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ACTIVE STABILIZATION OF ROTORCRAFT EXTERNAL SLUNG LOADS IN HIGH SPEED FLIGHT USING AN ACTIVE CARGO HOOK

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Abstract

For rotorcraft carrying slung loads, the stability is often a concern due to the complex slung load dynamics when coupled to rotorcraft flying at high airspeeds. This thesis demonstrates active stabilization of slung loads in high-speed flight by actuating the cargo hook. To show close reference to future vertical lift configurations, the selected aircraft model is the generic tiltrotor model developed by the US Army Technology Development Directorate (TDD). The slung load is the M119 howitzer featured in many airdrop missions and rapid deployments. The dynamic models of the load were identified using the frequency-response method for system identification. The load was initially wind-tunnel tested at Tel Aviv University to collect flight data of the load responses to cargo hook commands for a flight envelope from hover through 200 knots in fullscale flight conditions. The wind tunnel test showed unstable load behavior at high airspeeds reflected as Limit Cycle Oscillation (LCO) of the load pendulum motion within a certain speed range. A set of linear models describing the slung load pendulum dynamics were identified using system identification software CIFER® at airspeeds where the system was stable. An active cargo hook (ACH) controller was designed using classical control methods to increase damping of the load pendulum motions. A nonlinear rotorcraft-load-coupled model was developed in Simulink to simulate the rotorcraft response due to the coupling effects from the load and to test the performance of the designed ACH controller. The simulation results verified the controller performance at increasing damping of the load pendulum motions and showed the effectiveness

of the controller stabilizing the load in high-speed flight conditions. The simulation also showed good agreement with the closed-loop wind tunnel test results.

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List of Symbols

Variables

A, B, C, D	State matrix, input matrix, output matrix, feedthrough matrix
а	Acceleration magnitude
$\beta_0, \beta_{1c}, \beta_{1s}, \beta_2$	Rotor flapping angles
C(s)	Compensator transfer function
d	Lag compensator frequency
δ_x , δ_y	Commanded lateral and longitudinal cargo hook actuation
$\delta_a, \delta_e, \delta_r$	Aileron, Elevator, Rudder inputs
δ_{nac}	Symmetrical nacelle angle
$\Delta\theta_0, \Delta\theta_{1c}', \Delta\theta_{1s}'$	Differential collective, phased lateral cyclic, phased
	longitudinal cyclic
F	Force magnitude
G(s)	Filter transfer function
K	ACH controller gain
K_P, ζ_p, ω_P	Transfer function parameters (gain, damping, and natural
	frequency)

L_F	Full-scale dimensional length
L_M	Model-scale dimensional length
l	Cable length
λ	Froude scaling factor
$\lambda_0, \lambda_{1c}, \lambda_{1s}$	Rotor inflow states
m	Mass
р	Wash-out filter frequency
<i>p</i> , <i>q</i> , <i>r</i>	Roll, pitch and yaw rates
$ec{r}_A$	Active cargo hook position vector
\vec{r}_{CG}	Tiltrotor cg position vector
$ec{r}_{\Delta}$	Cargo hook actuation position vector
$\phi, heta, \psi$	Tiltrotor Roll, pitch and yaw angles
ϕ_c	Lateral cable angle
<i>Q</i> , <i>R</i>	Weighting matrices for system state and control vectors
S	Complex frequency
Т	Trim cable tension force magnitude
T_{EB}	Earth to rotorcraft body transformation matrix
$ au_a$	Actuator time constant
τ	Time delay

$\theta_0, \theta_{1c}', \theta_{1s}'$	Symmetric collective, phased lateral cyclic, phased
	longitudinal cyclic
$ heta_c$	Longitudinal cable angle
$u_{lon}, u_{lat}, u_{col}, u_{ped}$	Pilot control inputs
u, v, w	Inertial velocity components
W	Load weight
x_{pos}, y_{pos}	Longitudinal and lateral cargo hook position
x_{cmd}, y_{cmd}	Longitudinal and lateral cargo hook command inputs
$\widetilde{\omega}_{b/e}$	Angular speed of rotorcraft with respect to the inertial frame
$\zeta_0,\zeta_{1c},\zeta_{1s},\zeta_2$	Rotor Lag Angles
Ċ	Time rate of change
$\widehat{()}$	Unit vector
$\overrightarrow{()}$	Vector
()_L	Load
() _T	Tiltrotor
O^b	Expressed in rotorcraft body-fixed coordinates
() ^e	Expressed in inertial coordinates
NUMA	

Acronyms

BDCB	Base Data Collection and Control Box
CAF	Cable Angle Feedback
cg	Center of Gravity
DM	Delay Margin
FS	Full-Scale
FVL	Future Vertical Lift
GM	Gain Margin
GTR	Generic Tiltrotor
HDCB	Head Data Collection Box
kt	Knots
LCAF	Lagged Relative Cable Angle Feedback
LCO	Limit Cycle Oscillation
nac	Nacelle Angle
РМ	Phase Margin
TDD	U.S. Army Technology Development Directorate

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Chapter 1 Introduction

1.1 Motivation

External slung load carriage has been studied and discussed frequently for various applications of rotorcraft transport of large loads such as civil construction, military combat, logging, and firefighting work. For rotorcraft slung load missions, the dynamics of the load and the aircraft are important attributes for the overall safety and maneuverability of the flight system. In forward flight, rotorcraft carrying slung loads frequently encounter drastic or unstable load motion that requires a restriction on the operational envelope of the rotorcraft. The speed of the rotorcraft is largely limited when carrying a slung load in consideration of the safety of the flight and the workload on the pilot. With load stabilization, the vehicle can usually fly faster with less demand from the pilot input. As Future Vertical Lift rotorcrafts are designed to operate at airspeeds over 200 knots including the requirement for high-speed carriage of slung loads, it is important to develop a methodology to stabilize the load across an operational envelope with a higher speed limit.

1.2 Background

Past studies have shown load stabilization methods in passive or active manners. For passive stabilization, methods such as using customized rigging or employing stabilizing add-ons can both extend the flight envelope of the load. In Ref [1], rigid and fabric fins were installed on the back of a CONEX box carried by a UH-60 rotorcraft to provide directional stability. Ref [2] used active rotation with controlled anemometric cups to suppress the load pendulum motions. Both methods

can extend the operational limit of the system from 60 knots to over 100 knots. However, while passive stabilization methods are easy to implement, it usually requires customization for different types of load. The additional weight and drag caused by the add-ons can degrade the performance and handling qualities of the rotorcraft.

For active stabilization, the load motion can be stabilized by the motion of the rotorcraft when providing the correct control algorithm or by an active device designated specifically for the slung load. In Ref [3], a control system was developed that uses Cable Angle Feedback (CAF) as well as conventional fuselage feedback to improve load damping and handling qualities for hover/lowspeed operations. This control system was designated for a UH-60 platform and the effectiveness of the system was demonstrated in a piloted fixed-base simulation. In Ref [4], Krishnamurthi and Horn introduced lagged cable angle feedback (LCAF), which eliminates noisy cable angle and rate sensor signals. The designed controller with LCAF was shown effective at reducing load oscillations at hover/low speed with trade-offs in pilot response. Using active stabilization can also be applied in the area of precision handling of the load. In Ref [5], an active controller was designed and integrated with the primary flight control system that allows a UAV helicopter to safely transport a slung load and to place it precisely on a moving platform such as a truck or a ship.

Another type of technique was developed by the Boeing Company that uses an active stabilization device called the active cargo hook (ACH). The active cargo hook automatically varies the hook position where the slung load is attached to and stabilizes the load without the requirement for pilot intervention. Such technique was flight-tested on an H-6 flying testbed where the cargo hook was moved longitudinally and laterally based on LCAF [6]. The test results showed a significant increase in load damping and reduction of pilot workload during low-speed flight. In Ref [7], the stabilization method of using ACH with LCAF was further investigated across the

flight envelope of a UH-60 helicopter with a focus on the higher speed range. Simulations of the rotorcraft/load coupled system showed that the controller was successful in providing system stability throughout the target flight speed envelope from hover to 100 knots.

Based on the results mentioned above, a research program for the development of active stabilization methods of external loads during high-speed flights was recently initiated by the Combat Capabilities Development Command (CCDC) and U.S. Army Technology Development (TDD). The project aims to expand the flight envelope of slung-load carriage missions and investigate the feasibility of using both CAF to the rotorcraft primary flight control system and the ACH for load stabilization during highspeed flights. The application of ACH will be discussed in this thesis.

1.3 Objectives.

The objective of this thesis is to present a thorough study of the slung-load system that incorporates an FVL tiltrotor model and an M119 howitzer load model. The study focuses on the application of ACH stabilization of the load during highspeed flights and applies both wind-tunnel testing and dynamic simulations to investigate the performance of the ACH. Figure 1.1 shows a V-22 Osprey as an example of the FVL tiltrotor.



Figure 1.1. Bell Boeing V-22 Osprey carrying an M119 Howitzer (downloaded from https://www.alamy.com).

The research contains the joint effort between organizations from Israel and the U.S and is conducted under the Rotorcraft Project Agreement (RPA) between the two. Figure 1.2 shows a simplified flowchart of the research. The Israeli researchers first performed the open-loop wind tunnel tests across the flight envelope from zero to 23 m/s (equivalent to 200 knots in full-scale condition). During the tests, multiple cargo hook inputs were tested to expose the dynamics of the load throughout a range of speed of interest. Frequency-sweep inputs were utilized to generate frequency response of the load pendulum dynamics. Researchers at TDD conducted system identifications based on the results from the wind tunnel tests to provide linear models of the load pendulum dynamics at different airspeeds.

After the initial wind-tunnel tests and system identifications, a preliminary control design based on the load linear models was conducted. The controller was designed to use CAF as the inputs and calculate the corresponding hook motion to stabilize the load pendulum motions. The load was later integrated with the FVL tiltrotor in Simulink to perform simulations of the coupled load/rotorcraft model as well as to conduct linear analysis accessing the stability of the coupled system. The performance of the controller was validated from the simulation results and the closedloop wind-tunnel tests.



Figure 1.2. Research Flowchart.

The work in this thesis is organized as described in the following. In Chapter 2, the development of the coupled load/aircraft model is presented. The first part of this chapter summarizes the fundamental work performed by the other organizations, which includes the open-

loop wind-tunnel tests conducted by the Israeli researchers and system identification conducted by TDD. The second part of the chapter presents the generic tiltrotor model (GTR) developed by TDD. A detailed description of the coupling procedure of the load and the tiltrotor model is shown after the introduction of the rotorcraft. At the end of the chapter, the stability analysis of the lowfrequency dynamics of the coupled system is discussed. In Chapter 3, the preliminary controller design for the isolated load is demonstrated. The design process compares two ACH controllers using lead and lag compensations to stabilize the load and also shows a detailed analysis of the performance of the controllers in terms of damping ratio, stability margins, and resulting cargo hook usage for control action. In Chapter 4, the closed-loop wind tunnel results are compared with the simulation results. The coupled system was simulated at different airspeeds with the same sets of command inputs as the closed-loop wind-tunnel experiment to check the accuracy of the simulation model. The simulation results including responses of the tiltrotor and load and the actuation of the cargo hook are plotted for better visualization of the system dynamics. Chapter 5 summarizes the studies of the M119 slung load and coupled tiltrotor/load system and suggests the future work that could be done to improve the performance of the ACH controller.

