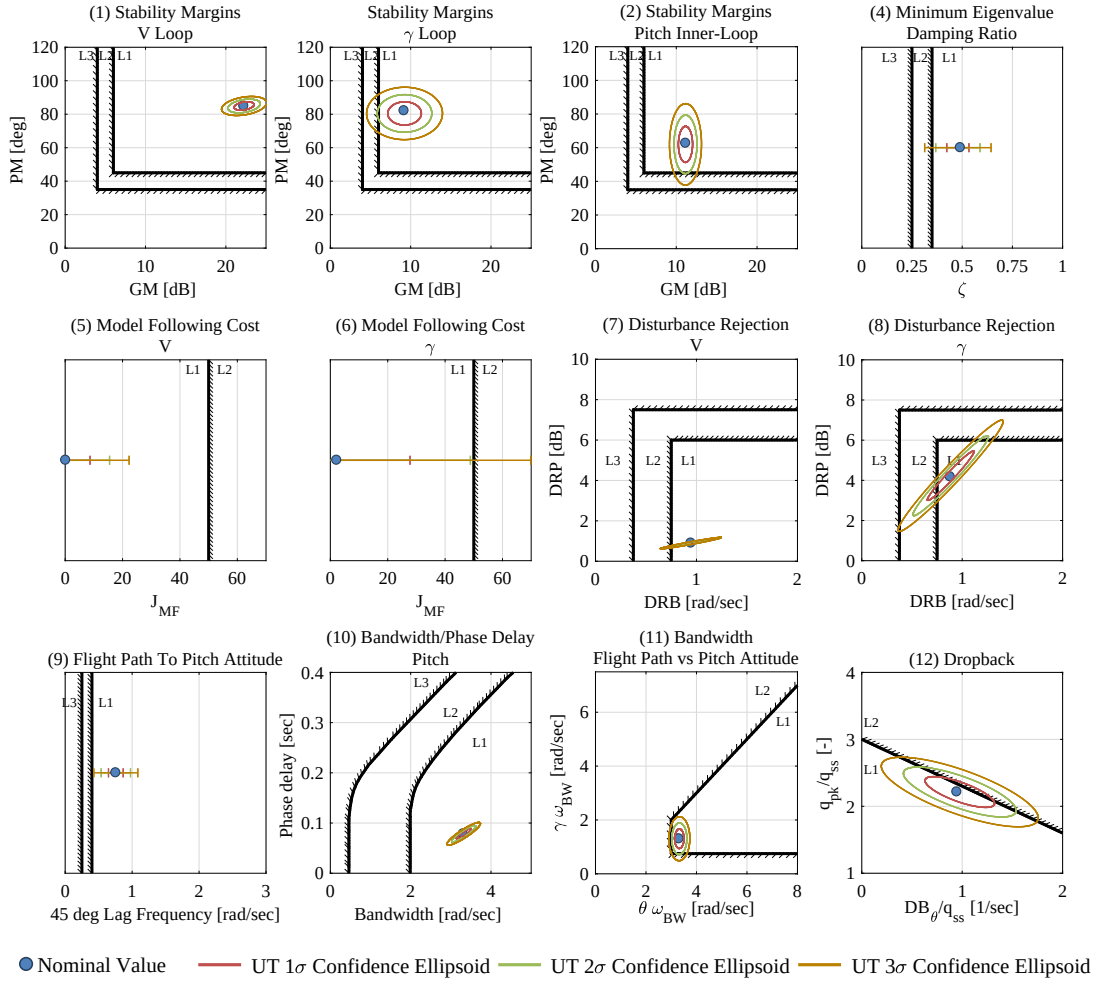


Figure 8.10. Unscented Transform robustness analysis results (tiltrotor, outer loop, 180 kts).

shows the inner pitch loop stability margins with the outer loops closed. For this specification, the 2σ confidence ellipsoid remains within Level 1, while the worst case 3σ confidence ellipsoid phase margin degrades to Level 2. Overall, there is good stability robustness for the outer-loop control laws.

Plot 4 in Figure 8.10 shows the minimum eigenvalue damping ratio for the outer loops, with the 2σ confidence ellipsoid remaining within Level 1 and the 3σ confidence ellipsoid encroaching into Level 2.

Plots 5 and 6 in Figure 8.10 show the airspeed and flight path model following cost requirements. The worst case 3σ confidence ellipsoid model following cost for the airspeed response is less than $J_{MF_V} = 30$, while for the flight path response the worst case 3σ confidence ellipsoid model following cost is less than $J_{MF_\gamma} = 70$, below

the $J_{MF} = 100$ limit for off-nominal configurations. This demonstrates the good performance robustness for the outer-loop control laws.

Plots 7 and 8 in Figure 8.10 show the airspeed and flight path DRB/DRP specifications. For the airspeed loop, the worst case DRB degrades from the nominal value of $DRB_V = 0.88$ rad/sec to $DRB_V = 0.64$ rad/sec. In the case of the flight path loop DRB/DRP specification, worst case DRB for the 3σ confidence ellipsoid degrades from the nominal value of $DRB_\gamma = 0.75$ rad/sec to $DRB_\gamma = 0.4$ rad/sec while the worst case DRP degrades from a nominal value of $DRP_\gamma = 4.18$ dB to $DRP_\gamma \approx 7$ dB. Since there are currently no published high-speed airspeed or flight path DRB/DRP requirement, future studies will have to determine if these degradations result in Level 2 ratings.

Plots 9–12 in Figure 8.10 show several of the outer-loop piloted response specifications. The ADS-33E flight path response to pitch attitude specification values (plot 9) and pitch attitude bandwidth/phase delay (plot 10) all remain within Level 1. For the MIL-STD-1797B flight path vs. pitch attitude bandwidth requirement (plot 11), the 2σ confidence ellipsoid remains within Level 1, while the 3σ confidence ellipsoid degrades into Level 2 with $\omega_{BW_\gamma} \approx 0.5$ rad/sec. Finally, for the pitch attitude dropback requirement (plot 12), the major axes of the confidence ellipsoids are aligned with the Level 1/Level 2 boundary, but all encroach into the Level 2 region.

8.5 Discussion

8.5.1 Unscented Transform Robustness Analysis

The Unscented Transform (UT) approach worked well to assess control system robustness to bare-airframe parameter variations. Comparisons of the UT results with Monte Carlo analysis results were shown for both a simple example based on a UH-60 first-order roll rate response as well as the coaxial-pusher inner-loop control laws. In both cases, confidence ellipsoids for the resulting control system performance and handling-qualities metrics generated from the Monte Carlo and UT analyses were in excellent agreement.

8.5.2 Inner-Loop Versus Outer-Loop Robustness

As expected, the performance robustness of the outer-loop control system, as captured by the model following specifications, was improved over that on the inner-loop control system for both coaxial-pusher and tiltrotor. This is because the inner-loop control system suppresses the variations in the bare-airframe below the inner-loop crossover frequency and provides the outer-loop control system with less variations in the aircraft response.

8.6 Summary

This chapter described the robustness analysis of the coaxial-pusher and tiltrotor inner- and outer-loop control laws. To make the robustness analysis more computationally efficient for these cases with a large number of bare-airframe parameters that can be perturbed and a large number of stability, handling-qualities, and performance requirements that need to be evaluated, the Unscented Transform approach was used instead of a more traditional Monte Carlo analysis. In addition, only the fuselage stability and control derivatives were perturbed in the analyses performed here, since those have the largest effect on the aircraft response in the frequency range where the control system is active ($\omega \lesssim 10$ rad/sec). To accurately assess the robustness of the control laws, the fuselage derivatives of the full-order bare-airframe models were varied by perturbations with standard deviation $\sigma = 10\%$ of their quasi-steady (6 DOF) values. The inner- and outer-loop control laws for both aircraft were shown to have good stability and performance robustness with $2\sigma - 3\sigma$ confidence ellipsoids within the Level 1 region of most specifications, covering $95\% - 99.7\%$ of perturbation cases, corresponding to $20\% - 30\%$ variations in the model parameter values.

Chapter 9 | Conclusions and Future Work

9.1 Conclusions

Chapter 2 described flight dynamics models of generic lift offset coaxial and tiltrotor aircraft developed in the comprehensive rotorcraft simulation code HeliUM. To use the models for control law design and real-time piloted simulation, a set of linear models and trim data were extracted from HeliUM spanning the flight envelope of each aircraft. The linear models were used to develop a pseudo-inverse based control allocation scheme for each aircraft. The linear models and trim data were combined into stitched simulation models that provide real-time simulation capabilities. The following conclusions are drawn:

1. The models for both aircraft showed good agreement with previously published flight data for similar aircraft configurations. The coaxial-pusher model was compared to X2 flight data, while the tiltrotor model was compared to XV-15 flight data. Although the aircraft modeled in this paper are larger than the flight vehicles they were compared to, excellent agreement in trim trends and frequency response characteristics were seen. This gives confidence in using the models presented here for future flight-control and handling-qualities research for these types of advanced rotorcraft configurations.
2. Control allocation using weighted pseudo inverse worked well to allocate demanded moments between the multiple rotor and aerosurface controls of each aircraft as flight condition varied. In the case of the coaxial-pusher, the weighted pseudo inverse solution provided the optimal phasing to of the cyclic rotor controls for the very stiff rotor.
3. Primary on-axis frequency responses were shown at several different flight conditions. Both aircraft have classical helicopter frequency responses in hover, with unstable low-frequency pitch and roll hovering cubics, and rotor modes at high frequency. At high speed, both aircraft exhibit classical fixed-wing responses. Namely, a lightly-damped Dutch roll mode is present in the lateral/direction

axes for both aircraft. In addition, both aircraft have well-damped phugoid modes at high speed. In the case of the coaxial-pusher, the short period mode at high speed is composed of two real poles, with one being unstable.

4. Finally, inherent inter-axis coupling is present in both designs despite their geometrically symmetric configurations. In the case of the coaxial-pusher, pitch-roll coupling is most pronounced at hover/low-speed and is significantly reduced at high speed, like typical helicopter configurations. In the case of the tiltrotor, large adverse roll-yaw coupling to rudder inputs is present at high speed due to the tail configuration. For both configurations, coupling should be addressed in the control system design.

