Cascaded Pumping Cycle Control for Rigid Wing Airborne Wind Energy Systems

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Airborne wind energy is an emerging technology that uses tethered unmanned aerial vehicles for harvesting wind energy at altitudes higher than conventional towered wind turbines. To make the technology competitive to other renewable energy technologies a reliable control system is required that allows autonomously operating the system throughout all phases of flight. In the present work a cascaded nonlinear control scheme for reliable pumping-cycle control of a rigid-wing airborne wind energy system is proposed. The high level control strategy in form of a state machine as well as the flight controller consisting of path-following guidance and control, attitude and rate loop is presented along with a winch controller for tether force tracking. A mathematical model for an existing prototype will be derived and results from a simulation study will be used to demonstrate the robustness of the proposed concept in presence of turbulence and wind gusts.

I. Introduction

Airborne wind energy (AWE) is an emerging branch within the sustainable energy systems portfolio that aims to exploit wind energy resources at altitudes higher than conventional towered wind turbines by means of kites and tethered aircraft. In general, AWE systems can be subdivided into two main categories. On the one hand, AWE systems with on-board generators can fly crosswind patterns with constant tether length. The kinetic energy of the relative flow is in this case directly converted into electrical power and the electricity is transmitted via a conductive tether to the ground. On the other hand, AWE systems with a ground-based generator operate in a so-called *pumping-cycle mode* and use the aerodynamic force of the kite or aircraft to uncoil the tether from a drum, which turns a generator that converts

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the mechanical torque into electrical power on the ground. When the maximum tether length is reached, the aircraft will fly back towards the ground station, while the tether is reeled in. Since the generator acts as a motor during this phase a fraction of the produced power is consumed. Once the minimum tether length is reached, the cycle starts all over again [1, 2]. For a more detailed comparison of the different concepts it is referred to [3]. In the present work the focus lies on the controller development for AWE systems operated in pumping cycle mode, although the controller can partially also be implemented for AWE systems which fly on a constant tether length.

Historically, most researchers in this field started to study the potential of flexible kite power systems, which is also reflected by the fact that most of the published papers are dedicated to the design of control systems applicable to flexible wing kite power systems [4–8]. However, due to better scaleability and efficiency the trend goes towards rigid wing AWE systems reflected by the fact that almost all companies in the field operate rigid wing prototypes. Nevertheless, available publications on rigid wing kite control are rare. Although the reliability of the control system plays a paramount role that decides upon the success of this new technology most of the available literature focuses on flight path optimization instead of the development of more robust control solutions. One recent control approach that is not dedicated to flight path optimization is presented in [9]. In the paper, the authors focus on take-off and landing control, including a transition to a loiter-like figure of eight flight pattern on a constant tether length using linear controllers.

To the best authors knowledge no modular control architecture for the full operational envelope for rigid wing AWE systems has been published yet. The term modularity is used to clearly distinguish the control approach from more integral approaches, usually based on nonlinear model predictive control such as in [10]. The present work tries to fill this gap where a modular control architecture similar to the one presented in [9], but eventually applicable to the whole range of operational modes including take-off, transition, pumping cycle mode and landing is presented. Moreover, instead of using linear control techniques a model-based nonlinear flight controller is developed that eventually increases the operational envelope and the performance of the AWE system in situations where linear control techniques might fail. In the future, the presented control approach could be augmented with adaptive control techniques to increase the robustness towards failures or unforseen environmental conditions. The modularity of the control architecture aims to achieve a high degree of reusability especially of the outer-loop module, such that it can be implemented conveniently on different platforms. The modules have defined interfaces that allow to exchange, modify and test different parts of the entire controller conveniently. This enables operators with existing prototypes to only use specific modules without the need to re-implement the entire control system. Especially the guidance module might be of interest for AWE companies, since it is entirely model independent, and can be implemented for AWE systems either operated in pumping-cycle mode or on a fixed tether length with airborne generators. Furthermore, applying systematically the concept of pseudo control hedging [11] a flight envelope protection system is implemented ensuring that no unfeasible commands are passed to the next loop. Constraining states is of particular importance in this application since the aircraft is usually operated at near stall conditions while following a three dimensional curved path which requires to constrain commands from the outer loops in a systematic manner. Such an envelope protection for airborne wind energy systems has not been presented yet apart from model predictive control approaches where constraints are directly embedded in the optimal control problem formulation [10].

The performance of the control system is demonstrated by means of a simulation study. To create a realistic simulation framework a detailed aerodynamic analysis using computational fluid dynamics (CFD) and *XFLR5* calculations of the 5 kW prototype of *Kitemill AS* have been carried out. The robustness of the control system towards wind gusts and atmospheric turbulence is assessed using three-dimensional transient wind field data generated by large-eddy simulations (LES) of a pressure-driven boundary layer.

The contributions of the present paper to the research community can be summarized as follows. First, an extension of the path-following controller which has been previously developed by one of the authors for flexible kite power systems is presented such that it can also be implemented for rigid-wing AWE systems. Furthermore, we present an intuitive way to calculate the required tangential plane course rate according to the three-dimensional path curvature to keep the aircraft on the path. Moreover, an approach for radial direction control using tether force tracking is presented and it will be demonstrated that this approach can be used at the same time for gust load alleviation. For a complete pumping cycle control we additionally propose a retraction phase controller which has not been presented for rigid wing AWE systems in the literature yet. Finally, we present a detailed description of the Kitemill 5 kW prototype, which can be used in the future as a reference model for other researchers in this field.