Chapter 2 System Description and Modeling

2.1 Introduction

This section will describe the development of the M119 Howitzer slung load model. Researchers at Technion University designed and conducted wind-tunnel experiments to a scaled model of the M119 load. The wind-tunnel experiment featured a newly designed hardware-in-theloop (HIL) setup that incorporates a moving cargo hook, multiple sensors for data measurements, and a computer for data processing. Initial open-loop tests were performed by placing the load model in an airspeed-increasing wind field. The response of the load motion was recorded along with the airspeed to evaluate the dynamic stability of the load model across the flight envelope. Frequency sweep tests were performed after the open-loop tests to identify dynamic load models. The load was excited by the moving cargo hook at varying frequency and the resulted cable angles were recorded to generate frequency response of the load pendulum motions to the hook command. The definition of cable angles will be explained in later sections. The tests were repeated at different airspeeds where the load was dynamically stable. The linear models of the load pendulum motions were then identified by TDD through the frequency response at each test speeds.

The aircraft model used in the study was a generic tilt rotor model developed by the TDD that has a flight envelope from hover to 260 knots. The model was constructed based on 228 linear models at different flight conditions, which were extracted from a nonlinear flight dynamics model developed using HeliUM [9]. Model stitching technique was applied during the development of

the simulation model that combined a collection of linear state-space models for discrete flight conditions, with corresponding trim data, into one continuous, full-envelope flight-dynamics simulation model [10].

At later stages, model coupling was conducted to the slung load model and the tiltrotor model to analyze the interactions between the different dynamics from the two models and study the effects of the slung load on the aircraft across the flight envelope.

2.2 External Load Model

The M119 Howitzer is a lightweight 105mm gun used by the United States Army, it is frequently transported by rotorcraft on the combat field. An M119 howitzer gun weighs 4400 lb [11] and has two configurations during transportation. Figure 2.1(a) shows the howitzer being carried in the folded position such that the gun barrel is folded into the firing platform and figure (b) shows the firing configuration with the barrel fully extended.



(a)



(b)

Figure 2.1. M119 Howitzer model: (a) Folded configuration (b) Firing configuration (from [11]).

2.2.1 M119 Model Wind Tunnel Test

A model-scale howitzer was used in the wind tunnel test that has a scale ratio of λ =20 compared to the full-scale model.

$$\lambda = \frac{L_F}{L_M} \tag{2.1}$$

The model-scale howitzer has the same shape as the full-scale model such that geometrical similarity requirements for Froude scaling were satisfied [12]. Extra masses were added to the model to achieve proper scaling of the weight and inertia. Table 2.1 lists the Froude scaling table that was used throughout the study [13].

Physical Parameter	Unit	Scale Factor
Length	[m]	λ
Mass	[kg]	λ^3
Force	[N]	λ^3
Moment	$[N \cdot m]$	λ^4
Frequency	s ⁻¹	$1/\lambda^{0.5}$
Speed	m/s	$\lambda^{0.5}$
Time	S	$\lambda^{0.5}$

Table 2.1. Froude scaling.

The experiment setup is shown in Figure 2.2. The load was attached to the moving hook installed at the tunnel ceiling through slings [14]. The motion hook was controlled by a stepper motor and can move in the longitudinal and lateral axis. The simulated motion is the sum of the rotorcraft motion and the motion of the active cargo hook relative to the fuselage. The slung load was paired with an instrumentation package that measures the linear acceleration, angular rate, and the local magnetic field of the load, which provide an estimation of the angular attitude and heading [14]. The load cell attached to the hook would record the combined tension force transmitted through the slings and other sensors and electronic modules were used to record measurements such as the position of the hook and the cable angles. The above information would all be collected by the Head Data Collection Box (HDCB), which sends the measurements to the Base Data Collection and Control Box (BDCB) that communicates with a laptop outside the wind tunnel. The laptop controls the entire set by receiving measurements and sending control inputs to the moving hook in real-time.



Figure 2.2. M119 Howitzer model in the wind tunnel (from [14]).

At the early stages of the experiment, open-loop tests were performed in the wind tunnel without any control action on the cargo hook. The open-loop tests included multiple procedures for evaluating the dynamics of the slung load through excitation of the cargo hook. One of them is to excite the load with an impulse hook command to check for stability of the load at different airspeeds. Another procedure is to excite the hook with varying frequency to identify a model of the load response to ACH commands. The type of input used in this procedure is called the frequency-sweep input, which allows the operator to excite the system with a range of frequency of interest, and it is commonly used to extract frequency response of a system dynamics. Therefore,

multiple sweeps were performed at different wind tunnel speeds including zero and speeds from 6 m/s (52 kt FS) up to instability of the load with increments of 1 m/s (8.67 kt FS) [14]. At later stages, closed-loop tests with the designed ACH controller were performed, which are discussed in Chapter 4.



Figure 2.3. Longitudinal and lateral cable angles (from [14]).

Figure 2.3 shows a schematic drawing of the load cable angles that defines the orientation of the load. The coordinate system is defined such that the x-axis is pointing into the incoming wind, the y-axis is pointing to the right, and the z-axis pointing down. The cable angles measure the cg position of the load and express it in terms of longitudinal and lateral cable angles relative to axes aligned with the direction of flight, ϕ_c , θ_c (first pitch, positive forward, and then roll, positive left) [14].

The following figures present a group of airspeed sweeps conducted on the load without any hook motion to understand the general behavior of the load across the flight envelope. Figure 2.4 displays the load responses of the folded configuration including attitude angles, cable angles, and the hook force transmitted through the sling towards increasing wind tunnel speed. The tunnel speed increased gradually from zero to 23 m/s (200 kt FS), and the load remained stable until 14 m/s where oscillation could be observed from all three load attitude angles about their mean values

with the same amplitudes. This behavior can be categorized as a Limit Cycle Oscillation (LCO). It is a general dynamical behavior of a nonlinear system and is defined as the phenomena that occur in a dynamical system where the system oscillation follows a steady or quasi-steady trajectory regardless of the initial condition as time goes to infinity [15]. The mean values of the roll and yaw angles appeared to be insensitive to increasing tunnel speed while the negative pitch angle increased with airspeed. The same conclusion can be drawn from the cable angles as the lateral cable angle varied slightly from 0° while the longitudinal cable angle increased with increasing tunnel speed. This is expected as the longitudinal cable angle, which is also called the trailing angle, is related to the drag of the load that increases as the speed increases. The hook force also increased significantly at higher speeds due to the combined large negative lift and drag force generated by the trailing slung load.



Figure 2.4. Load attitude, cable angles, and hook force of the folded configuration during an increase of the wind tunnel speed (from [14]).

The same test procedures were reproduced with the load firing configuration as the data shown in Figure 2.5. The load attitude angles were seen to vary much less in the pitch axis but vary more in the yaw axis as compared with the folded configuration. The LCO observed from the folded load above 13m/s (113kt FS) tunnel speed did not occur for the firing load. However, after the speed reached 15m/s (130kt FS), the load response drastically changed and became divergent. Another notable difference between the two configurations was the hook force magnitude. The tension force of the firing load had a much smaller increase until the load was unstable, which

indicates the combined lift and drag forces from the load was much smaller as compared to the folded load.



Figure 2.5. Load attitude, cable angles, and hook force of the firing configuration during an increase of the wind tunnel speed (from [14]).

The airspeed sweep results shown for the two configurations demonstrate the stability issues of the slung load during high-speed flights. An unstable load will induce oscillations in the rotorcraft and penalize the handling and ride qualities. Also, large load oscillations can lead to the load striking the rotorcraft fuselage, causing concern to flight safety.

2.2.2 Identification of Slung Load Linear Models

The open-loop tests showed the pendulum dynamics of the load oscillating about its steadystate positions in stable or unstable fashions. In order to design a controller for active stabilization of the load, dynamic models of the load response to ACH commands are required. As discussed in [16], the frequency domain approach to system identification is a well-proven and well-suited approach for identifying rotorcraft dynamics. By exciting the system with the correct type of input with good spectral content, the frequency response can accurately characterize the system dynamics and describe them with linear functions. For the commonly used inputs, frequencysweep inputs are the most reliable way to excite the system and provide frequency contents [16]. The term "frequency-sweep" refers to a class of control inputs that has a quasi-sinusoidal shape of increasing frequency. The range of frequencies is strictly controlled by the operator to extract the targeted dynamic mode and avoid excitation on other higher-order modes such as the dynamics of the mechanical linkages. In this case, the hook was driven in either longitudinal or lateral direction through the range of frequencies of interest to extract the load pendulum dynamics. Between the speed from 0 to 14m/s (hover to 122 kt FS), frequency sweeps in longitudinal and lateral directions found that linear models of the load pendulum motions could be identified based on the cable angle response to the cargo hook position. Table 2.2 lists the parameters of the sweep tests.

Parameters	Model Scale	Full Scale
Pendulum Length	273.6 mm	18.0 ft
Simple Pendulum Frequency	6.00 rad/s	1.34 rad/s
Sweep Frequency Range	0.22 to 53.7 rad/s	0.05 to 12 rad/s
Sweep Duration	44.7 sec	200 sec
Desired airspeed Range	0 to 23 m/s	0 to 200 knots
Hook Position Limits	<u>+</u> 80 mm	<u>+</u> 5.25 ft
Hook Rate Limits	$\pm 100 \text{ mm/s}$	±6.56 ft/s

Table 2.2. Frequency sweep test parameters (from [14]).

Researchers at TDD used CIFER® and its associated utilities to process the data obtained from the frequency-sweep tests and generated the corresponding frequency responses of the load pendulum dynamics at each given speeds. The CIFER® is an acronym for Comprehensive Identification from Frequency Responses and it is a software developed by the TDD for system identification based on the frequency response approach. The software includes core analysis programs built around a sophisticated database, along with a set of user utilities for dynamics studies of aeronautical systems [17]. The frequency responses were obtained using the advanced Fast Fourier Transform (FFT) algorithm and the associated windowing techniques implemented in CIFER® [16].

A selected case of the frequency-sweep tests at zero tunnel speed is presented in the following. Figure 2.6(a) shows the hook being driven longitudinally and the corresponding cable angle response. The hook input amplitude started at 9 mm (0.6 ft FS) and gradually decreased after 11.4 rad/s to avoid exceeding the hook rate limit. The actuator of the moving hook was found to have good linear input/output response and excellent signal to noise ratio out to 20 rad/s as identified from its frequency response [14]. The cable angle response showed mainly on-axis ($\theta_c/x, \phi_c/y$) response to the motion of the hook and the off-axis ($\theta_c/y, \phi_c/x$) correlation was found to be small for both longitudinal and lateral pendulum dynamics.

The frequency response of the longitudinal cable angle is shown in Figure 2.6(b) along with the fitted transfer function represented by the black dash line. The transfer-function modeling is a function provided by CIFER® that compose parametric models of the system dynamics and express them in the form of transfer functions. The values of the transfer-function gain, pole locations, and zero locations are determined numerically to provide the best match (in a leastsquares sense) to the frequency-response data [16]. The correlation between the two curves shows good accuracy over the range of 2 to 20 rad/s with an identification cost of 148.7. The coherence provides a key measure of the quality of data from the frequency-response. It indicates whether the system has been satisfactorily excited across the entire frequency range and shows whether the system being modeled is well characterized as a linear process in this frequency range [16]. Usually, a coherence above 0.6 represents good accuracy at the corresponding frequency. The cost function indicates the level of accuracy of the fitted transfer function. A cost function smaller than 100 generally reflects an acceptable level of accuracy for dynamics modeling [16]. The parameters of the transfer function are listed at the bottom of figure (b). Symbols Z and W represent the damping and the natural frequency of a 2nd order linear system along with a delay factor in seconds that account for unknown delays and higher dynamics in the system.