Chapter 3 described the development and handling qualities based optimization of inner-loop flight control systems for both generic lift offset coaxial and tiltrotor aircraft. The feedback parameters were optimized using the Control Designer's Unified Interface (CONDUIT[®]) and design methods of Ref. 50. The control systems were optimized to meet a wide range of stability, handling-qualities, and performance specifications throughout the flight envelope of each aircraft. Stability specifications were taken from the general flight control system specification SAE AS94900 [97]. Handling-qualities specifications for hover/low-speed were taken from the Army's ADS-33E specification [98]. For mid- and high-speed, handling-qualities specifications from ADS-33E were used in addition to specifications from the DoD flying qualities specification MIL-STD-1797B [17]. Finally, performance specifications were chosen to ensure the control system would not saturate the actuators. The following conclusions are drawn:

1. The explicit model following architecture used for both aircraft worked well and met all Tier 1 design requirements throughout their large flight envelopes (hover-300 kts). This architecture made it simple to transition between response types, while maintaining similar performance of the feedback loop (crossover frequency and disturbance rejection bandwidth).
2. The multi-objective optimization method used to tune the control system parameters was able to determine Pareto optimum designs which concurrently met a large number (48) of frequency- and time-domain specifications while minimizing over-design (i.e., most economical use of actuators and noise sensitivity). Some overlap was identified between the Tier 1 requirements from ADS-33E and Tier 2 requirements from MIL-STD-1797B.
3. Ideal crossfeeds between commanded roll and pitch rate and differential cyclic commands determined from the differential flapping equations were used to develop a tip clearance feed-forward controller for the coaxial-pusher. This controller improved tip clearance performance by 90% during large amplitude maneuvers while not affecting the on-axis or off-axis responses of the aircraft.

Chapter 4 described the handling qualities assessment of the inner-loop control systems described in Chapter 3. A piloted simulation experiment was conducted in the NASA Ames Vertical Motion Simulator facility with participation from five Army experimental test pilots. Seven high-speed handling-qualities demonstration maneuvers were used: Pitch and Bank Attitude Capture and Hold, Pitch and Roll Sum-of-Sines Tracking, Break Turn, and High-Speed Acceleration and Deceleration. Overall, Level 1 handling qualities were assigned to the coaxial-pusher with inner-loop control laws in transition and high-speed, while the tiltrotor with inner-loop control laws was assigned Level 1 handling qualities at high-speed and Level 2 in transition. The results of the simulation experiment support the following conclusions:

1. The tip clearance feed-forward controller for the coaxial-pusher maintained > 1 ft separation between the upper and lower rotors during all MTEs tested. In addition, the coaxial-pusher collective trim map implementation resulted in very low pilot workload to maintain altitude while accelerating/decelerating through large speed ranges.
2. Turn coordination through the use of a sideslip-command/sideslip-hold response type in the yaw axis worked well for both aircraft. Pilots were able to complete the Break Turn maneuver without use of the pedals.
3. The Pitch and Bank Angle Capture and Hold, Pitch and Roll Sum-of-Sines Tracking, Break Turn, and High-Speed Acceleration/Deceleration MTEs worked well to assess the high-speed handling qualities of both aircraft tested. Pilots commented that the MTEs were appropriate, but noted that the Precision, Non-Aggressive version of the Pitch Sum-of-Sines Tracking tasks was too easy and that the Break Turn timing tolerance may be too tight.
4. When performing single flight condition tasks, the HQRs and performance between both aircraft were very similar (< 0.5 mean HQR difference between both aircraft). In addition, pilot control strategy was nearly identical as seen by the Pilot-Vehicle System analysis shown for the Pitch Sum-of-Sines Tracking task (Pilot-Vehicle System crossover frequency ω_c within 2% for both aircraft). These results validate that the quantitative specifications and mission task elements are mission-oriented and not configuration specific.

Chapter 5 discussed the development and optimization of full-flight envelope outer-loop control systems for both coaxial-pusher and tiltrotor aircraft. A dynamic inversion control system architecture was used for both aircraft, and was tuned to meet a common comprehensive set of stability, handling-qualities, and performance specifications. A lower-order (two state, three input) model was used to represent the airspeed V and climb rate $\dot{h}$ dynamics of the closed-loop bare-airframe and inner-loop control system. System identification of the closed-loop bare-airframe and inner-loop control system was used to determine the parameters of the lower-order model used

for dynamic inversion. Weighted pseudo inverse control allocation was used to allocate the outer-loop pitch attitude, symmetric main rotor collective, and pusher propeller collective command to the inner-loop control laws for the coaxial-pusher. In the case of the tiltrotor, the nacelle tilt angle was scheduled as a function of airspeed and commanded acceleration while the outer loops controlled pitch attitude and symmetric main rotor collective. Outer loop optimization results were shown for both aircraft at a range of airspeeds spanning their respective flight envelopes and demonstrated the predicted Level 1 handling qualities of both configurations. The following conclusions are drawn:

1. A simple two-state model well captures the airspeed and climb rate/flight path dynamics of both coaxial-pusher and tiltrotor aircraft with their respective inner-loop control systems closed. Using a system identification based method to determine the parameters of the two-state model led to the most accurate representation of the actual aircraft dynamics.
2. For the coaxial-pusher, weighted pseudo-inverse control allocation worked well to allocate airspeed and climb rate/flight path commands to pitch attitude, symmetric main rotor collective, and pusher propeller collective commands across the entire airspeed envelope.
3. A dynamic inversion architecture based on the identified two-state model worked well to control airspeed and climb rate/flight path throughout the full flight envelopes (hover-300 kts). Actual aircraft responses tracked the commanded responses well, with nearly no off-axis responses.
4. As with the inner-loop control system, the multi-objective optimization method used to tune the feedback gains of the outer-loop dynamic inversion control system proved capable of determining designs which concurrently met a large number of frequency- and time-domain specifications while minimizing over-design (i.e., most economical use of actuators and noise sensitivity).
5. A trade-off between pitch attitude bandwidth/dropback and flight path bandwidth was seen for both aircraft. The design space was more restricted for the tiltrotor with its higher value of flight path-attitude lag T_{θ_2} , which led to tiltrotor designs at low speed ($V_{CAS} = 60$ kts) with Level 2 pitch attitude bandwidth and designs in mid speed ($V_{CAS} = 120$ kts) with Level 2 dropback. At high speed (above $V_{CAS} = 180$ kts), all of the specification were met (Level 1).
6. For the inner-loop control laws, using a pitch attitude turn compensation command based on the normal acceleration required to coordinate a turn and the bare-airframe value of n/α reduced the amount of altitude loss during a turn. This is especially useful for control systems with integral error feedback gains on pitch attitude.

Chapter 6 described the handling qualities assessment of the outer-loop flight control systems described in Chapter 5. A piloted simulation experiment was conducted in the Pennsylvania State University (PSU) Flight Simulator facility with participation from two Army experimental test pilots. Five high-speed handling-qualities demonstration maneuvers were used: Pitch Attitude Capture and Hold, Pitch Sum-of-Sines Tracking, Break Turn, and High-Speed Acceleration and Deceleration. Overall, the outer-loop control laws for both coaxial-pusher and tiltrotor aircraft were assigned Level 1 handling qualities for the Break Turn and High-Speed Acceleration/Deceleration tasks, and reduced pilot workload over the inner-loop control laws. The outer-loop control laws also improved performance and reduced pilot workload in a formation flying task developed for this experiment. The coaxial-pusher outer loops received borderline Level 1/Level 2 ratings for the Pitch Attitude Capture and Hold and Sum-of-Sines Tracking tasks, while the tiltrotor outer-loop control system (with its increased value of pitch attitude dropback) received Level 2 ratings. The results of the simulation experiment support the following conclusions:

1. The outer-loop control laws significantly improved handling qualities (inner-loop control laws HQR 4 to outer-loop control laws HQR 1.5) and workload (inner-loop control laws BWLR 7 to outer-loop control laws BWLR 3) for the tiltrotor High-Speed Acceleration and Deceleration tasks by automatically managing nacelle tilt angle, symmetric rotor collective, and aircraft pitch attitude/angle of attack for the pilot. For the coaxial-pusher, with its direct thrust control, the outer-loops control laws had slightly better overall High-Speed Acceleration and Deceleration Handling Qualities Ratings than the inner-loop control laws.
2. Gain scheduling the outer-loop airspeed command model gain as a function of excess power additionally helped to reduce pilot workload during acceleration and deceleration.
3. Even though the lateral/directional axes of both inner- and outer-loop control laws were the same, the outer-loop control laws improved Break Turn performance and ratings by allowing the pilots to concentrate on the roll axis only by automatically regulating airspeed and altitude during the turn. Times to complete the maneuver were lower and all within desired performance when flown with the outer-loop control laws as compared to the inner-loop control laws.
4. The outer-loop control laws also improved the pilots' ability to get into formation with a simulated KC-130 aircraft, thus potentially improving handling qualities during an aerial refueling task.
5. A trade-off between the improved handling qualities and performance of the outer-loop control laws for the Break Turn, High-Speed Acceleration/Deceleration, and Formation Flying tasks was seen with the reduced performance and handling

qualities during Pitch Sum-of-Sines Tracking and Attitude Capture and Hold tasks. This was due to the increased level of pitch attitude dropback of the outer-loop flight path rate command response type as compared to the inner-loop pitch rate command response type.

6. Overall, pilots preferred the flight path Rate-Command/Attitude-Hold (RCAH) response type of the outer-loops for gross maneuvering and flight path control, while for pitch tracking/pointing tasks they preferred a pitch RCAH response type. This suggests that the outer-loops should be implemented as a pilot selectable mode, which the pilot can engage based on the task they are performing (e.g., flight path vs. pitch attitude control).