(b)

Figure 2.6. (a): Longitudinal frequency sweep and cable angle response at zero wind tunnel speed (firing configuration) (b): Frequency response of the longitudinal pendulum dynamics with the fitted transfer function in CIFER®.



(b)

Figure 2.7. (a): Lateral frequency sweep and cable angle response at zero wind tunnel speed (firing configuration). (b): Frequency response of the lateral pendulum dynamics with the fitted transfer function in CIFER®.

Figure 2.7 shows the same frequency sweep in the lateral direction. The lateral load pendulum motion showed less off-axis dynamics compared to the longitudinal pendulum motion as the longitudinal cable angle reacted slightly to the hook excitation. The frequency response also showed better agreement with the fitted transfer function with a much lower cost of 17.3, which indicates that the transfer function produced a match that is nearly indistinguishable from the frequency-response data. For both longitudinal and lateral load pendulum dynamics, the damping ratios were found to be small and the identified pendulum frequencies were close to the expected pendulum frequency of 6 rad/s.

$$\frac{\theta_c}{x}, \frac{\phi_c}{y} = \frac{K_p s^2}{\left(s^2 + 2\zeta_p \omega_p s + \omega_p^2\right)} e^{-s \cdot \tau}$$
(2.2)

The identification procedures were performed for both the firing and folded configurations over the speed range where the load was stable. The identified transfer-function structure is shown in Equation 2.2. The parameters of the identified linear models are listed in Table 2.3-4. The firing configuration was tested up to 14m/s (122 kt FS) while the folded configuration was tested to a higher speed of 20m/s (174 kt FS) for the longitudinal dynamics and 14m/s (122 kt FS) for the lateral dynamics. The pendulum motions for the two configurations were both lightly damped (ζ_p = .005 to .06) and had similar frequency over the tested speed range. Therefore, it was assumed that the load pendulum dynamics were approximately independent of airspeed and the identified linear models would not change significantly out to the higher speed range. This suggests that the controller designed from the available models may remain effective over the speed range where linear models could not be identified due to instability. This turns out to be the case as shown in later chapters.

WT	FS	Kp	$\zeta_{ m P}$	ω_p	τ	cost
Speed	Speed			(rad/s)	(sec)	
(m/s)	(kts)					
Longitudinal, $[\theta_c/x]$						
0	0	152	.010	5.51	.014	58
6	52	152	.024	5.60	.014	71
8	70	169	.027	5.58	.014	92
10	87	175	.027	5.61	.018	114
12	104	170	.041	5.73	.010	81
14	122	159	.058	6.06	.006	162
Lateral, $[\phi_c/y]$						
0	0	.176	.007	5.45	.025	23
6	52	.183	.016	5.50	.021	14
8	70	.185	.025	5.56	.020	17
10	87	.176	.028	5.65	.021	33
12	104	.177	.033	5.74	.023	17
14	122	.170	.037	5.97	.021	49

Table 2.3. Identified pendulum models for the firing configuration (from [14]).

WT	FS	Kp	ζp	ω _p	τ	cost
Speed	Speed		-	(rad/s)	(sec)	
(m/s)	(kts)			· ·		
Longitudinal, $[\theta_c/x]$						
0	0	172	.010	5.55	.019	149
6	52.	180	.017	5.46	.013	36
8	70	191	.022	5.39	.013	28
10	87	173	.019	5.33	.014	141
12	104	176	.022	5.24	.013	64
14	122	172	.037	5.19	.007	166
16	139	210	.027	5.27	.010	40
18	157	158	.041	5.45	.007	50
20	174	174	.033	5.54	.004	83
Lateral, $[\phi_c/y]$						
0	0	.176	.006	5.48	.022	17
6	52.	.194	.017	5.59	.020	22
8	70	.187	.017	5.62	.020	18
10	87	.193	.020	5.70	.019	48
12	104	.214	.040	5.86	.018	108
14	122	.225	.047	6.19	.021	183

Table 2.4. Identified pendulum models for the folded configuration (from [14]).

2.3 Tiltrotor Model

For the mission purpose of high-speed load carriage, a specific rotorcraft model was selected to analyze the coupled system dynamics. The selected model is the generic tiltrotor (GTR) that was derived from a combination of multiple tiltrotors including the XV-15, V-22, and the notional NASA Large Civilian Tilt-Rotor 2 (LCTR2) [9]. Figure 2.8 shows the rendering image of the GTR model. The physical parameters of the tiltrotor are listed in Table 2.5. The generic tiltrotor has similar properties to the other three rotorcraft with differences in dimensions. The nacelle angle can be tilted between 0 to 90 degrees. When the nacelle angle is 90 degrees, the nacelles are perpendicular to the wings as the tiltrotor is in helicopter mode. As forward speed increases, the nacelles gradually tilt forward and the nacelle angle decreases until it reaches 0 degrees when the tiltrotor is in fixed-wing mode. The nacelles were modeled as rigid bodies with a rotational degreeof-freedom pivoted around the wingtips.



Figure 2.8. Generic tiltrotor rendering (from [9]).

The flight dynamics models for the tiltrotor were developed using HeliUM [18,19], which uses finite-element methods to capture inertial, structural, and aerodynamic loads along the blade

segment. The aerodynamics of each component including the blade, wing, and fuselage came from nonlinear lookup tables, and the rotor air-wakes were modeled using a dynamic inflow model. The modeling accuracy from HeliUM was validated for various aircraft, the results of all these validation efforts provide confidence in the modeling fidelity for the generic tiltrotor model [9].

Gross Weight $32,100$ lbsMax Continuous Power (SL) $9,400$ hpMaximum Speed (VTAS) 280 ktsWing Span 45 ftWing Sweep 0 degNacelle Range $0-90$ degRotor DataRadius17.8 feet	Aircraft Data				
Max Continuous Power (SL)9,400 hpMaximum Speed (V_{TAS})280 ktsWing Span45 ftWing Sweep0 degNacelle Range0 – 90 degRotor DataRadius17.8 feet	Gross Weight	32,100 lbs			
Maximum Speed (V_{TAS})280 ktsWing Span45 ftWing Sweep0 degNacelle Range0 - 90 degRotor DataRadius17.8 feet	Max Continuous Power (SL)	9,400 hp			
Wing Span45 ftWing Sweep0 degNacelle Range0 - 90 degRotor DataRadius17.8 feet	Maximum Speed (V _{TAS})	280 kts			
Wing Sweep0 degNacelle Range0 - 90 degRotor DataI7.8 feet	Wing Span	45 ft			
Nacelle Range0 – 90 degRotor DataRadius17.8 feet	Wing Sweep	0 deg			
Rotor Data	Nacelle Range	$0-90 \deg$			
Radius 17.8 feet	Rotor Data				
	Radius	17.8 feet			
Number of Blades/Rotor 4	Number of Blades/Rotor	4			
Rotational Speed 40-30 rad/sec	Rotational Speed	40-30 rad/sec			
Precone 2.5 deg	Precone	2.5 deg			
Twist -44 deg	Twist	-44 deg			

 Table 2.5. Tiltrotor Configuration Data (from [9]).

To provide a MATLAB 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 generic tiltrotor. The tiltrotor linear models contain 51 states and 9 control inputs that are listed as the following.

51 tiltrotor states:

- Fuselage states $(u, v, w, p, q, r, \phi_T, \theta_T, \psi_T)$
- Rotor states per rotor $(\dot{\zeta}_0, \dot{\zeta}_{1c}, \dot{\zeta}_{1s}, \dot{\zeta}_2, \zeta_0, \zeta_{1c}, \zeta_{1s}, \zeta_2, \dot{\beta}_0, \dot{\beta}_{1c}, \dot{\beta}_{1s}, \dot{\beta}_2, \beta_0, \beta_{1c}, \beta_{1s}, \beta_2)$
- Inflow states per rotor $(\lambda_0, \lambda_{1c}, \lambda_{1s})$
- Nacelle states per nacelle (nacelle angle and nacelle angular rate)

9 control inputs:

- Symmetric phased lateral cyclic (θ'_{1c})
- Differential collective $(\Delta \theta_0)$
- Symmetric phased longitudinal cyclic (θ'_{1s})
- Symmetric collective (θ_0)
- Differential phased longitudinal cyclic $(\Delta \theta'_{1s})$
- Aileron (δ_a)
- Elevator (δ_e)
- Rudder (δ_r)
- Symmetric nacelle angle (δ_{nac})

The rotor states reflect four second-order rotor dynamics for each of the two-blade modes associated with each rotor. The nacelle states contain two second-order nacelle rotational dynamics for each nacelle. The stitched model is stitched in total x body axis velocity u and symmetric nacelle angle δ_{nac} [9], and it can be trimmed, simulated, and linearized at any flight condition within the flight envelope of the aircraft as shown in Figure 2.9. The tiltrotor stitched model provides a good platform for control law development and piloted simulations.



Figure 2.9. GTR Stitched model conversion corridor and linear model points.

2.4 Coupled Tiltrotor-Load System.

The stitched model of the tiltrotor and the identified load models allows the study of the aircraft-load-coupled system through linear analysis and simulation. The identified system model was approximated to have a single stiff cable connecting the point of mass load and the tiltrotor while the actual model had three non-stiff cables and rigid body dynamics. The motion of the cargo hook, which was a combination of the motion of the aircraft and the cargo hook actuation would excite the load to oscillate. While the load oscillated, it generated a net external force and moment that were transferred to the aircraft through the cargo hook. Figure 2.10 shows a schematic drawing of the coupling between the tiltrotor and the load.



Figure 2.10. Schematic diagram of the tiltrotor-load coupled system.

The control allocation module transforms the pilot inputs into mechanical inputs on the rotorcraft. The pilot inputs consist of lateral cyclic (u_{lat}) , longitudinal cyclic (u_{lon}) , collective (u_{col}) and pedal (u_{ped}) . The transformation follows the same procedure in Ref. [20], in which it uses a control allocation structure similar to the XV-15 tiltrotor.

The tiltrotor stitched model contains the trim data and linearized model at the desired flight condition. The block receives pilot input and outputs system states X_{TR} . The next block uses the state variables to compute the acceleration of the cargo hook in the longitudinal and lateral directions. The cargo hook accelerations are then transmitted to the linear model of the slung load, which outputs the load cable angles. Lastly, the Cable Force and Moment block calculate the external force and moment based on the cable angles and aircraft states, and by feeding back the force and moment terms, the load is coupled to the aircraft. Previous sections have discussed the Tiltrotor Stitched Model and the Slung Load Dynamics blocks in the figure, and the following section will disclose the calculation of the Cargo Hook Velocity and Acceleration and the Cable Force and Moment blocks.



Figure 2.11. Coordinate systems of the tiltrotor-load coupled model.

The computation of the hook velocity and acceleration requires expressing the terms in the inertial coordinate system. Therefore, the hook velocity and acceleration were first calculated and expressed in the aircraft body frame and later transformed into the inertial frame. Figure 2.11 shows the inertial coordinate system (x_E, y_E, z_E) and the body coordinate system (x_b, y_b, z_b) . The nominal attachment point of the cargo hook (with no actuation) is indicated by CH and the actual position of the hook (with actuation) is represented by A. The distance between points CH and A is expressed by \vec{r}_{Δ} that represents the hook actuation away from the attachment point. The vector \vec{r}_{CH} measures the location of the cargo hook attachment point from the center of gravity of the aircraft. The vector \vec{r}_A expresses the location of the cargo hook in the inertial frame, which is the

frame of interest to measure the location of the cargo hook. The derivation of the vector \vec{r}_A is shown in the following equations that include transformation and conversion between the two reference frames and the associated coordinate systems.

$$\vec{r}_A = \vec{r}_{CG} + \vec{r}_{CH} + \vec{r}_\Delta \tag{2.3}$$

Taking the derivatives of both sides of the equation:

$$\vec{v}_{A}^{\ e} = \vec{v}_{CG}^{\ b} + \dot{\vec{r}}_{CH}^{\ b} + \vec{\omega}_{b/e} \times \vec{r}_{CH}^{\ b} + \dot{\vec{r}}_{\Delta}^{\ b} + \vec{\omega}_{b/e} \times \vec{r}_{\Delta}^{\ b}$$
(2.4)

Vector \vec{r}_{CH} is fixed in the body frame, therefore, $\dot{\vec{r}}_{CH}^{b} = 0$. The acceleration of the cargo hook position A can be derived as the following:

$$\vec{a}_{A}^{\ e} = \dot{\vec{v}}_{CG}^{\ b} + \vec{\omega}_{b/e} \times \vec{v}_{CG}^{\ b} + (\dot{\vec{\omega}}_{b/e} + \vec{\omega}_{b/e} \times \vec{\omega}_{b/e}) \times \vec{r}_{CH}^{\ b}$$

$$+ \vec{\omega}_{b/e} \times (\dot{\vec{r}}_{CH}^{\ b} + \vec{\omega}_{b/e} \times \vec{r}_{CH}^{\ b}) + \ddot{\vec{r}}_{\Delta}^{\ b} + \vec{\omega}_{b/e} \times \dot{\vec{r}}_{\Delta}^{\ b}$$

$$+ (\vec{\omega}_{b/e} \times \vec{\omega}_{b/e}) \times \vec{r}_{\Delta}^{\ b} + \dot{\vec{\omega}}_{b/e} \times \vec{r}_{\Delta}^{\ b} + \vec{\omega}_{b/e} \times (\dot{\vec{r}}_{\Delta}^{\ b} + \vec{\omega}_{b/e} \times \vec{r}_{\Delta}^{\ b})$$

$$+ \vec{\omega}_{b/e} \times \vec{r}_{\Delta}^{\ b})$$

$$(2.5)$$

The cross product between a vector with itself is equal to 0. Hence, simplifying the expression and eliminating the zero terms, the expression of the cargo hook acceleration in the inertial coordinate is:

$$\vec{a}_{A}^{\ e} = \dot{\vec{v}}_{CG}^{\ b} + \vec{\omega}_{b/e} \times \vec{v}_{CG}^{\ b} + \dot{\vec{\omega}}_{b/e} \times \left(\vec{r}_{CH}^{\ b} + \vec{r}_{\Delta}^{\ b}\right)$$

$$+ \vec{\omega}_{b/e} \times \left[\vec{\omega}_{b/e} \times (\vec{r}_{CH}^{\ b} + \vec{r}_{\Delta}^{\ b})\right] + \ddot{\vec{r}}_{\Delta}^{\ b} + 2\vec{\omega}_{b/e} \times \dot{\vec{r}}_{\Delta}^{\ b}$$
(2.6)

While the velocity, acceleration, and rotational speed of the rotorcraft are expressed as the following:

$$\vec{v}_{CG}^{\ b} = [u, v, w], \, \dot{\vec{v}}_{CG}^{\ b} = [\dot{u}, \dot{v}, \dot{w}], \, \vec{\omega}_{b/e} = [p, q, r]$$
(2.7)

The cargo hook actuation, velocity, and acceleration are:

$$\vec{r}_{\Delta}^{\ b} = [x_{CH_b}, y_{CH_b}], \dot{\vec{r}}_{\Delta}^{\ b} = [\dot{x}_{CH_b}, \dot{y}_{CH_b}], \ddot{\vec{r}}_{\Delta}^{\ b} = [\ddot{x}_{CH_b}, \ddot{y}_{CH_b}]$$
(2.8)

Notice the $[\ddot{x}_{CH_b}, \ddot{y}_{CH_b}]$ in Equation 2.8 are different than the terms in Figure 2.10. The former represents the accelerations of the cargo hook actuation measured in the body frame and the latter is the accelerations measured in the inertial frame and expressed in body coordinates.

The hook accelerations are then taken as input by the linear load model to compute the load cable angles. The transfer function block essentially appends 4 extra states, which are derived from the two second-order load pendulum models, to the GTR stitched model. The cable-angle states combined with tiltrotor states will compute the external force transferred to the cargo hook that is generated by the load. The trim cable force measured from the wind tunnel test will account for both the aerodynamic and inertial forces of the load. It was assumed that dynamic force increments were only due to the load inertial force computed from the linear acceleration and that the load aerodynamic force was assumed to be unchanged from the trim value. While coupled with the rotorcraft, the cable angles need to be expressed relative to the tiltrotor such that the action of the rotorcraft is taken into account. The identified load transfer function outputs the cable angles as if they are measured in the inertial frame. However, the desired cable angles should be expressed relative to the rotorcraft body-fixed frame since the force and moment are computed by the relative motion between the load and the tiltrotor. The derivations of the relative acceleration and the cable angles between the load and the rotorcraft are described below.

In Figure 2.11, the position vector of the load center of mass can be written as the following where l indicates the length of the hypothetical line between the cargo hook and the load center of gravity:

$$\vec{l}^e = l \cdot \begin{bmatrix} C\phi_c S\theta_c \\ -S\phi_c \\ C\phi_c C\theta_c \end{bmatrix}$$
(2.9)

C and S represent operations of cos and sin. The velocity vector of the cable can be expressed as:

$$\dot{\vec{l}}_{E}^{e} = l \cdot \begin{bmatrix} -\dot{\phi}_{c} S \phi_{c} S \theta_{c} + \dot{\theta}_{c} C \phi_{c} C \theta_{c} \\ -\dot{\phi}_{c} C \phi_{c} \\ -\dot{\phi}_{c} S \phi_{c} C \theta_{c} - \dot{\theta}_{c} C \phi_{c} S \theta_{c} \end{bmatrix}$$
(2.10)

The subscript E indicates that the derivative is taken in the inertial (Earth) frame. The acceleration vector of the cable can be expressed as:

$$\vec{\tilde{l}}_{E}^{e} = l \cdot \begin{bmatrix} -\ddot{\phi}_{c}S\phi_{c}S\theta_{c} - \dot{\phi}_{c}^{2}C\phi_{c}S\theta_{c} - \dot{\phi}_{c}\dot{\theta}_{c}S\phi_{c}C\theta_{c} + \ddot{\theta}_{c}C\phi_{c}C\theta_{c} - \dot{\theta}_{c}\dot{\phi}_{c}S\phi_{c}C\theta_{c} - \dot{\theta}_{c}^{2}C\phi_{c}S\theta_{c} \\ -\ddot{\phi}_{c}S\phi_{c} + \dot{\phi}_{c}^{2}C\phi_{c} \\ -\ddot{\phi}_{c}S\phi_{c}C\theta_{c} - \dot{\phi}_{c}^{2}C\phi_{c}C\theta_{c} + \dot{\phi}_{c}\dot{\theta}_{c}S\phi_{c}S\theta_{c} - \ddot{\theta}_{c}C\phi_{c}S\theta_{c} + \dot{\theta}_{c}\dot{\phi}_{c}S\phi_{c}S\theta_{c} - \dot{\theta}_{c}^{2}C\phi_{c}C\theta_{c} \end{bmatrix}$$

$$(2.11)$$

Based on Poisson's kinematical equations [21], the position, velocity, and acceleration vectors of the cable could be expressed in the rotorcraft body frame as:

$$\vec{l}^b = \tilde{T}_{EB} \vec{l}^e \tag{2.12}$$

$$\dot{\vec{l}}_{E}^{\ b} = \dot{\tilde{T}}_{EB}\vec{l}^{e} + \tilde{T}_{EB}\dot{\vec{l}}_{e}^{\ e} = -\tilde{\omega}_{b/e}\tilde{T}_{EB}\vec{l}^{e} + \tilde{T}_{EB}\dot{\vec{l}}_{E}^{\ e}$$
(2.13)

$$\ddot{\vec{l}}_{E}^{\ b} = \ddot{\vec{T}}_{EB}\dot{\vec{l}}_{E}^{\ e} + 2\dot{\vec{T}}_{EB}\dot{\vec{l}}_{E}^{\ e} + \tilde{T}_{EB}\ddot{\vec{l}}_{E}^{\ e} = (\tilde{\omega}_{b/e}^{\ 2} - \dot{\tilde{\omega}}_{b/e} - 2\tilde{\omega}_{b/e})\tilde{T}_{EB}\dot{\vec{l}}_{E}^{\ e} + \tilde{T}_{EB}\ddot{\vec{l}}_{E}^{\ e}$$
(2.14)

Where \tilde{T}_{EB} is the Inertial to body coordinate transform matrix and is calculated following the order yaw (ψ), pitch (θ), and roll (ϕ):

$$\tilde{T}_{EB} = \begin{bmatrix} C\theta_T C\psi_T & C\theta_T S\psi_T & -S\theta_T \\ -C\phi_T S\psi_T + S\phi_T S\theta_T C\psi_T & C\phi_T C\psi_T + S\phi_T S\theta_T S\psi_T & S\phi_T C\theta_T \\ S\phi_T S\psi_T + C\phi_T S\theta_T C\psi_T & -S\phi_T C\psi_T + C\phi_T S\theta_T S\psi_T & C\phi_T C\theta_T \end{bmatrix}$$
(2.15)

The rotation angles are described by the tiltrotor attitude angles $(\psi_T, \theta_T, \phi_T)$. $\tilde{\omega}_{b/e}$ and $\dot{\tilde{\omega}}_{b/e}$ are the angular velocity and acceleration of the tiltrotor relative to the inertial frame expressed in matrix form:

$$\widetilde{\omega}_{b/e} = \begin{bmatrix} 0 & -r & q \\ r & 0 & -p \\ -q & p & 0 \end{bmatrix}$$
(2.16)
$$\dot{\widetilde{\omega}}_{b/e} = \begin{bmatrix} 0 & -\dot{r} & \dot{q} \\ \dot{r} & 0 & -\dot{p} \\ -\dot{q} & \dot{p} & 0 \end{bmatrix}$$
(2.17)

Combing the equations listed above, the relative velocity and acceleration between the load and the aircraft can be written as the following:

$$\vec{v}_L^{\ e} = \dot{\vec{l}}_E^{\ b} + \widetilde{\omega}_{b/e}\vec{l}^b \tag{2.18}$$

$$\vec{a}_L^{\ e} = \ddot{\vec{l}}_E^{\ b} + 2\widetilde{\omega}_{b/e}\dot{\vec{l}}_E^{\ b} + \dot{\widetilde{\omega}}_{b/e}\vec{l}^b + \widetilde{\omega}_{b/e} \times (\widetilde{\omega}_{b/e} \times \vec{l}^b)$$
(2.19)

The relative cable angles are expressed as:

$$\theta_c = \operatorname{atan}\left(\frac{l_x}{l_z}\right) \tag{2.20}$$

$$\phi_c = -\operatorname{asin}\left(\frac{l_y}{l}\right) \tag{2.21}$$

The external force applied to the cargo hook that was generated by the load is described below:

$$\vec{F}_L = \vec{W} + \vec{F}_{aero} + \vec{F}_{inertial} = \vec{T} - m_L \vec{a}_L^{\ e}$$
(2.22)

Where vector \vec{T} contains the recorded hook force from the wind tunnel test that accounts for the weight and the aerodynamic force. The term $m_L \vec{a}_L^{\ e}$ represents the load inertial force and it has the opposite direction to the trim cable tension force. The external moment can be calculated based on the external force and the hook position.