Chapter 7 described an investigation into the effect of pitch response type and command model order on handling qualities for the generic lift offset coaxial and tiltrotor rotorcraft. A piloted simulation experiment was conducted in the PSU Flight Simulator facility with participation from two Army experimental test pilots. Two pitch command model forms were investigated: 1) a first-order transfer function typically used for rotorcraft RCAH response types, and 2) a first-over-second order transfer function based on the form of a conventional airplane short-term pitch rate transfer function. Designs of varying levels of pitch attitude bandwidth, flight path bandwidth, CAP, and pitch attitude dropback were evaluated for both coaxial-pusher and tiltrotor. The results of the simulation experiment support the following conclusions:

1. For the non-aggressive version of the Pitch Sum-of-Sines Tracking tasks, the ADS-33E Pitch Attitude Bandwidth requirement with Target Acquisition & Tracking boundaries matches the assigned pilot Handling Qualities Ratings well. The MIL-STD-1797B pitch attitude bandwidth requirement with Category A boundaries seems to be too stringent. However, if designing for the aggressive version of the Pitch Sum-of-Sines Tracking tasks, the MIL-STD-1797B pitch attitude bandwidth requirement with Category A boundaries should be included as a Tier 1 requirement (enforced during control system optimization).
2. For both tasks tested, values of pitch attitude dropback that are within Level 1 of the pitch attitude dropback specification received Level 2 takings. The pilot ratings and comments suggest that an upper boundary lies between $DB_{\theta}/q_{ss} = 0.3 - 0.8 \text{ sec}^{-1}$, depending on the task.
3. A flight path-oriented task such as formation flying is needed to assess the location of the flight path bandwidth requirement level boundaries, as even designs with low values of flight path bandwidth received Level 1 rating for pitch tracking tasks.
4. The MIL-STD-1797B Control Anticipation Parameter (CAP) specification is best kept as a check-only (Tier 2) requirement in control law design. Designs that

violate the lower CAP limit also violate the pitch attitude bandwidth requirement and designs that violate the upper equivalent short period damping boundary still received Level 1 ratings. The lower equivalent short period damping boundary is $\zeta_{sp} = 0.35$ which matches the ADS-33E damping requirement and is therefore redundant.

Chapter 8 described the robustness analysis of the coaxial-pusher and tiltrotor inner- and outer-loop control laws. To make the robustness analysis more computationally efficient for the coaxial-pusher and tiltrotor which both have a large number of bare-airframe parameters that can be perturbed and a large number of stability, handling-qualities, and performance requirements that need to be evaluated, an Unscented Transform approach was used instead of a more traditional Monte Carlo analysis. In addition, only the fuselage stability and control derivatives were perturbed in the analyses performed here, since those have the largest effect on the aircraft response in the frequency range where the control system is active ($\omega \lesssim 10$ rad/sec). To accurately assess the robustness of the control laws, the fuselage derivatives of the full-order bare-airframe models were varied by perturbations with standard deviation σ equal to 10% of their quasi-steady (six degree-of-freedom) values. The results of the robustness analysis support the following conclusions:

1. The Unscented Transform provided an accurate and efficient method to assess control system stability and performance robustness by passing the bare-airframe parameter statistics through the nonlinear transformation given by evaluation of the design specifications. For the UH-60 roll rate example shown, resulting covariance matrices of the Monte Carlo and Unscented Transform analyses differed by less than 5%.
2. The inner- and outer-loop control laws for both aircraft were shown to have good stability and performance robustness with $2\sigma - 3\sigma$ confidence ellipsoids within the Level 1 region of most specifications, covering 95% – 99.7% of perturbation cases, corresponding to 20% – 30% variations in the model parameter values.
3. The performance robustness of the outer-loop control system, as captured by the model following specifications, was improved over that on the inner-loop control system for both coaxial-pusher and tiltrotor. Overall, maximum 3σ model following costs were reduced from $J_{MF} \approx 100 - 120$ for the inner loop to $J_{MF} \approx 30 - 60$ for the outer loop. This is because the inner-loop control system suppresses the variations in the bare-airframe below the inner-loop crossover frequency and provides the outer-loop control system with less variations in the aircraft response.

9.2 Control System Design Guidance

Based on the results of both inner- and outer-loop control law simulation evaluations, the following control system design guidance is provided for advanced high-speed rotorcraft:

1. Control allocation using a weighted pseudo inverse provides the ideal swashplate phase angle for rigid coaxial rotors at each airspeed. Even if the pseudo inverse control allocation solution is not used directly in the control system, it can be used to develop a swashplate phase angle airspeed schedule which is implemented in the control system.
2. An explicit model following control system architecture worked well across the entire flight envelope, for both hover and forward flight response types. Lower-order on-axis inverse models were used in the control system with excellent model following performance attained. The coaxial-pusher configuration required a first-over-first order roll-axis inverse model in hover/low-speed due to the effect of the rigid rotors on the phase of the roll rate frequency response. A conventional fixed-wing first-over-second order pitch rate inverse worked well at high speed for both configurations.
3. A collective trim map should be included in the control system for both configurations. In the case of the coaxial-pusher, it helped reduce pilot work load in transition and high-speed by eliminating the need for collective inputs. Pilots were able to use the pusher propeller to control thrust and pitch attitude to control flight path. In the case of the tiltrotor, a collective trim map as a function of nacelle angle can make transition easier by improving the vertical response to nacelle tilt angle.
4. The control system for a coaxial rotor configuration should be able to command differential lateral and longitudinal cyclic. These were used for tip clearance maintenance during maneuvering flight using a feed forward controller. The tip clearance controller can be implemented in the feedback to also account for responses to gusts and turbulence.
5. Dynamic inversion outer-loop control laws based on a simple two-state model of the inner-loop airspeed and climb rate/flight path response worked well and provided Level 1 handling qualities for gross maneuvers. For the coaxial-pusher, a weighted pseudo inverse control allocation scheme worked well throughout the entire flight envelope to allocate the outer loop commands to pitch attitude, symmetric rotor collective, and pusher propeller collective. For the tiltrotor, having a nacelle angle schedule based on airspeed and commanded acceleration worked well to maintain reasonable pitch attitudes during maximum acceleration and deceleration through the conversion corridor.

6. Performance of both the inner- and outer-loop control laws for both aircraft validates the specification-driven multi-objective optimization approach to flight control design used here. Appendix D lists the recommended specification for inner- and outer-loop control design.

Appendix A |

Mission Task Element Description and Performance Criteria

This appendix provides a description and lists the task performance requirements for the Mission Task Elements (MTEs) used to assess handling qualities.

A.1 Break Turn

A.1.1 Objectives

- Ensure that handling qualities do not degrade during aggressive maneuvering in all axes.
- Check for undesirable coupling between pitch, roll, and yaw during aggressive maneuvering.

A.1.2 Description of Maneuver

Starting at airspeed of $0.8V_H$ and AGL of 300–1000 ft, accomplish a 90 deg change in directional flight path and achieve wings-level attitude in as little time as possible. Some overshoot of the 90 deg flight path change is allowed (up to tolerances listed below), but flight path change must be at least 85 deg. Some airspeed and altitude variations are allowed up to tolerances listed below. Use of pedals to induce lateral acceleration in the direction of the turn is acceptable. Use of auxiliary propulsion controls to regulate airspeed is acceptable. Perform the maneuver both to the right and the left, with the maneuver flown from the pilot seat that provides sufficient cueing to each side.

A.1.3 Description of Test Course

Setting up 90 degrees to a linear geographical feature (road, power lines, etc.) or runway and using it as a visual cue for the roll out portion of the maneuver is

recommended, as shown in Figure A.1. The use of two perpendicular reference lines (roads, runways, etc.) can also be used to provide guidance on both initial and final directional flight path. Setting the “heading bug” at the final heading position can provide an additional useful cue to the pilot.

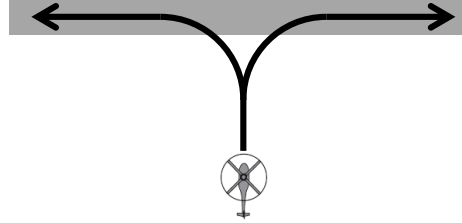


Figure A.1. Suggest Break Turn MTE course cueing.

A.1.4 Performance Standards

Table A.1 lists the performance standards for the Break Turn MTE.

Table A.1. Break Turn Performance Criteria

	Desired	Adequate
• Complete maneuver within time $T < \Delta T + T_{\text{ideal}}$	$\Delta T = 3.5 \text{ sec}$	$\Delta T = 7.0 \text{ sec}$
• Final change in directional flight path shall be at least 85 deg and no more than X degrees	95 deg	105 deg
• When rolling out to wings-level attitude, the overshoot in roll attitude shall not exceed X degrees	5 deg	10 deg
• Final airspeed loss shall be no more than X% of initial airspeed (0.8VH)	10%	20%
• Maintain altitude within $\pm X$ feet	75 ft	150 ft
• Any oscillations or interaxis coupling shall not be	Undesirable	Objectionable

A.1.5 Timing Requirement

The Break Turn MTE requires the calculation of an ideal time to complete the maneuver T_{ideal} , based on the aircraft operational flight envelope (OFE) normal load factor limit $n_{z_{\text{lim}}}$:

$$T_{\text{ideal}} = \frac{\pi V(45 - \Delta\psi_{\text{transient}})}{90g\sqrt{n_{z_{\text{lim}}}^2 - 1}} + 2T_{\text{transient}} \quad (\text{A.1})$$

where:

- V is the true airspeed at which the maneuver is performed in units of ft/sec,
- $n_{z\text{lim}}$ is the OFE normal load factor limit in units of g,
- $T_{\text{transient}}$ time needed for one transient phase (roll-in or roll-out) of the maneuver,
- and
- $\Delta\psi_{\text{transient}}$ is the heading change during the transient phase in units of deg.

Details of determining $T_{\text{transient}}$ and $\Delta\psi_{\text{transient}}$ are provided in Ref. 42, and are expanded on here.

The transient phase of the maneuver is assumed to happen at constant roll acceleration $\dot{p}_0$. The roll acceleration is calculated to meet the agility requirement of the MIL-STD-1797B [17] time to bank specification, which specifies a required bank angle to reach ϕ_{req} in a required time t_{req} based on aircraft Class and Flight Phase Category.

Aircraft Classes in MIL-STD-1797B are used to particularize the requirements according to broad categories of intended use, and are related qualitatively to maximum design gross weight and limit load factor [17]. The following are the definitions for Classes I–IV as defined in MIL-STD-1797B:

- **Class I:** Small, light air vehicles (e.g., light utility, primary trainer, or light observation)
- **Class II:** Medium weight, low-to-medium maneuverability air vehicles (e.g., heavy utility, light or medium transport/cargo, tactical bomber, heavy attack)
- **Class III:** Large, heavy, low-to-medium maneuverability air vehicles (e.g., heavy transport/cargo/tanker, heavy bomber)
- **Class IV:** High maneuverability air vehicles (e.g., fighter/interceptor, attack, tactical reconnaissance)

Flight Phase Categories are used to particularize the requirements according to the type of tasks that need to be accomplished. The following are the definitions for Flight Phase Categories A–C as defined in MIL-STD-1797B:

- **Category A:** Nonterminal flight phases that require rapid maneuvering, precise tracking, or precise flight-path control (e.g., air-to-air combat, terrain following, close formation flying, precision hover)
- **Category B:** Nonterminal flight phases that are normally accomplished using gradual maneuvers and without precision tracking (e.g., climb, cruise, loiter)
- **Category C:** Terminal flight phases that are normally accomplished using gradual maneuvers and usually require accurate flight-path control (e.g., takeoff, power approach, landing)

Since the Break Turn MTE is an aggressive task, the time to bank requirements for Category A should be used, and are reproduced here in Table A.2.

Based on the appropriate time to bank requirement $(\phi_{\text{req}}, t_{\text{req}})$, a constant roll acceleration $\dot{p}_0$ is calculated as:

$$\dot{p}_0 = \frac{2\phi_{\text{req}}}{t_{\text{req}}^2} \quad (\text{A.2})$$

The corresponding roll accelerations are also provided in Table A.2.

A limit bank angle ϕ_{lim} is calculated based on maintaining level flight at the OFE normal load factor limit $n_{z\text{lim}}$:

$$\phi_{\text{lim}} = \cos^{-1} \left(\frac{1}{n_{z\text{lim}}} \right) \quad (\text{A.3})$$

Then, the time to reach the limit bank angle ϕ_{lim} assuming constant positive $\dot{p}_0$ for half that time and constant negative $\dot{p}_0$ for the second half of the time is defined as the time for one transient (roll-in or roll-out) $T_{\text{transient}}$ and is given by:

$$T_{\text{transient}} = 2\sqrt{\phi_{\text{lim}}/\dot{p}_0} \quad (\text{A.4})$$

To determine the change in heading during this transient phase $\Delta\psi_{\text{transient}}$, an equation for the bank angle as a function of time during the transient phase is constructed as:

$$\phi(t) = \begin{cases} \frac{1}{2}\dot{p}_0 t^2, & 0 \leq t \leq \frac{T_{\text{transient}}}{2} \\ -\frac{1}{2}\dot{p}_0 \left(t - \frac{T_{\text{transient}}}{2}\right)^2 + \dot{p}_0 \frac{T_{\text{transient}}}{2} \left(t - \frac{T_{\text{transient}}}{2}\right) + \frac{1}{2}\dot{p}_0 \left(\frac{T_{\text{transient}}}{2}\right)^2, & \frac{T_{\text{transient}}}{2} < t \leq T_{\text{transient}} \end{cases} \quad (\text{A.5})$$

Next, assuming coordinated flight $[\dot{\psi} = (g/V) \tan \phi]$, the heading change $\Delta\psi_{\text{transient}}$ is given by:

$$\Delta\psi_{\text{transient}} = \frac{180}{\pi} \int_0^{T_{\text{transient}}} \frac{g}{V} \tan \left[\frac{\pi}{180} \phi(t) \right] dt \quad (\text{A.6})$$

where $\Delta\psi_{\text{transient}}$ and ϕ are in units of deg.

Unfortunately, there is no analytical solution to Eq. A.6, and it must be solved numerically. However, an approximation to Eq. A.6 was determined in this study based on the first two terms in the Taylor expansion of $\tan \phi$ and a correction factor for large bank angles (where the Taylor approximation is not as good):

$$\Delta\psi_{\text{transient}} \approx \frac{\frac{g}{V} [0.125(\dot{p}_0 T_{\text{transient}}^3) + 5.2 \times 10^{-7}(\dot{p}_0^3 T_{\text{transient}}^7)]}{(1 - 8.0 \times 10^{-9} \phi_{\text{lim}}^4)} \quad (\text{A.7})$$

where $\Delta\psi_{\text{transient}}$ and ϕ_{lim} are in units of deg and $\dot{p}_0$ is in units of deg/sec².

Figure A.2 shows a sample time history of roll rate p , bank angle ϕ , heading rate $\dot{\psi}$, and heading ψ for the T_{ideal} calculation.

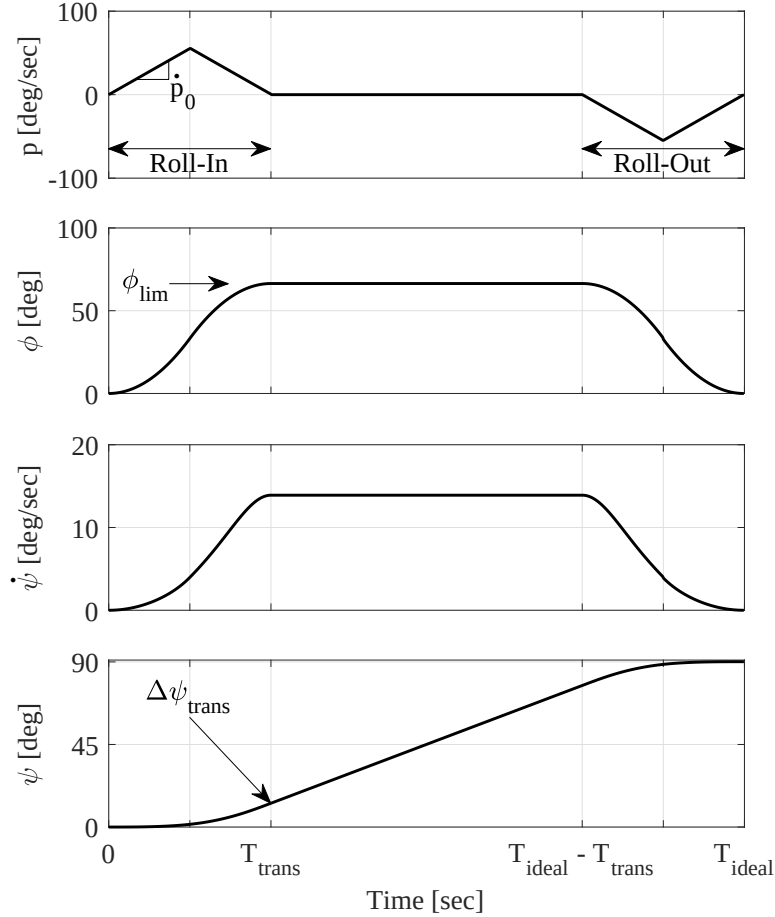


Figure A.2. Break Turn ideal time to complete T_{ideal} calculation time history.

Figure A.3 shows the T_{ideal} values for a range of time to bank requirements, airspeeds, and normal load factor limits. The results were generated by solving Eq. A.6 numerically (labeled “Actual” in Figure A.3) and by using the approximation in Eq. A.7 (labeled “Approx” in Figure A.3). The results generated using the approximation in Eq. A.7 match the actual results almost perfectly.

To summarize, the steps to calculate T_{ideal} are as follows:

1. Determine the true airspeed at which the maneuver is performed V in units of ft/sec and the OFE normal load factor limit $n_{z\text{lim}}$ in units of g
2. Look up roll acceleration $\dot{p}_0$ to meet the appropriate MIL-STD-1797B time to bank requirement in Table A.2

3. Calculate ϕ_{lim} using Eq. A.3
4. Calculate $T_{\text{transient}}$ using Eq. A.4
5. Calculate $\Delta\psi_{\text{transient}}$ using Eq. A.7
6. Calculate T_{ideal} using Eq. A.1

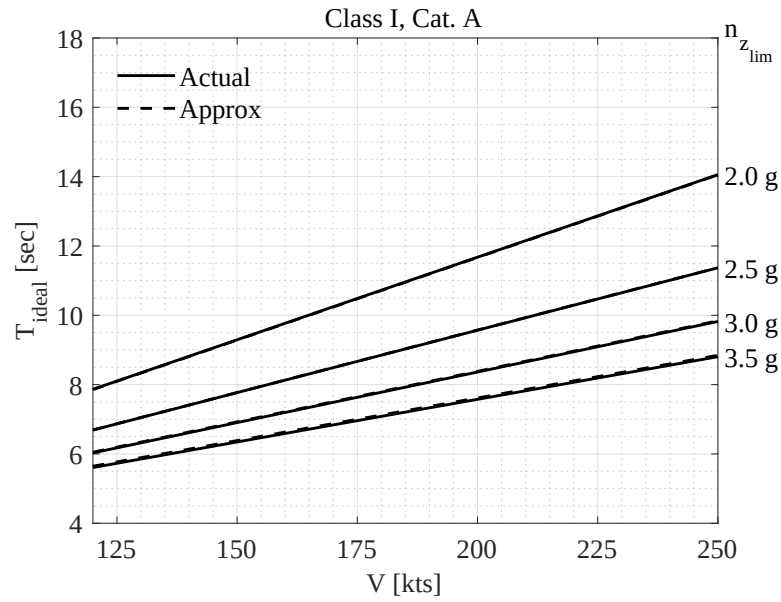


Figure A.3. Break Turn ideal time to complete T_{ideal} values for Aircraft Classes I, Flight Phase Category A.

Table A.2. Time to Bank Requirements from MIL-STD-1797B (Category A, High Speed Range) and Corresponding Roll Accelerations

Class	Required Bank Angle ϕ_{req} [deg]	Required Time t_{req} [sec]	Roll Acceleration $\dot{p}_0$ [deg/sec ²]
I	60	1.3	71.0
II	45	1.4	45.9
III	30	2.0	15.0
IV	50	1.1	82.6

A.2 High-Speed Acceleration/Deceleration

A.2.1 Objectives

- Check pitch and heave axes handling qualities for aggressive maneuvering throughout the speed range envelope.
- Check for undesirable coupling between the longitudinal, lateral-directional, and unconventional (e.g., tilt rotor/wing, if applicable) axes.
- Ensure that handling qualities do not degrade during transitional flight regimes (e.g., rotor-borne to wing-borne flight), if applicable.
- Check for harmony between the heave and pitch axis controllers, as well as unconventional/auxiliary control axes/inceptors, if applicable.