2.5 Linear Analysis of System Stability

The linearized GTR stitched model has 51 states which include rotor dynamics, inflow, fuselage rigid body dynamics, and the nacelle angle dynamics. For the bare aircraft model, the yaw angle ψ was completely decoupled from the rest of the states, hence, the yaw attitude was simply removed from the full order system for linear analysis. The rest of the states were divided into two categories that represent the fast and slow dynamics of the system. The fast dynamics including rotor, inflow, and the nacelle angle dynamics, were assumed to reach steady-state much faster than the fuselage states. The slow dynamics represent the rigid body modes of the rotorcraft that are of major interest as they have lower frequencies in the range important for handling qualities and are unstable at low airspeed flight.

For the bare-airframe model, the full-order model is reduced to an 8-state model to decouple the effects from the rotor states on the fuselage states. The model order reduction process can be formulated as the following.

$$X = [X_{slow}, X_{fast}] \tag{2.23}$$

$$X_{slow} = [u, v, w, p, q, r, \phi, \theta]$$
(2.24)

$$X_{fast} = [rotor states, inflow states, nacelle angle states]$$
(2.25)

$$\dot{X} = \begin{bmatrix} A_{11} & A_{12} \\ A_{21} & A_{22} \end{bmatrix} * \begin{bmatrix} X_{slow}, X_{fast} \end{bmatrix}^T + \begin{bmatrix} B_1 \\ B_2 \end{bmatrix} * u$$
(2.26)

$$\dot{X}_{slow} = \hat{A} * X_{slow} + \hat{B} * X_{slow}$$
(2.27)

$$\hat{A} = A_{11} - A_{12}A_{22}^{-1}A_{21}, \hat{B} = B_1 - A_{12}A_{22}^{-1}B_2$$
(2.28)

The reduced model contains only the rigid body modes of the aircraft and they have much lower frequency compared to the rotor and inflow dynamics. Figure 2.12 plots the low-frequency eigenvalues from both the full-order model and the 8-state model at 120 knots with a 30° nacelle angle. It shows proximity between the low-frequency eigenvalues of the two models. The model order reduction was repeated for multiple combinations of velocity and nacelle angle and the eigenvalues remained close between the two models for all cases. Hence the model reduction is valid for the extraction of the rigid body modes. The eigenvalues are listed in table 2.6.



Figure 2.12. Bare-airframe low-frequency eigenvalues of the full-order model and 8-state model (120 knots, 30° nac).

Full-order Model	8 States Model	Rigid Body Modes
(rad/sec)	(rad/sec)	
-1.718 <u>+</u> 3.33 <i>i</i>	-1.68 <u>+</u> 3.32 <i>i</i>	Longitudinal Short Period
-1.95	-1.96	Roll Subsidence
-0.434, 0.0383	-0.434,0.0383	Longitudinal Phugoid
-0.277 <u>+</u> 1.28 <i>i</i>	-0.278 <u>+</u> 1.29 <i>i</i>	Dutch Roll
-0.123	-0.123	Spiral Mode

Table 2.6. Eigenvalues of the GTR stitched model (120 knots, 30° nacelle angle).

Figure 2.13 plots the eigenvalues migration of the rigid body modes along with the speed envelope. The longitudinal modes include short period mode and phugoid. The short period mode is represented by a pair of complex conjugate poles that have a high damping ratio and faster frequency compared to the other modes. The frequency of the mode increases with airspeed, and the damping ratio decreases. The phugoid mode usually represents a slow rocking motion in the longitudinal axis and is an oscillatory unstable mode during hover and low-speed flights, which can be identified by the grouping of eigenvalues with a positive real part at the low-speed range. In forward flight, the oscillatory mode divides into two real eigenvalues on the real axis with one of the poles being on the right side of the real axis. The lateral modes include roll subsidence, Dutch Roll, and spiral mode. The Dutch Roll mode is oscillatory. The frequency of the mode increases with increasing speed, and the damping ratio of the mode remains nearly fixed across the envelope. The roll subsidence is represented by a stable pole on the real axis that has a quick transient response to perturbation while the spiral mode is a stable real pole with a small frequency as it is close to the origin.



Figure 2.13. Airspeed sweep of the bare-airframe low-frequency eigenvalues.

The coupled system contains two extra second-order equations from the load longitudinal and lateral transfer functions, which results in four extra states to be included by the system as shown in Equation 2.29. Figure 2.14 shows the eigenvalues of the load-coupled system being plotted alongside the bare-airframe model. From the plot, it can be seen that the coupling effect between the load and the rotorcraft at such airspeed is fairly small, as indicated by the proximity between the poles of the isolated and coupled systems. As listed in table 2.7, the comparison between the eigenvalues of the same mode before and after coupling the load can draw some insight into how the system dynamics are changed due to the coupling effect. The longitudinal short period mode shows a small increase of its damping ratio (from 0.452 to 0.458) and frequency (from 3.72 rad/s to 3.92 rad/s). The lateral Dutch Roll mode has its damping ratio decreased from 0.211 to 0.184 and frequency from 1.32 rad/s to 1.28 rad/s. The Yaw/Spiral mode becomes an oscillatory mode after the load is added as the aircraft heading angle is now coupled into the system dynamics. In a general view, the aircraft is destabilized due to the coupling effect as the damping ratios of the low-frequency modes are decreased. The damping ratio of the longitudinal pendulum mode increases by about 60% (from 0.0582 to 0.0926), and the lateral pendulum mode increases by 206% (from 0.0373 to 0.114). Therefore, the load pendulum motions are stabilized by the rotorcraft from the extra damping provided by the inertial coupling.

$$X_{SLOW} = [u, v, w, p, q, r, \varphi_T, \theta_T, \psi, \dot{\phi}_c, \phi_c, \dot{\theta}_c, \theta_c]$$
(2.29)



Figure 2.14. Low-frequency eigenvalues of the load-coupled model (120 knots, 30° nacelle angle, firing configuration).

Stitched Model	Freq.(rad/s)/Damp	Coupled Model	Freq.(rad/s)/Damp.	Dynamic
				Modes
-1.68 <u>+</u> 3.32i	3.72/0.452	-1.80 <u>+</u> 3.48i	3.92/0.458	Short Period
-1.96	1.96/1	-1.86	1.86/1	Roll Subsidence
-0.434, 0.0383		-0.395,0.0384		Phugoid
-0.278 <u>+</u> 1.29i	1.32/0.211	-0.236 <u>+</u> 1.26i	1.28/0.184	Dutch Roll
-0.123	0.123/1	-0.057 <u>+</u> 0.041i	0.0701/0.813	Spiral/Heading
-0.0787 <u>+</u> 1.35i	1.35/0.0582	-0.139 <u>+</u> 1.49i	1.5/0.0926	Long. Pendulum
(Isolated Load)				
-0.0497 <u>+</u> 1.33i	1.33/0.0373	-0.188 <u>+</u> 1.64i	1.65/0.114	Lat. Pendulum
(Isolated Load)				

Table 2.7. Eigenvalues of the stitched model and load coupled model (120 knots, 30° nacelle angle).

An airspeed sweep of the load-coupled model was conducted for both load configurations (firing and folded), and the resulting eigenvalues are plotted in Figures 2.15-16. The plots show a similar migration pattern of the system's low-frequency eigenvalues compared to the bare-airframe model. The load pendulum modes remain independent of airspeed as seen from the close groupings on both configurations. The unstable character of the Phugoid mode is unchanged and does require stability augmentation for a stable flight with the load present.



Figure 2.15. Airspeed sweep of load-coupled model low-frequency eigenvalues (firing configuration).



Figure 2.16. Airspeed sweep of load-coupled model low-frequency eigenvalues (folded configuration).

Chapter 3 Controller Design

3.1 Introduction

This chapter includes the controller design for active stabilization of the M119 external load and the design of a SAS controller for the GTR model. The preliminary design of the ACH controller for the M119 slung load applies classical control design methodology. The design uses longitudinal and lateral cable angle feedback to damp longitudinal and lateral load pendulum motions by translation of the active cargo hook. The preliminary design is based on tuning the controller to achieve maximum load damping while meeting desired stability margins. The ACH controller is a direct control mechanism that automatically varies the hook position to improve load damping and system stability without pilot inputs. As described earlier, the idea of using onboard active actuator control was examined in the past research for hover and low-speed flights [22, 23], and demonstrated in flight [6]. The current paper focuses primarily on controller performance during high-speed flights where the potential for load instability exists.

For the tiltrotor, a basic stability augmentation system is required to provide minimum stability for the aircraft. Although the SAS design is not the focus of the research, it is needed for the investigation of the effectiveness of the load stabilization. Unstable or barely stable aircraft will perturb the performance of the ACH controller. The design of the SAS controller uses a linearquadratic regulator (LQR) to stabilize the rotorcraft.

3.2 Control Design for the Isolated Load

As mentioned previously, the slung load pendulum dynamics were obtained as linear models identified from open-loop frequency sweep tests in the wind tunnel. Test results showed that the cable angle response to ACH translation correlated highly with on-axis responses (θ_c/x , ϕ_c/y) and off-axis responses were negligible. The identified linear model parameters (K, ζ_p , ω_P , Δt) were scheduled by airspeed due to the dependence of system dynamics on dynamic pressure.

Figure 3.1 shows a model of the open-loop system with the active cargo hook and external slung load. The active cargo hook is modeled as a first-order actuator with a bandwidth of 20 rad/s determined by the time constant (τ_a). A time delay is included in the slung load dynamic model to represent actuator command and sensor delays. The time delay was neglected during the design process but would be included later to analyze closed-loop stability.



Figure 3.1. Model of the open-loop system with active cargo hook and external slung load.

The open-loop transfer functions can be converted into two sets of equations of motion representing the longitudinal and lateral axes of the load pendulum motion as shown in Equations (3.1-4).

$$\ddot{\theta}_c + 2\zeta_p \omega_P \dot{\theta}_c + \omega_P^2 \theta_c = K_P \ddot{x}$$
(3.1)

$$\tau_a \cdot \dot{x} + x = x_{cmd}(t - \tau) \tag{3.2}$$

$$\ddot{\phi_c} + 2\zeta_p \omega_P \dot{\phi_c} + \omega_P^2 \phi_c = K_P \ddot{y}$$
(3.3)

$$\tau_a \cdot \dot{y} + y = y_{cmd}(t - \tau) \tag{3.4}$$

It can be seen that cable angle motion is related to the acceleration of the cargo hook (\ddot{x}, \ddot{y}) for each axis. Tables 2.3-4 show that the damping of the load pendulum motion increases with airspeed, especially for the lateral axis. As indicated earlier, at high airspeeds (greater than 122kt FS for the firing configuration and 174 kt FS for the folded configuration), the non-linear behavior of the load such as LCO could not be captured by the linearized models. However, it was assumed that if the controller was designed for airspeeds where the load was lightly damped, then the same controller would stabilize the load and prevent it from ever reaching the LCO in the unstable regions. This was ultimately proven to be true in experiments as discussed in Chapter 4.