A.2.2 Description of Maneuver

Prior to the maneuver for score, the Minimum Acceleration and Deceleration Times (maximum performance) must be determined. This can be done analytically (via offline simulation, performance charts, or other means) or empirically (via real-time piloted simulation or in flight). When determining the minimum time the other task tolerances should be used as a guide but do not have to be adhered to strictly. While the tolerances are used as guides, altitude and heading should not be exchanged for performance. Each phase can have a different Minimum Time.

For scoring evaluation, from 50 knots forward flight at altitude, initiate a maximum performance constant altitude acceleration to capture $V_H - 10$ knots. The second portion of the maneuver is to initiate a maximum-performance deceleration from $V_H - 10$ knots while holding altitude constant until 50 kts is captured. Each phase will be rated separately. Automation of unconventional/auxiliary control axes is acceptable.

A.2.3 Description of Test Course

No special test course is required, although the use of a linear geographical feature (road, power lines, etc.) is recommended to provide lateral cueing.

A.2.4 Performance Standards

Table A.3 lists the performance standards for the High-Speed Acceleration/Deceleration MTE.

Table A.3. High-Speed Acceleration/Deceleration Performance Criteria

	Desired	Adequate
• Complete acceleration within time $T < \Delta T + T_{\text{minimum}}$	$\Delta T = 8.0$ sec	$\Delta T = 15.0$ sec
• Complete deceleration within time $T < \Delta T + T_{\text{minimum}}$	$\Delta T = 10.0$ sec	$\Delta T = 20.0$ sec
• Final airspeed to be captured within:	± 2 kts	± 3 kts
• Maintain altitude within X feet of initial altitude:	± 100 ft	± 150 ft
• Maintain heading within X degrees of initial heading:	± 5 deg	± 10 deg
• Maintain bank angle (from trim) within:	± 5 deg	± 10 deg
• Any oscillations or inter-axis coupling shall not be:	Undesirable	Objectionable
• Control harmony between axes shall not be:	Undesirable	Objectionable
• Rotor RPM shall remain within the limits of X without undue pilot compensation:	OFE	SFE

A.2.5 Timing Requirement

The High-Speed Acceleration/Deceleration MTE requires the calculation of a minimum time to complete each portion of the MTE (accelerate and decelerate) T_{min} . The minimum time is based on the maximum performance, and can be determined either empirically (via real-time piloted simulation or in flight) or analytically (via offline simulation, performance charts, or other means). An analytical method is presented here which uses performance charts/power curves.

The analytical method presented here is based on energy methods. Using a simplified energy model, the total energy of the helicopter E is a combination of its potential and kinetic energies:

$$E = Wh + \frac{1}{2} \left(\frac{W}{g} \right) V^2 \quad (\text{A.8})$$

where:

- W is the weight of the aircraft [lbs],
- h is the altitude of the aircraft [ft],
- g is the acceleration due to gravity [32.174 ft/sec²], and
- V is the airspeed [ft/sec].

The rate of change of energy $\dot{E}$ is equal to the change in power ΔP , and is given by:

$$\Delta P = \dot{E} = W\dot{h} + \frac{W}{g}V\dot{V} \quad (\text{A.9})$$

where ΔP is in units of ft-lbs/sec.

Assuming a constant level acceleration (i.e., $\dot{h} = 0$), the velocity profile for a given change in power ΔP is given by:

$$\dot{V} = \frac{\Delta P}{(W/g)V} \quad (\text{A.10})$$

For a maximum performance acceleration, the velocity profile can be calculated based on the maximum continuous power available P_{MCP} and the power required curve P_{req} which is a function of airspeed $P_{\text{req}}(V)$:

$$\dot{V} = \frac{P_{\text{MCP}} - P_{\text{req}}(V)}{(W/g)V} \quad (\text{A.11})$$

For a maximum performance deceleration, the velocity profile can be calculated based on the power required curve $P_{\text{req}}(V)$ and a minimum power setting used to decelerate P_{min} :

$$\dot{V} = \frac{P_{\text{min}} - P_{\text{req}}(V)}{(W/g)V} \quad (\text{A.12})$$

A.3 Pitch/Roll Sum-of-Sines Tracking

A.3.1 Objectives

- Evaluate handling qualities in a tight, closed-loop tracking task.
- Evaluate feel system and control sensitivity characteristics.
- Identify bobble or Pilot Induced Oscillation (PIO) tendencies.

A.3.2 Description of Maneuver

This task is driven by an automated command signal selected by the flight test engineer. From steady, wings level flight aggressively track the displayed signal and attempt to keep errors within the specified tolerances.

A.3.3 Description of Test Course

This maneuver does not require a test course. The maneuver does require a display with the desired and adequate performance criteria displayed to the pilot (example on next page). Examples of pitch and roll angle command signals are shown in Figure A.4.

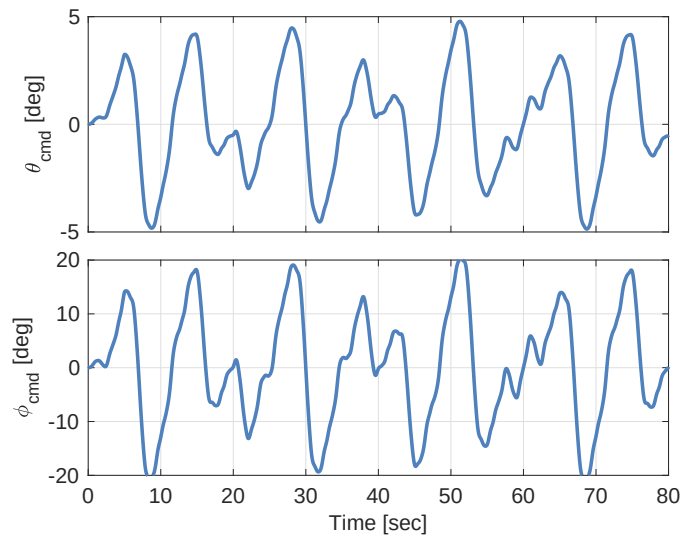


Figure A.4. Example sum-of-sines tracking signals.

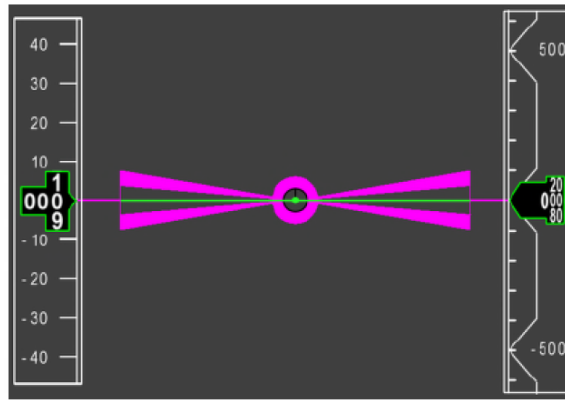
The length for scoring time is recommended to be 60 seconds.

A.3.4 Performance Standards

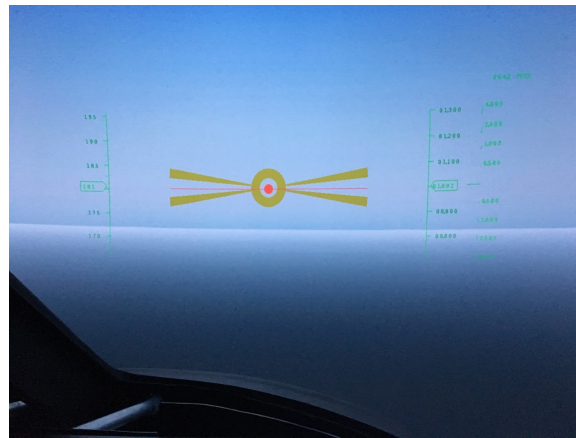
Table A.4 lists the desired and adequate performance criteria for the Pitch/Roll Sum-of-Sines Tracking tasks.

Table A.4. Pitch/Roll Sum-of-Sines Tracking Performance Criteria

	Desired	Adequate
<i>Pitch</i>		
• Remain within ± 1 deg for X percent of the time • Remain within ± 2 deg for X percent of the time • PIO Considerations:	50 % - No PIO tendencies	- 75 % No divergent PIO tendencies
• Inter-axis coupling shall not be	Undesirable	Objectionable
<i>Roll</i>		
• Remain within ± 5 deg for X percent of the time • Remain within ± 10 deg for X percent of the time • PIO Considerations:	50 % - No PIO tendencies	- 75 % No divergent PIO tendencies
• Inter-axis coupling shall not be	Undesirable	Objectionable



(a) VMS



(b) PSU Simulator

Figure A.5. Display for Sum-of-Sines Tracking and Attitude Capture and Hold tasks used in (a) the VMS and (b) the PSU Simulator.

A.4 Pitch/Bank Angle Capture and Hold

A.4.1 Objectives

- Evaluate ability to pitch or roll and capture a desired attitude angle.
- Identify maneuverability limitations and Pilot Induced Oscillation (PIO) tendencies.

A.4.2 Description of Maneuver

The attitude capture and hold tasks are precision, non-aggressive maneuvers. The tasks are flown using the display shown in Figure A.5, which is driven by a reference signal composed of a series of step changes in attitude. From steady, wings level flight the aircraft is pitched or banked to capture and maintain the commanded angle within

the specified tolerance for 5 sec. Pitch attitudes of ± 5 deg are used and roll attitudes of ± 30 deg are used.

A.4.3 Description of Test Course

This maneuver does not require a test course. The maneuver does require a display with the desired and adequate performance criteria displayed to the pilot. Examples of pitch and roll angle command signals are shown in Figure A.6.

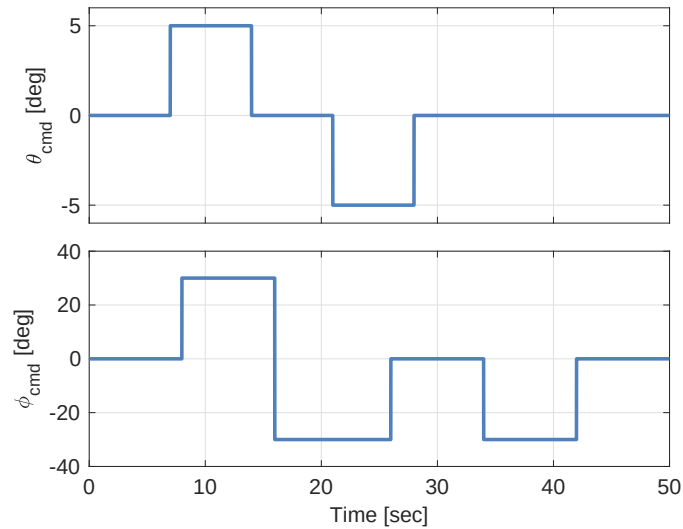


Figure A.6. Example capture and hold tracking signals.

A.4.4 Performance Standards

Table A.5 lists the desired and adequate performance criteria for the Pitch/Bank Angle Capture and Hold tasks.

Table A.5. Pitch/Bank Angle Capture and Hold Performance Criteria

	Desired	Adequate
<i>Pitch</i>		
• Pitch angle error (from command) tolerance:	± 1 deg	± 2 deg
• Airspeed deviation tolerance:	± 5 kts	± 10 kts
• No more than one pitch attitude overshoot on the initial capture of each attitude. Magnitude of overshoot is less than:	1 deg	2 deg
• PIO considerations:	No PIO tendencies	No divergent PIO tendencies
• Inter-axis coupling shall not be	Undesirable	Objectionable
<i>Roll</i>		
• Bank angle error (from command) tolerance:	± 5 deg	± 10 deg
• Airspeed deviation tolerance:	± 5 kts	± 10 kts
• No more than one bank angle overshoot on the initial capture of each attitude. Magnitude of overshoot is less than:	5 deg	10 deg
• PIO considerations:	No PIO tendencies	No divergent PIO tendencies
• Inter-axis coupling shall not be	Undesirable	Objectionable

Appendix B |

Pilot Questionnaire and Rating Scales

This appendix provides the questionnaire and rating scales that were provided to the pilots during the simulation handling qualities experiments. Pilots were first asked to answer the questionnaire to provide a structured way to comment about the handling qualities, inceptor characteristics, and MTEs. Subsequently, pilots were asked to work their way through the Cooper-Harper Rating Scale decision tree to provide their Handling Qualities Rating (HQR). In general, half ratings (e.g., 2.5) were not accepted from the pilots with one exception: 4.5. A rating of 4.5 was allowed for cases where desired performance was attainable but with consideration pilot workload, or when adequate performance was attainable with moderate pilot workload.

Task Performance

1. Describe ability to meet DESIRED / ADEQUATE performance standards.
2. Describe ability to be AGGRESSIVE / PRECISE when performing the task.
3. If trying for DESIRED performance resulted in unacceptable oscillations, did decreasing your goal to ADEQUATE performance alleviate the problem?
4. Please comment on the appropriateness of the MTE task setup and performance standards.

Aircraft Characteristics

5. Describe any objectionable controller characteristics (e.g. force, sensitivity, etc.).
6. Describe any non-linearity of response, objectionable oscillations, or tendency to overshoot (if PIO prone provide rating using PIO scale).
7. Describe any problems with harmony of pitch and roll, speed control, with height control, and with heading hold/turn coordination.
8. Describe ride quality when performing the task.

Demands on the Pilot

9. Describe overall control strategy in performing the task (cues used, scan, etc.)

10. Describe any control compensation you had to make you to account for deficiencies in the aircraft.
11. Describe any modifications you had to make to what you would consider “normal” control technique in order to make the aircraft behave the way you wanted.
12. Please comment on any outer loop control law enhancements that would reduce pilot workload when performing the task (e.g. position hold, velocity hold, etc.).

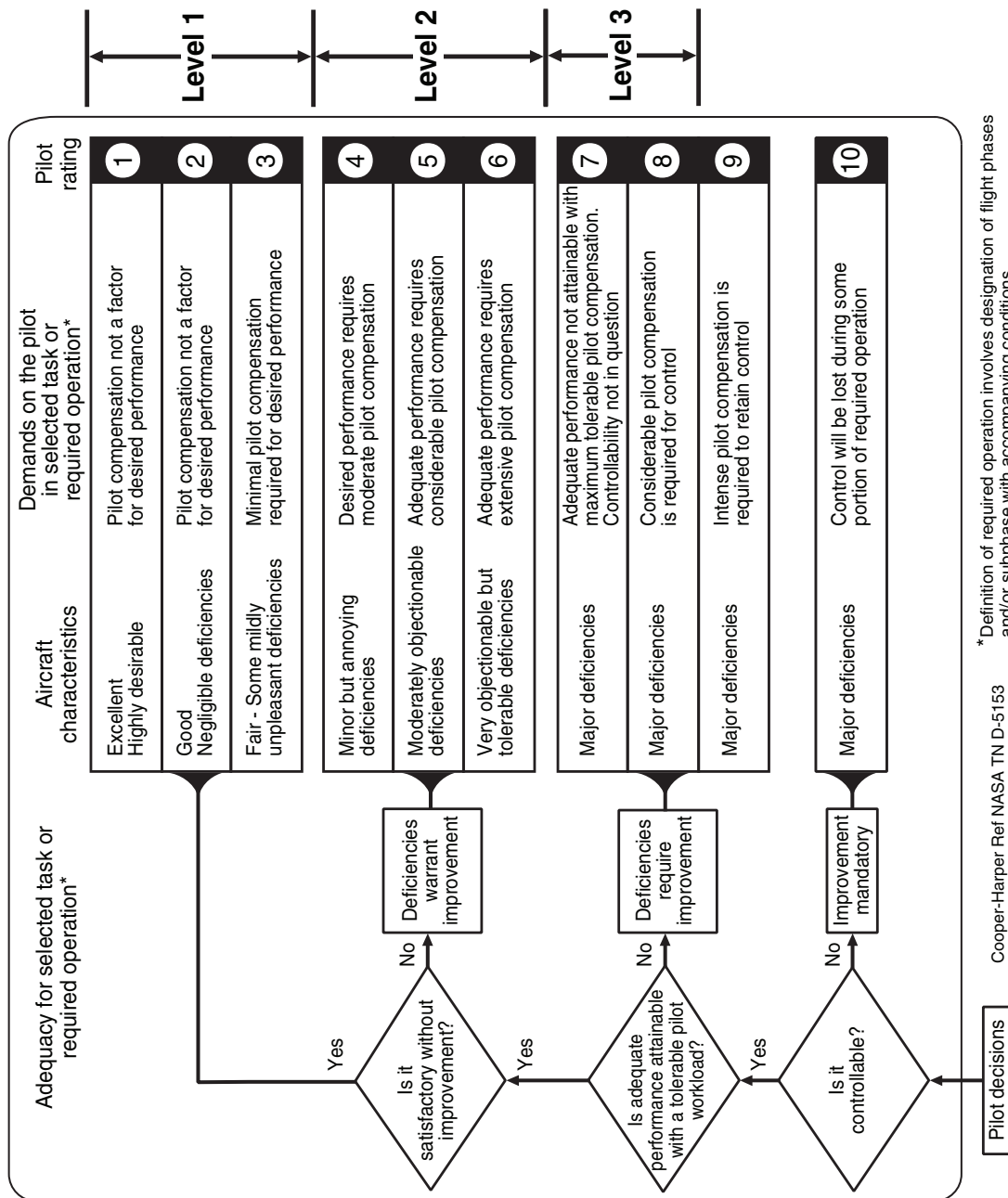
MISC.

13. Please comment on anything else that may have influenced you.

ASSIGN HANDLING QUALITIES RATING FOR OVERALL TASK

14. Using the Cooper-Harper rating scale, please highlight your decision-making process and adjectives that are best suited in the context of the task. If assigned HQR is Level 2, briefly summarize any deficiencies that make this configuration unsuitable for normal accomplishment of this task, i.e., justify why the procuring activity should reject this configuration as a means to accomplish this task.
15. What was the critical sub-phase of the task (e.g., entry, steady-state, exit) or major determining factor in the overall Handling Quality Rating (HQR).

Figure B.1. Pilot questionnaire.



*Definition of required operation involves designation of flight phases and/or subphase with accompanying conditions.

Cooper-Harper Ref NASA TN D-5153

Figure B.2. Cooper Harper Handling Qualities Rating Scale.