The load at the hover condition was selected to demonstrate the controller design approach since it had the smallest damping ratio of any stable condition. The firing configuration was selected as the primary configuration for controller design. However, the pendulum dynamics for the firing and folded configurations are similar. The lateral axis controller design was considered before the longitudinal axis. Figure 3.2 shows the step response of the open-loop system at hover for the firing configuration and lateral axis, which demonstrates the low damping of the pendulum mode without any compensation.



Figure 3.2. Open-loop simulation of lateral pendulum motion step response, (hover, firing configuration).

Previous investigations of helicopter external slung loads showed that dynamic compensation was effective in increasing system stability and reducing swinging motion of the load at low airspeed [22,24]. Thus, the use of dynamic compensation was considered for the ACH controller design. The block diagram in Figure 3.3 shows the model of the closed-loop system. Lead and lag compensation were both considered for the controller design. The controller filter, compensator, and gain parameter values for the lateral pendulum controller are listed in Table 3.1. The design process that determined the choices of the parameters is discussed in the following.



Figure 3.3. Block diagram of the closed-loop system.

	Filter G(s)	Compensator C(s)	Gain K
Lead Design	$\frac{1}{s+p}$, p = 7.04 $\frac{rad}{s}$	S	$-4.12, \frac{\text{mm}}{\text{deg}}$
Lag Design	$\frac{s}{s+p}$, $p = 0.1 \frac{rad}{s}$	$\frac{1}{s+d}, d = 1.85 \frac{rad}{s}$	28.6, $\frac{\text{mm}}{\text{deg}}$

Table 3.1. Controller parameters for the lateral pendulum mode (hover, firing configuration).

Using proportional derivative (PD) feedback has the benefit of improving system damping and providing a quicker transient response. For the lead design, the controller transfer function is composed by combining a pure derivative term (*s*) and low pass filter $(\frac{1}{s+p})$, which is used to decrease controller response to high-frequency disturbances to cable angle. The lead design uses a negative gain value that shifts the phase of the output of the controller by 180 deg so that the cargo hook motion is opposite in sign to the lateral cable angular rate.

Lead compensator parameters were obtained by manual tuning using the control system toolbox in MATLABTM. The design started by placing a system zero at the origin and the pole of the lag filter at the pendulum natural frequency. The lead was gradually adjusted until the shape of the pole trajectories became stable and well-damped. The compensation induces phase lead at

the resonance of the swinging load by driving the cargo hook to move in lead of the load response. Figure 3.4 shows the root locus plot of the closed-loop system with the lead design. The closed-loop pole trajectories of the pendulum mode shift toward the left of the s-plane, thus improving system stability and damping. However, the closed-loop system now has an extra set of poles due to the mixing of the actuator and the low pass filter that could shift to the unstable region when the gain is high.



Figure 3.4. Root locus plot of the lateral pendulum mode with lead compensation, (hover, firing configuration).

The broken loop frequency response in Figure 3.5 shows two sets of phase margins occurring at two different gain crossover frequencies. The smallest between the two will be selected as the minimum margin. The delay margin (DM) indicates the tolerance of the system towards time delay

and is calculated based on the phase margin (PM) and the gain crossover frequency (ω_{gc}) of the closed-loop system as shown in Equation 3.5.

$$DM = \frac{PM}{\omega_{ac}} * \frac{\pi}{180^{\circ}} (s) \tag{3.5}$$

For a phase margin of 22.4 deg at 7.71 rad/s, the resulting delay margin (DM) of 0.0506s just barely surpasses the measured time delay (0.0246s). Additional time delay has a destabilizing effect on the control system and decreases the effective damping of the controller. Therefore, a higher delay margin represents good system robustness toward a change of delay factors during operation.



Figure 3.5. Broken-loop Bode diagram of the lateral pendulum mode with lead compensation, (hover, firing configuration).

Figure 3.6 shows the simulated closed-loop step response of the load lateral pendulum motion with lead compensation. Compared with the open-loop step response, there is a significant decrease in the time required for the lateral cable angle to settle back to the zero position, and the oscillation of the lateral angle is damped quickly, despite an increase in overshoot during the initial transient response. The second plot shows the lateral control command to the ACH actuator δy . The control signal moves in the opposite sign to the lateral cable angle and has a lead tendency. The time history plot shows that the lead design improves the damping of the lateral pendulum mode. However, as shown by the Bode plot, small stability margins reflect weak robustness, which imposes a higher risk of the system becoming unstable due to changes in load and control system parameters.



Figure 3.6. Closed-loop simulation of lateral pendulum motion step response with lead compensation, (hover, firing configuration).

Another approach for slung load control is to use lagged cable angle feedback (LCAF). In Ref. [4], it was shown in the simulation that a compensator with a lag filter could be tuned to achieve proper phasing between load swing and control input to dampen the swing motion at hover and low-speed flights. The approach was modified for application to the ACH controller. A washout filter $\left(\frac{s}{s+p}\right)$ was used to remove actuator response to a change of the steady-state trailing angle, which occurs during a change in flight speeds. The lag compensator $\left(\frac{1}{s+d}\right)$ created additional phase lag in the system over the inherent phase delay from the actuator and provided filtering of high-frequency signals from the sensor. This control structure was demonstrated in flight on an H-6 helicopter by Boeing and the U.S. Army Technology Development Directorate [6]. The parameters of the controller are listed in Table 3.1 for the selected hover case. The washout filter frequency was chosen to be very small, and the lag compensator frequency was manually tuned to achieve the proper phasing. The lag compensator has an opposite sign on the gain value compared to the lead compensator.



Figure 3.7. Root locus plot of the lateral pendulum mode with lag compensation, (hover, firing configuration).

The root locus plot for the lag design (Figure 3.7) shows better damping compared to the lead design and all the closed-loop trajectories are in the stable region. The broken-loop response (Figure 3.8) shows significant increases in the gain margin and phase margin as compared to the lead design. The minimum delay margin of 0.1755 s also increases significantly and is well above the measured time delay of the identified linear model (0.0246 s). The negative phase margin indicates that the closed-loop system can be unstable if phase lead is added to the system around the corresponding gain crossover frequency. This is reasonable as the controller is intentionally adding phase lag to phase the control response to the swinging load.



Figure 3.8. Broken-loop Bode diagram of the lateral pendulum mode with lag compensation, (hover, firing configuration).

The higher damping ratio achieved by the lag design is reflected by the time history plot comparison (Figure 3.9) of the simulated closed-loop system for the two controllers. The lateral cable angle response shows a smaller amplitude of the oscillation and shorter settling time compared with the lead design. The actuator usage is also significantly reduced for the lag design as shown in the second plot. Therefore, the lag compensation has better performance in terms of system damping and stability than the lead design at hover for the firing configuration.



Figure 3.9. Closed-loop simulation of the lateral pendulum motion step response with lag and lead compensation, (hover, firing configuration).

3.4 Airspeed Scheduled Control Design for High-Speed Flight Conditions

Lag compensation shows an improvement of the damping of the load motion for the hover condition for the firing configuration and lateral axis. The next design objective is to tune controller parameters for forward-flight conditions and analyze controller performance. During the design process, it was discovered that controller performance was not very sensitive to increases in airspeed. For the controlled system, the damping of the pendulum modes varied over a range of 0.5-0.8 with a trade-off between overshoot and settling time. The gain values for other conditions also required relatively small adjustments for maximum damping. Therefore, the controller

designed for the hover condition was tested for forward-flight conditions, and the responses were found to show only small sensitivity to airspeed.



Figure 3.10. Closed-loop simulation of the lateral pendulum motion step response with lag compensation, (multiple airspeeds, firing configuration).

Figure 3.10 shows the time history of the lateral cable angle at various airspeeds subjected to the same step input as the hover condition. The three non-zero airspeeds, corresponding to 52, 87, and 122 knots in full-scale condition, show very close responses compared with the hover case using the same controller. Therefore, the compensation proved to be effective throughout the flight envelope where the linear models were available.

Lag compensation shows effectiveness in increasing damping and stability of the load lateral pendulum motion in both hover and forward flight. As previously mentioned, firing and folded configuration pendulum dynamics are very similar. Additionally, the pendulum dynamics are similar for the longitudinal and lateral axes in both configurations.



Figure 3.11. Root locus plot of the longitudinal pendulum mode with lag compensation, (hover, firing configuration).


Figure 3.12. Broken-loop Bode diagram of the longitudinal pendulum mode with lag compensation, (hover, firing configuration).

Figures 3.11-12 show the root locus plot and broken-loop frequency response of the longitudinal pendulum mode with the designed controller at hover and firing configuration. For the longitudinal axis at all airspeeds, the closed-loop systems all have a damping ratio above 0.5. This implies that a universal controller could work for both axes across all airspeeds.

The controller gain for the folded configuration was slightly adjusted to achieve maximum damping for the hover case and the controller performance was verified in other airspeeds. Table 3.2 lists the airspeed independent controller parameters for the firing and folded configurations and the lateral and longitudinal axes, they were designed for the hover condition for maximum damping in both lateral and longitudinal axes.

	Firing (Lon)	Firing (Lat)	Folded (Lon)	Folded (Lat)
K (gain) (mm/deg)	-28.6	28.6	-29.0	29.0
d (lag) (rad/sec)	1.85	1.85	1.85	1.85
p (washout) (rad/sec)	0.10	0.10	0.10	0.10

Table 3.2. Preliminary ACH controller parameters.

Tables 3.3-4 list the system stability margins for two airspeeds in model-scale with the lag compensation. They show that the proposed controller can provide good damping and stability to the external slung load in both configurations under different airspeeds without gain scheduling.

Airspeed (MS)	Axis	GM (dB)	PM (deg)	DM (sec)
6 m/s	θ_c/x_{cmd}	38.1	-78.3	0.186
	ϕ_c/y_{cmd}	36.6	-74.3	0.173
14 m/s	θ_c/x_{cmd}	39.7	-88.0	0.187
	ϕ_c/y_{cmd}	38.7	-82.5	0.175

Table 3. 3. System stability margins in the firing configuration.

Airspeed (MS)	Axis	GM (dB)	PM (deg)	DM (sec)
6 m/s	θ_c/x_{cmd}	36.4	-74.1	0.175
	ϕ_c/y_{cmd}	36.2	-74.0	0.164
14 m/s	θ_c/x_{cmd}	36.1	-75.4	0.193
	ϕ_c/y_{cmd}	36.8	-80.4	0.147

Table 3.4. System stability margins in the folded configuration.

The last step of the design is to verify the robustness of the system to time delay uncertainty. The measured time delays of various linear models are all below 0.025 s with the largest being 0.0246 s at hover. The system was simulated in Simulink[™] with an integrated transport delay block acting as the delay factor.



Closed-Loop Lateral Pendulum Motion Step Response with Time Delay

Figure 3.13. Closed-loop simulation of the lateral pendulum motion step response with time delay, (hover, firing configuration).

Figure 3.13 shows the step response of the closed-loop system at hover for the firing configuration. The orange line represents the lateral response of the load with a time delay equal to the delay margin (0.175 s). The response is highly oscillatory, which is due to the decrease of the effective damping of the control system caused by the time delay. The delay factor is slowly increased until the system becomes unstable (shown in blue line), which happens at $\tau = 0.181$ s. This confirms that the tolerance of the system towards time delay is around the calculated delay margin by a small deviation. The purple line represents the system with the actual time delay from the linear model and the result shows that the damping and stability of the system are not affected by such time delay, which assures good robustness of the control system during operation.

3.5 GTR Stability Augmentation System (SAS) Control Design

The linear analysis of the GTR model shows an unstable dynamic mode of the system at some flight conditions. To test the effect of the designed ACH controller on the load-coupled model, having a stable platform is required so that the performance of the controller would not be affected by the instability of the aircraft. Considering most modern rotorcraft have onboard flight control systems such as the AFCS (Automatic Flight Control System), which provides stability and improves handling qualities of the aircraft, it is necessary to design a controller for the GTR model for flight simulation.

Ref. [25] showed that using optimal control technique was an effective design for rotorcraft. It demonstrated the design of an LQR (Linear Quadratic Regulator) on a generic non-linear helicopter model and showed the success of improving the damping of the rotorcraft rigid body modes and the advantages of using optimal controller design compared to conventional controllers. For LQR problems, it requires to specify the weighting factors for the system states and the control input [25]. The system is usually modeled as a state-space linear system as the following:

$$\dot{\hat{x}} = A\hat{x} + B\hat{u} \tag{3.6}$$

$$\hat{y} = C\hat{x} + D\hat{u} \tag{3.7}$$

A gain matrix K that gives the optimal feedback control vector needs to be determined:

$$u_{fb} = -K\hat{x} \tag{3.8}$$

Such that the result of the cost function shown below is minimized [25]:

$$J = \frac{1}{2} \int_{0}^{t_{f}} [\hat{x}^{T} Q \hat{x} + \hat{u}^{T} R \hat{u}] dt$$
(3.9)

Where the matrices Q and R are the weighting matrices for the state vectors and the control vectors. The matrix Q is chosen to be positive semidefinite, and R to be positive definite. Symbolically, these are expressed as [26]:

$$Q = Q^T \ge 0, R = R^T > 0 \tag{3.10}$$

The feedback matrix K can be solved via a Lagrange multiplier-based optimization technique and is given by [25]:

$$K = (R^{-1})B^T P (3.11)$$

Where P is a non-negative definite matrix satisfying the Algebraic Riccati equation:

$$A^{T}P + PA + Q - PBR^{-1}B^{T}P = 0 (3.12)$$

It is of major interest to the author to use a simple and effective controller to stabilize the tiltrotor model at hover and forward flight conditions, and also being able to define the weightings on the aircraft states and the control inputs. Therefore, optimal control offers the simplicity and the adjustability for the author to construct a preliminary SAS design for the GTR stitched model.

The design of the controller focuses on using minimum control effort to move the unstable poles of the aircraft to the stable region without over-stabilizing the modes. Ref. [26] showed an LQR design that includes using rate command to stabilize the rotorcraft by defining constraints on the angular rates and the control inputs. The weighting matrices of the LQR are computed as the following [27]:

$$Q(i,i) = \frac{1}{{\sigma_s}^2}, \quad R(i,i) = \frac{1}{{\sigma_c}^2}$$
 (3.13)

where the elements of the matrices are defined by the allowed perturbations of the states σ_s and control inputs σ_c from their trim values. For the current task, changes in the aircraft velocity and the attitude angles are not very important as the focus of the controller is to stabilize the aircraft motion. Using rate command, the Q matrix is defined with constraints on the angular rates (p, q, r) that have deviations equal to 30 deg/s, and the R matrix is defined with constraints on the 4 control inputs $(u_{lon}, u_{lat}, u_{col}, u_{ped})$ that have deviations equal to 10%. The SAS design was conducted on the GTR bare-airframe model with the integrated control allocation module. The designed gain matrix K is airspeed-scheduled from hover to 260 knots and is designed for multiple nacelle angles at specific airspeeds in the conversion corridor. The SAS was integrated into the load-coupled model.

Figure 3.14 plots the low-frequency eigenvalues of the system at hover and 120 knots forward flight for firing configuration. At hover, the unstable Phugoid and Dutch Roll modes are stabilized as the pole locations are moved to the left side of the imaginary axis. The effect of the SAS on the load pendulum modes is not significant as the poles shift slightly from before. This is expected since the stabilization of the rotorcraft prevents the energy exchange between the load and the rotorcraft and the damping of the load motions through the aerodynamic damping provided by the rotors. At forward flight, the unstable Phugoid mode is moved to the negative real axis, the damping of the oscillatory modes such as the Short Period and the Dutch Roll modes are slightly increased, and the other modes remain close to the positions before applying the SAS.



Load-Coupled Model Low-Frequency Eigenvalues with SAS (120kt, 30nac)



Figure 3.14. Eigenvalues of the load-coupled model (firing configuration) with the SAS controller: (a) Hover, (b) 120 knots, 30° nacelle angle.

3.6 Control Design Conclusion

This chapter demonstrated the controller design process for the M119 isolated load and the GTR bare-airframe model. For the M119 load, the design included two concepts (lead and lag designs) that both used dynamic compensation to stabilize the load pendulum motion. Both designs are proven to be able to increase the load damping across the tested speed range for both firing and folded configurations. The lag design was chosen as it provides better stability margins, which guarantees the close-loop system is more robust under the change of conditions. It was found that a universal controller was able to increase the damping of the load pendulum motions in both the longitudinal and lateral axis. The simulation results showed a close response of the load cable angles across multiple airspeeds from hover to 122 knots while applying the same controller, and it was expected that the load would be stabilized by such controller at higher airspeed where stable linear load models were not able to be identified.

A SAS using an LQR controller was designed for the GTR bare-airframe model that provides basic stability to the aircraft when the load is coupled. The SAS was able to stabilize the lowfrequency phugoid and Dutch-Roll modes by moving the poles of the system to the stable region without over-stabilizing them. The eigenvalues of the load-coupled model showed that the SAS has minimum effect on the load pendulum modes, which is important for isolating the control effect of the SAS and the ACH controller on the load pendulum dynamics.

Chapter 4 Simulation Results

4.1 Introduction

This chapter discusses the closed-loop wind tunnel tests performed with the integration of the designed ACH controller and the simulation results of the GTR-load-coupled system. The closed-loop wind tunnel tests were conducted by the researchers from Technion University, who had performed the open-loop tests previously for the load. The designed universal ACH controller was integrated into the dynamic wind-tunnel set-up and tested for both firing and folded load configurations across the flight envelope from hover to 200 knots for full-scale flight conditions. In order to validate the performance of the ACH controller, the same airspeed sweeps as previously conducted in the open-loop tests were repeated for the closed-loop system. Following the airspeed sweeps, a set of inputs involving frequency-sweep input and gust disturbance were fed into the system to excite the cargo hook at different airspeeds while the controller remained active. This test was performed to provide samples for comparison with the simulation results.

The derived rotorcraft-load-coupled system was simulated in Simulink with different configurations. First, an isolated load model without coupling to the rotorcraft was simulated with the same hook commands used in the closed-loop wind tunnel tests. The load response was compared with the closed-loop test results to verify the accuracy of the simulation model. Then, the GTR and load model was simulated with a lateral pilot control input. The input was applied to three different model configurations including the bare-airframe, the coupled system without the

ACH controller, and the coupled system with the ACH controller. The models were simulated at hover and 120 knots for forward-flight.

4.2 Comparison with Closed-Loop Wind Tunnel Test

It is necessary to verify the closed-loop response of the load model with the integration of the ACH controller to ensure the simulation model is credible. The closed-loop wind tunnel test was performed to test the controller across the flight envelope, especially at higher airspeed where the load was unstable. The designed controller was implemented into the control loop of the moving hook, the computer received the signals from the sensor that measured the cable angles and calculated the corresponding control inputs to the hook. Ref. [14] recorded an in-depth analysis of the closed-loop test results and concluded that the designed controller was able to stabilize the tested load model from hover to 200 knots for both load configurations. Part of the results is shown in the following including the airspeed sweep of the load folded configuration and time validations of the simulation model.

Figure 4.1 shows the load responses in terms of attitude angles, cable angles, and the ratio between the cable force and load weight during a slow increase of the tunnel speed from 87 knots up to 190 knots for full-scale flight conditions [14]. The response of the closed-loop system with the ACH controller and the response from the open-loop test are shown in figures (a) and (b) respectively for better comparison. The behavior of the closed-loop load steady-state values is shown to be similar to the load behavior during the open-loop test. The stability of the load at high airspeeds is much improved as the controller suppresses the LCO of the load over the speed range where it was observed previously. The results of the airspeed sweep demonstrated that the designed

controller was able to stabilize the load at a speed range higher than the maximum speed from the identified linear models.



Figure 4.1. (a): Closed-loop system load response to increasing wind-tunnel speed with the ACH controller (folded configuration) (from [19]). (b): Open-loop system load response to increasing wind-tunnel speed without the ACH controller (folded configuration) (from [14]).

Figure 4.2 shows the load firing configuration undergoing the same airspeed sweep procedure from 87 knots to 165 knots for full-scale flight conditions. The maximum tunnel speed was lower than the folded configuration due to the reason that the load longitudinal trailing angel exceeded 42 degrees, which was close to the predetermined safety limit of 45 degree [14]. The controller tested for the firing configuration was slightly modified in order to test the optimized version of the ACH controller derived by TDD. The optimization was based on the preliminary designed controller and the parameters of the controller were fine-tuned using the fight control design software CONDUIT®, which uses a specification-driven optimization approach that enforces all performance requirements with minimum actuator usage. The optimization procedure was presented in Ref. [14] and will not be discussed here. Comparing the behavior of the load at firing configuration with and without the ACH controller, the closed-loop system is shown to be much more stable as the load responses remain close to their steady-state values at higher airspeeds, where severe instability above 130 knots was observed during the open-loop tests. The preliminary controller was also tested with the firing configuration and showed similar behavior as the optimized controller [14].



Figure 4.2. (a): Closed-loop system load response to increasing wind-tunnel speed with the ACH controller (firing configuration) (from [14]). (b): Open-loop system load response to increasing wind-tunnel speed without the ACH controller (firing configuration) (from [14]).

The following figures present the time domain verification of the simulation model against the wind-tunnel model for the closed-loop system. The cargo hook was excited with a series of inputs that include frequency-sweep and gust disturbance on both longitudinal and lateral directions across the flight envelope while the ACH controller was turned on. The same inputs were later applied to the simulation model and simulated for the same conditions as the windtunnel tests.

Figure 4.3 shows the comparison between the simulated response of the closed-loop system and the actual wind tunnel results due to a longitudinal hook excitation at 14 m/s wind tunnel speed, which is equal to 122 knots in full-scale flight condition. The system was first applied with excitation in the longitudinal direction and then in the lateral direction. The first plot records the section of the cargo hook being driven in the longitudinal direction (X) with increasing frequency and followed by a gust disturbance. The ACH controller was active during both the simulation and the actual test. The second plot shows the response of the longitudinal cable angle. The simulation curve shows reasonable agreement with the test results in terms of frequency and magnitude of the response. At the higher frequency range, the simulation curve has a small discrepancy with the test result such that the higher-order response of the cable angle was not approximated accurately by the linear model. The third plot shows the lateral cable angle response. Notice that there was no input given in the lateral direction and the small oscillation of the lateral cable angle was due to the fluctuation of the lateral hook position. The tension force curves recorded in the fourth plot show good approximation from the simulation model as the low order response was tracked nicely by the red line.