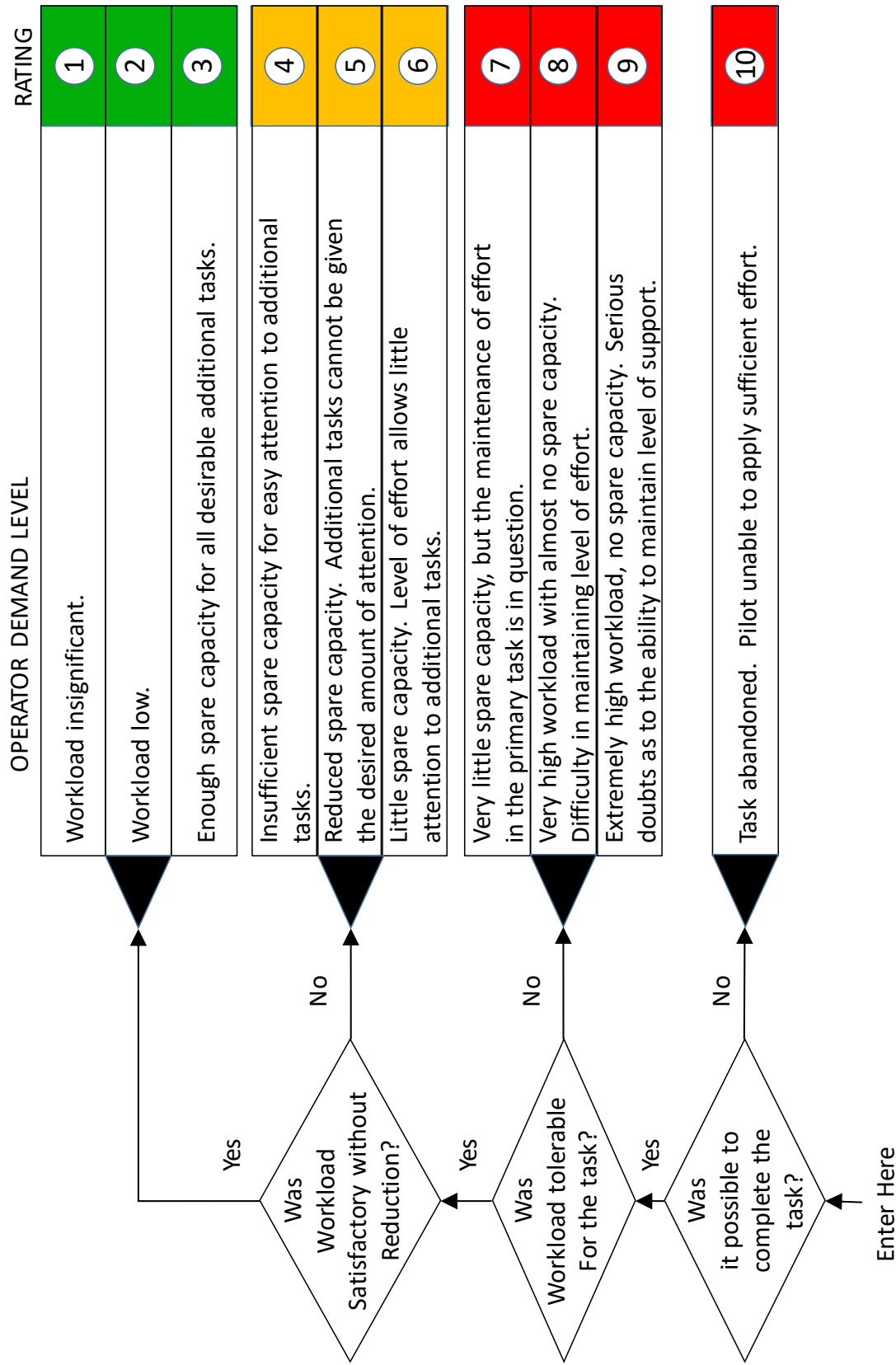


Figure B.3. Bedford Workload Rating Scale.

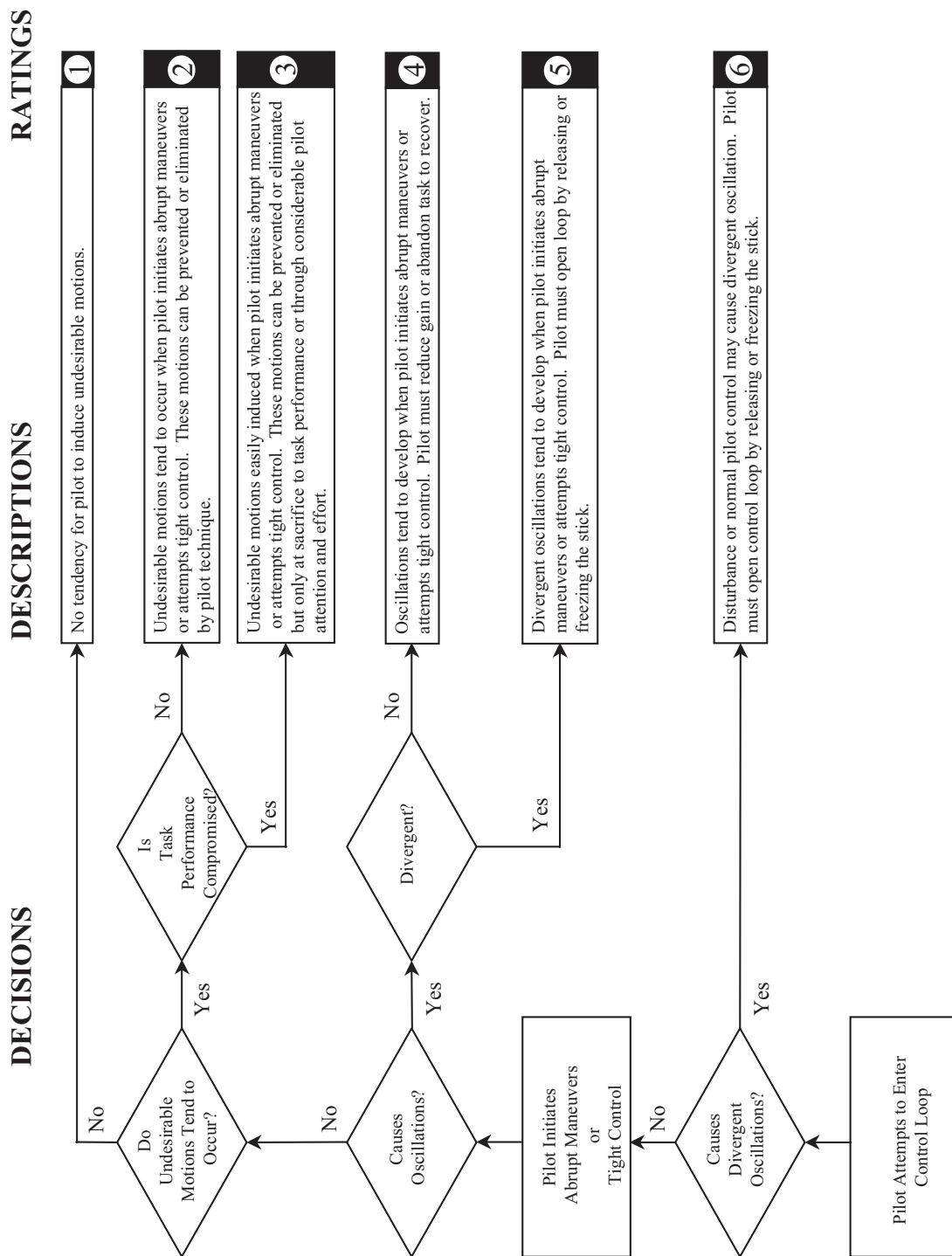


Figure B.4. Pilot Induced Oscillation Rating Scale.

Appendix C |

Command Model Tuning

Given an estimate of the overall forward path time delay τ and the order of the command model, it is possible to tune the parameters of the command model to achieve a desired attitude phase bandwidth ω_{BW} (based on the ADS-33E definition of bandwidth, i.e., where the phase curve of the attitude response crosses $\varphi = -135$ deg). Equations to solve for the parameters of the command model were developed for first-order and first-over-second-order rate command response type command models and a second-order attitude command response type command model.

C.1 First-Order Command Model (Rate Command)

Given the first-order pitch rate command model:

$$\frac{q_{cm}}{\delta_{lon_s}} = \frac{K a}{s + a} \quad (C.1)$$

overall forward path delay τ and desired pitch attitude bandwidth ω_{BW} , the command model time constant a is given by:

$$a = \frac{-\omega_{BW}}{\tan(\tau\omega_{BW} - \pi/4)} \quad (C.2)$$

C.2 Second-Order Command Model (Attitude Command)

Given the second-order pitch attitude command model:

$$\frac{\theta_{cm}}{\delta_{lon_s}} = \frac{K\omega_n^2}{s^2 + 2\zeta\omega_n s + \omega_n^2} \quad (C.3)$$

overall forward path delay τ , damping ratio ζ , and desired pitch attitude bandwidth ω_{BW} , the command model natural frequency ω_n is given by:

$$\omega_n = \frac{-\omega_{\text{BW}} \left(\zeta - \sqrt{\zeta^2 + \tan^2(\tau\omega_{\text{BW}} - 3\pi/4)} \right)}{\tan(\tau\omega_{\text{BW}} - 3\pi/4)} \quad (\text{C.4})$$

C.3 First-Over-Second-Order Command Model (Rate Command)

Given the first-over-second-order pitch rate command model:

$$\frac{q_{\text{cm}}}{\delta_{\text{longs}}} = \frac{K\omega_n^2(\frac{1}{b}s + 1)}{s^2 + 2\zeta\omega_n s + \omega_n^2} \quad (\text{C.5})$$

overall forward path delay τ , damping ratio ζ , numerator zero b , and desired pitch attitude bandwidth ω_{BW} , the command model natural frequency ω_n is given by:

$$\omega_n = \frac{-\omega_{\text{BW}} \left(\zeta - \sqrt{\zeta^2 + \tan^2[\tau\omega_{\text{BW}} - \pi/4 - \tan^{-1}(\omega_{\text{BW}}/b)]} \right)}{\tan(\tau\omega_{\text{BW}} - \pi/4 - \tan^{-1}(\omega_{\text{BW}}/b))} \quad (\text{C.6})$$

Appendix D | Recommended Handling Qualities Specifications

This appendix provides tables of recommended specifications for optimization of high-speed rotorcraft control laws for high-speed rotorcraft. Table D.1 lists specifications for optimization of inner-loop control laws, while Table D.2 lists specifications for optimization of outer-loop control laws.

The tables list the specification CONDUIT[®] names, a brief description and motivation for including each specification, the control axis that each specification is applied to, and the source for each specification.

Note that the CONDUIT[®] specifications listed in the tables have boundaries corresponding to Target Acquisition & Tracking for ADS-33E specifications and Category A, Class I for MIL-STD-1797B specifications, as these were used to optimize the control systems of the aircraft presented here based on their required mission. For other applications, this set of specifications is recommended, but it is up to the designer to select the appropriate boundaries based on the required mission (e.g., All Other MTEs for ADS-33E specifications or Class II for MIL-STD-1797B specifications).

Table D.1: Recommended Inner-Loop Handling Qualities Specifications

CONDUIT [®] Spec Name	Description (Motivation)	Axis*	Airspeed [kts]	Source
<i>Tier 1 Hard Constraints (Stability Requirements)</i>				
EigLcG1	Eigenvalues in L.H.P. (Stability)	All	All	Generic
StbMgG1	Gain and phase margin, loop broken at input to mixer/control allocation (Stability Margin Loop Break Point in Figure 3.1) (Stability)	All	All	SAE AS94900 §3.2.1.4
NicMgG1	Nichols margins, loop broken at input to mixer/control allocation (Robust Stability)	All	All	Ref. 105
<i>Tier 1 Soft Constraints (Handling Qualities Requirements)</i>				
ModFoG2	Command model following cost (HQ)	All	All	Ref. 50 §5.3.3

Table D.1: Recommended Inner-Loop Handling Qualities Specifications

CONDUIT® Spec Name	Description (Motivation)	Axis*	Airspeed [kts]	Source
DstBwG1	Disturbance rejection bandwidth for hold variables (Loads, Ride Quality)	All	All	Refs. 102, 103†
DstPkG1	Disturbance rejection peak for hold variables (Loads, Ride Quality)	All	All	Refs. 102, 103
CrsMnG2	Minimum $\omega_c \geq 4.0$ rad/sec, loop broken at input to mixer/control allocation (Robustness)	P,R	All	Ref. 50
	Minimum $\omega_c \geq 3.5$ rad/sec, loop broken at input to mixer/control allocation (Robustness)	Y	All	Ref. 50
	Minimum $\omega_c \geq 2.0$ rad/sec, loop broken at input to mixer/control allocation (Robustness)	H	0–45	Ref. 50
EigDpG1	Eigenvalue Damping (HQ, Loads)	All	All	Generic
OlpOpG1	Actuator rate limiting requirement, pilot input (PIO)	All	All	Ref. 104
	Actuator rate limiting requirement, disturbance input (PIO)	All	All	Ref. 104
BnwRoH1	Roll attitude bandwidth and phase delay, hover (HQ)	R	0–45	ADS-33E §3.3.2.1
BnwRoF1	Roll attitude bandwidth and phase delay, forward flight (HQ)	R	45–300	ADS-33E §3.4.6.1
BnwPiH1	Pitch attitude bandwidth and phase delay, hover (HQ)	P	0–45	ADS-33E §3.3.2.1
BnwPiF1	Pitch attitude bandwidth and phase delay, forward flight (HQ)	P	45–300	ADS-33E §3.4.1.1
BnwYaH1	Yaw attitude bandwidth and phase delay, hover (HQ)	Y	0–45	Refs. 103, 106
BnwYaF1	Yaw attitude bandwidth and phase delay, forward flight (HQ)	Y	45–300	ADS-33E
FrqHeH1	Heave response equivalent time constant, delay (HQ)	H	0–45	ADS-33E §3.3.10.1
MaxRoH3	Maximum achievable roll rate, hover (HQ)	R	0–45	ADS-33E §3.3.4
MaxRoF4	Maximum achievable roll rate, forward flight (HQ)	R	45–300	ADS-33E §3.4.6.3
MaxPiH3	Maximum achievable pitch rate, hover (HQ)	P	0–45	ADS-33E §3.3.4
MaxPiF1	Maximum achievable load factor, forward flight (HQ)	P	60–300	ADS-33E §3.4.2
MaxYaH3	Maximum achievable yaw rate, hover (HQ)	Y	0–45	ADS-33E §3.3.8
MaxHeH3	Maximum achievable vertical rate, hover (HQ)	H	0–45	ADS-33E §3.3.10.3
CouPRH2	Pitch-roll coupling, hover (HQ)	P,R	0–45	ADS-33E §3.3.9.2 – 3

Table D.1: Recommended Inner-Loop Handling Qualities Specifications

CONDUIT® Spec Name	Description (Motivation)	Axis*	Airspeed [kts]	Source
CouPRF1	Pitch-roll coupling, forward flight (HQ)	P,R	45–300	ADS-33E §3.4.5.2 – 4
CouYaH1	Yaw-collective coupling, hover (HQ)	Y,H	0–45	ADS-33E §3.3.9.1
CouCPF1	Pitch-heave coupling, forward flight (HQ)	P,H	45–300	ADS-33E §3.4.5.1
<i>Tier 1 Summed Objective (Performance Requirements)</i>				
CrsLnG1	Minimize crossover frequency (Act. Activity)	All	All	Ref. 50
RmsAcG1	Minimize actuator usage (Act. Activity)	All	All	Ref. 50
<i>Tier 2 Check Only</i>				
FlpPiF1	Flight path response to pitch attitude (HQ)	H	100–300	ADS-33E §3.4.3.1
QikRoH1	Roll attitude quickness, hover (HQ)	R	0–45	ADS-33E §3.3.3
QikRoF1	Roll attitude quickness, forward flight (HQ)	Y	45–300	ADS-33E §3.4.6.2
QikPiH1	Pitch attitude quickness, hover (HQ)	P	0–45	ADS-33E §3.3.3
QikYaH2	Yaw attitude quickness, hover (HQ)	Y	0–45	ADS-33E
FrqRoD4	LOES Roll model time constant (HQ)	R,Y	60–300	MIL-STD-1797B §5.2.3.1.1
FrqDrD3	LOES Dutch roll frequency (ω_{dr}) (HQ)	R,Y	60–300	MIL-STD-1797B §5.2.3.1.2
DmpDrD2	LOES Dutch roll damping (ζ_{dr}) (HQ)	R,Y	60–300	MIL-STD-1797B §5.2.3.1.2
TdlRoD1	LOES Roll axis time delay (HQ)	R,Y	60–300	MIL-STD-1797B §5.2.3.1.5
TdlYaD1	LOES Yaw axis time delay (HQ)	R,Y	60–300	MIL-STD-1797B §5.2.3.1.6
CapPiL1	LOES Control Anticipation Parameters (HQ)	P	60–300	MIL-STD-1797B §5.2.2.1.1.2
TdlPiL1	LOES pitch axis time delay (HQ)	P	60–300	MIL-STD-1797B §5.2.2.1.1.2
CosLoG1	Max LOES Cost ($J_{LOES} \leq 10$) (HQ)	P,R,Y	60–300	Generic
BnwPiL4	Bandwidth, phase delay (HQ)	P	60–300	MIL-STD-1797B §5.2.2.1.3.1
BnwFpL1	Transient flight-path response (HQ)	P	60–300	MIL-STD-1797B §5.2.2.1.3.3
RolPfd1	Time to achieve bank angle (HQ)	R	60–300	MIL-STD-1797B §5.2.3.5
DrpPiL1	Pitch dropback (HQ)	P	60–300	MIL-STD-1797B §5.2.2.1.2.2 [‡]
TrnCrD2	Turn coordination max n_y (HQ)	R,Y	45–300	SAE AS94900 §3.2.4.1.3.1

* R = Roll, P = Pitch, Y = Yaw, H = Heave

[†] 110% design margin applied to the roll and pitch axes specifications

[‡] Recommended boundaries in Figure 7.21 for pitch tracking tasks

Table D.2: Recommended Outer-Loop Handling Qualities Specifications

CONDUIT® Spec Name	Description (Motivation)	Axis*	Airspeed [kts]	Source
<i>Tier 1 Hard Constraints (Stability Requirements)</i>				
EigLcG1	Eigenvalues in L.H.P. (Stability)	All	All	Generic
StbMgG1	Gain and phase margin, loop broken at input to outer-loop control allocation (Outer-Loop Stability Margin Break Point in Figure 5.1) (Stability)	All	All	SAE AS94900 §3.2.1.4
StbMgG1	Gain and phase margin, loop broken at input to inner-loop control allocation (Stability Margin Loop Break Point in Figure 3.1) (Stability)	All	All	SAE AS94900 §3.2.1.4
NicMgG1	Nichols margins, loop broken at input to outer-loop control allocation (Robust Stability)	All	All	Ref. 105
<i>Tier 1 Soft Constraints (Handling Qualities Requirements)</i>				
ModFoG2	Command model following cost (HQ)	V, γ	All	Ref. 50 §5.3.3
DstBwG1	Disturbance rejection bandwidth for hold variables ≥ 0.75 rad/sec (Loads, Ride Quality)	V, γ	All	Ref. 102
DstPkG1	Disturbance rejection peak for hold variables ≤ 6 dB (Loads, Ride Quality)	V, γ	All	Ref. 102
CrsMnG2	Minimum $\omega_c \geq 1.0$ rad/sec, loop broken at input to outer-loop control allocation (Robustness)	V, γ	All	Ref. 50
EigDpG1	Eigenvalue Damping $\zeta \geq 0.5$ (HQ, Loads)	All	All	Generic
OlpOpG1	Open Loop Onset Point, pilot input (PIO)	V, γ	All	Ref. 104
	Open Loop Onset Point, disturbance input (PIO)	V, γ	All	Ref. 104
BnwPiL4	Bandwidth, phase delay (HQ)	P	60-300	MIL-STD-1797B §5.2.2.1.3.1
BnwFpL1	Transient flight-path response (HQ)	P, γ	60-300	MIL-STD-1797B §5.2.2.1.3.3
DrpPiL1	Pitch dropback (HQ)	P	60-300	MIL-STD-1797B §5.2.2.1.2.2 [†]
<i>Summed Objective (Performance Requirements)</i>				
CrsLnG1	Crossover Frequency (Act. Activity)	V, γ	All	Ref. 50
RmsAcG1	Actuator RMS (Act. Activity)	V, γ	All	Ref. 50
<i>Tier 2 Check Only</i>				
BnwPiF1	Pitch attitude bandwidth and phase delay, forward flight (HQ)	P	60-300	ADS-33E §3.4.1.1
FlpPiF1	Flight path response to pitch attitude (HQ)	γ	100-300	ADS-33E §3.4.3.1
CapPiL1	LOES Control Anticipation Parameters (HQ)	P	60-300	MIL-STD-1797B §5.2.2.1.1.2
TdlPiL1	LOES pitch axis time delay (HQ)	P	60-300	MIL-STD-1797B §5.2.2.1.1.2
CosLoG1	Max LOES Cost ($J_{\text{LOES}} \leq 10$) (HQ)	P	60-300	Generic

Table D.2: Recommended Outer-Loop Handling Qualities Specifications

CONDUIT [®] Spec Name	Description (Motivation)	Axis*	Airspeed [kts]	Source

* P = Pitch, V = Velocity loop, γ = flight path loop

[†] Recommended boundaries in Figure 7.21 for pitch tracking tasks

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