Figure 4.4 shows the comparison between the simulated response of the closed-loop system and the wind tunnel results due to a lateral hook excitation at 14 m/s (122 kt FS) wind tunnel speed.



Comparison between Closed-Loop Wind-Tunnel Test and Simulation at 14 m/s (122 kt FS) for Firing Configuration

Figure 4.3. Comparison between the closed-loop wind-tunnel test and the simulation results at 14m/s (122 knots full-scale) with longitudinal hook excitation (firing configuration).

The same types of input were applied to the cargo hook in the lateral direction. The second plot shows a small variation of the longitudinal cable angle due to fluctuation from the longitudinal hook position. The simulated lateral cable angle shown in the third plot reveals good agreement between the simulation result and the test result throughout the entire frequency range of the applied inputs. The tension force in the fourth plot shows reasonable agreement in the low-order force dynamics. The spikes from the black line could represent the elastic modes that are not captured by the low-order linear model.



Figure 4.4. Comparison between the closed-loop wind-tunnel test and the simulation results at 14m/s (122 knots full-scale) with lateral hook excitation (firing configuration).

The comparison test was also performed at other airspeeds and the results showed credible approximation to the simulation model. The comparison between the simulated result and the test result verified that the simulation model was able to produce an approximate response of the load due to the motion of the cargo hook. The ability of the designed ACH controller to stabilize the load pendulum motions at high airspeeds was proven by the closed-loop wind tunnel tests as well as by the simulation results.

4.3 GTR-Load Coupled System Simulation Results

Following the time domain validation of the simulation model, the GTR-load-coupled system was simulated with a lateral pilot input to observe the behavior of the coupled system with and without the controller. The models are simulated with three configurations including the bare-airframe, the load-coupled airframe without the controller, and the load-coupled airframe with the controller. Results of the simulation are shown for three different airspeeds that include 120 knots, 87 knots, and hover for full-scale fight conditions. The SAS was turned on throughout the entire simulation process.

4.3.1 Simulation at 120 knots

This section shows the simulation results of the load-coupled system operating at 120 knots with a nacelle angle of 30 deg. This speed approaches the envelope of high-speed slung load carriage and it is also the highest airspeed where a linear load model could be identified. The system was first simulated with a +/-30% lateral stick doublet input that was initiated at 3 seconds after the start of the simulation. Figure 4.5 shows the response of the aircraft attitude angles for the three configurations subjected to the same lateral input. The black lines indicate the aircraft responses when there was no load attached, and it shows that the aircraft was stable under the excitation from the control input which was expected since the tiltrotor behaves like a fixed-wing aircraft at such speed and is dynamically stable. The blue lines represent the aircraft response after the load was attached and show clear oscillation of the aircraft roll and yaw angles. The aircraft

pitch angle settles at a new trim value after the load was coupled and shows a small variation to the control input. The aircraft roll attitude trims at a nonzero value after the load was coupled. The red lines show the aircraft response with the ACH controller being active. The aircraft roll and yaw angles are shown to quickly flatten out after the initial peak, which indicates the damping of the aircraft lateral rigid body modes was increased due to the action of the ACH controller.



Figure 4.5. Tiltrotor attitude angle response to a 30% lateral doublet input at 120 knots, 30° nacelle angle.

Figure 4.6 shows the cable angle responses to the lateral doublet input. The black lines indicate the cable angles of the isolated load model after decoupling from the aircraft motion. The same cargo hook excitations were applied for the isolated load as for the other configurations. The cable

angle responses of the load-attached configuration display higher initial amplitude in the longitudinal axis due to the effect of the aircraft motion. The responses also show a higher damping ratio than the load-isolated configuration in both longitudinal and lateral axes. This is expected as the stability analysis shows that the damping of the load pendulum modes was increased by the coupling effect from the aircraft. The controller is shown to quickly reduce the cable angle magnitudes but triggers small oscillations with faster frequencies in both lateral and longitudinal directions.



Figure 4.6. Load cable angle response to a 30% lateral doublet input at 120 knots, 30° nacelle angle.

Figure 4.7 shows the cargo hook position in the longitudinal (X) and lateral (Y) axis. The actuation of the cargo hook is constrained to +/-3 ft from its attachment point to the aircraft in both lateral and longitudinal directions. High-frequency signals can be observed from the controller inputs, which are not desired as it may saturate the actuator and fatigue the mechanical components.

This is predicted to be associated with the gain value designed for the ACH controller and is expected to be improved by reducing the gain.



Figure 4.7. Cargo hook control action to a 30% lateral doublet input at 120 knots, 30° nacelle angle.



Figure 4.8. Load cable angle response and cargo hook control action with a gain-reduced controller.

Figure 4.8 displays the simulation results of the closed-loop system with the controller gain being reduced by 30% from the preliminary design value. The cable angle responses show slightly higher overshoot in both longitudinal and lateral axes compared with the preliminary design. The responses show a similar damping ratio with the gain-reduced controller as the preliminary design. The high-frequency oscillations that appeared in the preliminary design are eliminated after reducing the controller gain value. The cargo hook controller action also shows a low-frequency signal in both longitudinal and lateral axes. This proves the previous statement that the highfrequency oscillations in cable angles and controller action were due to the high controller gain from the preliminary design. Reducing the gain value helps to compensate for the high-frequency signals and relieving the controller effort.

4.3.2 Simulation at 87 knots

The load-coupled system was simulated at 87 knots to assess the controller performance at a transition condition between low and high-speed flights. The same lateral doublet input was applied to the system as the previous case. Figure 4.9 shows the Euler angles of the aircraft. The roll angle has a higher initial peak magnitude compared with the aircraft at 120 knots. It could be observed that the coupling effect destabilizes the aircraft and the ACH controller has a positive effect on stabilizing the aircraft motion by increasing the damping of the load pendulum motions. The cable angle response shown in Figure 4.9 is similar to the previous test speed such that the damping of the cable pendulum motion is increased by coupling with the aircraft and it is further increased when the ACH controller is turned on. The control action from the cargo hook in Figure 4.10 shows similar control actions as the previous case.



Figure 4.9. Tiltrotor attitude angle response to a 30% lateral doublet input at 87 knots, 70° nacelle angle.



Figure 4.10. Load cable angle response to a 30% lateral doublet input at 87 knots, 70° nacelle angle.



Figure 4.11. Cargo hook control action to a 30% lateral doublet input at 87 knots, 70° nacelle angle.

4.3.3 Simulation at hover

At hover, the aircraft response shows larger oscillation in its Euler angles that resulted from the low damping ratio of the rigid body modes. The cable angle response shows a small damping ratio of the load, which is expected as the damping of the identified load model at hover has the smallest value among all the airspeeds. The controller is shown to quickly stabilize the cable pendulum motions. However, small oscillations can still be observed, and the aircraft yaw angle shows different steady-state value when the controller is active. The control action is shown to require more time to stabilize the load.



Figure 4.12. Tiltrotor attitude angle response to a 30% lateral doublet input at 5 knots, 90° nacelle angle.



Figure 4.13. Load cable angle response to a 30% lateral doublet input at 5 knots, 90° nacelle angle.



Figure 4.14. Cargo hook control action to a 30% lateral doublet input at 5 knots, 90° nacelle angle.

4.3 Simulation Results Conclusions

The closed-loop wind tunnel test verified the performance of the controller as the load was stabilized by the designed ACH across the flight envelope. The comparison of the test results with the simulation results indicated a credible simulation model of estimating the linear dynamics of the load system. For the GTR-load-coupled system, the ACH controller was then simulated at three airspeeds with three different configurations. The coupled system appeared to have the highest damping ratios of its rigid body modes and load pendulum modes when the ACH controller was turned on. The designed ACH controller showed high-frequency input signals during stabilization and caused small oscillations on the aircraft attitude angles and the load cable angles. This is expected to be resolved after reducing the gain value of the controller. The target airspeed for the system is 260 knots where the linear load model is not available. However, it is expected that the designed ACH controller will remain effective up to the target speed, although it might require modifications of control gain values in the future.

Chapter 5 Conclusions and Future Work

5.1 Conclusions

The thesis presents the study of using active stabilization method on a rotorcraft slung load system consist of a generic tiltrotor (GTR) model and an M119 Howitzer. The objective of the study was to design an ACH controller that can stabilize the load and extend the flight envelope of the rotorcraft. The preliminary control design was based on the identified load pendulum dynamics which were obtained from wind-tunnel testing and system identification conducted by researchers from Technion University and U.S. Army TDD. The study also includes analysis of the rotorcraft/load coupled system, simulation tests of the system with the integrated controller, and verifications of the controller performance. The results from the study can be concluded as the following:

1. Stability analysis of the GTR-load-coupled system shows that the coupling between the load and the tiltrotor has a destabilizing effect on the rotorcraft rigid body modes and a stabilizing effect on the load pendulum motions. This result agrees with past research of conventional rotorcraft configurations. The stability analysis also shows unstable low-frequency eigenvalues of the coupled system during low airspeeds. A SAS was designed to provide basic stability to the tiltrotor bare-airframe to eliminate the effects of having an unstable aircraft platform on the overall performance of the ACH controller. The designed SAS shows the ability to stabilize the tiltrotor rigid-body modes without a significant impact on the load pendulum modes of the coupled-system.

2. Two ACH controllers were designed that both use dynamic compensation for load stabilization. Both lead and lag designs show the ability to increase the damping of the load pendulum motions. The lag design was chosen as it provides better stability margins and higher damping for the closed-loop system. The controller design showed that a universal lag compensator could be used to increase load damping across the flight envelope. The performance of the designed controller was verified through closed-loop wind tunnel tests. The test results showed stable load behavior from zero to 200 knots for full-scale flight conditions.

3. The simulation results of the isolated load model were compared with the closed-loop wind tunnel tests with the integration of the designed ACH controller. Good agreements between the simulation results and the test results are shown by the time domain validation of the closed-loop system. The simulation model shows a good prediction of the load pendulum dynamics and provides a reasonable approximation of the sling cable force dynamics.

4. From both simulation and wind tunnel results, the ACH controller was proved to be effective in increasing load damping, minimizing LCO, and stabilizing the load pendulum motions across the flight envelope from hover through 200 knots for full-scale conditions. The simulation results also show high-frequency control inputs from the controller that resulted in small oscillations in the tiltrotor attitude angles and load cable angles. It is shown that the problem was associated with the high controller gain and can be resolved by reducing the gain value.

5.2 Future Work

For the current control design, future efforts should be put towards improving the controller for real flight applications. One important aspect is the actuator constraints. This is because when implementing the ACH device on an actual rotorcraft, the hook motion will be limited in both range and speed depending on the size of the vehicle, which may cause saturation of the actuator and degrade the performance of the controller. Therefore, studying the effects of the actuator dynamics can help to understand the trade-offs between hook limitations and load stabilization. Another important aspect is the handling qualities of the flight vehicle. Since the coupling effect from the external load is destabilizing towards the rotorcraft, the vehicle's handling qualities will suffer degradation, which might lead to the demand for extra pilot workload or failure to satisfy the requirement for specific carriage mission. This can be investigated by conducting a pilot-inthe-loop flight simulation of the coupled system on a real-time flight simulator.

Other future research activities could involve developing a more advanced controller for load stabilization. One example is to use both helicopter motion and ACH to stabilize the load, which can address the problem of hook saturation due to a small stroke limit. Another example is to design a MIMO controller which can be used to control both the helicopter and the load for applications such as payload placement or package delivery using UAV.

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