The paper is structured as follows. In section II the simulation models for aircraft, tether, ground station as well as the wind field are presented. In section III a detailed derivation of the different controllers is presented. Simulation results are presented in section IV followed by a conclusion in section V.

II. Reference Frames and Simulation Models

A. Reference Frames

Fig. 1 displays the wind frame W where the x_W axis is pointing in downwind direction, the z_W axis is the local earth surface normal vector, and the y_W forms a right-hand coordinate system together with x_W and z_W. The origin of the W frame is at the ground station. Note, this definition of the wind frame differs from the conventional definition found in the aerospace literature where the wind frame is a local body fixed frame [12, p. 76]. Furthermore, Fig. 2 displays the tangential plane frame τ which will be used as a reference frame for the guidance loop. The z_τ axis is pointing towards the origin of the wind frame W, the x_τ axis points towards the zenith position which is located above the ground station. Note that the τ-frame is defined equivalently to the North-East-Down frame (O) (see [13, p. 12]) for a small earth with radius one and center at the origin of the W frame. The position of the aircraft with respect to the W frame will be given either in Cartesian coordinates x_W, y_W and z_W or in spherical coordinates using longitude λ and latitude ϕ as well as the Euclidean distance of the aircraft to the origin of W. The body-fixed frame B [14, p. 57], the kinematic frame K [14, p.

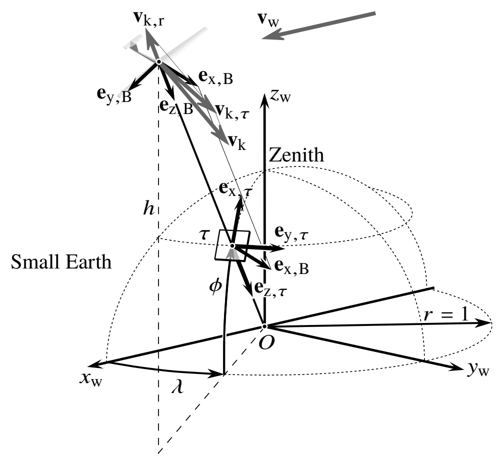

Fig. 1 Visualization of wind frame W, body-fixed frame B and tangential plane frame τ.

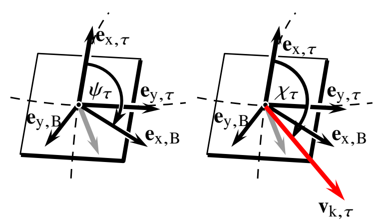

Fig. 2 Definition of the tangential plane heading Ψ_τ and tangential plane course χ_τ.

58] as well as the aerodynamic frame A [14, p. 61] are defined according to aerospace convention.

B. Tethered Aircraft Model

The control strategy in this work will be tested within a simulation environment. The aircraft simulation model represents the 5 kW prototype which has been developed by *Kitemill AS*. Relevant aircraft parameters are summarized in Table 1 and a visualization of the aircraft is shown in Fig. 3. The actuators of the aircraft are modeled as second order systems with natural frequency ω_0 and relative damping ζ, including limits on deflections and deflection rates. The

Table 1 Aircraft Parameters.

Parameters	Values	Units
Aircraft mass m_k	4.778	kg
Inertia $J_{xx,yy,zz,xz}$	1.74,0.28,1.83,-0.02	kg m^2
Wing area S_W	0.76	m^2
Wingspan b	3.7	m
Mean chord $\bar{c}$	0.22	m

numerical values are summarized in Table 2. The aircraft is modeled as a standard six degrees of freedom rigid body

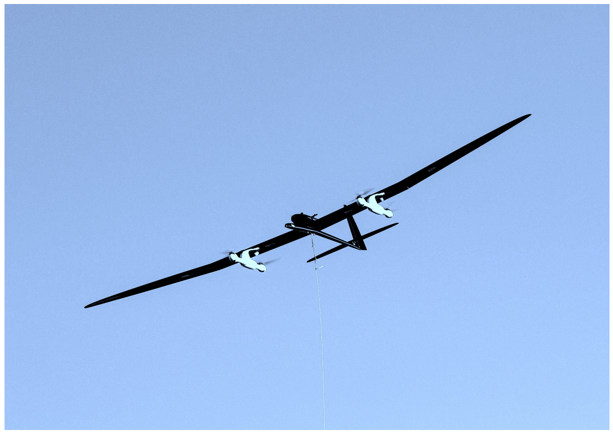

Fig. 3 5 kW prototype of Kitemill AS with vertical takeoff- and landing capabilities.

with an additional term in the translational equations of motion representing the tether force. No additional term in the rotational dynamics appears since it is assumed that the tether is attached to the center of gravity of the aircraft. A detailed derivation of the governing equations of motion can be found for instance in [12]. The translational dynamics in the bodyfixed frame B are defined as

$$(\dot{\mathbf{v}}_\mathrm{k})_\mathrm{B} = \begin{pmatrix} \dot{u}_\mathrm{k} \\ \dot{v}_\mathrm{k} \\ \dot{w}_\mathrm{k} \end{pmatrix}_\mathrm{B} = -\left(\boldsymbol{\omega}^{\mathrm{OB}}\right)_\mathrm{B} \times (\mathbf{v}_\mathrm{k})_\mathrm{B} + \frac{1}{m_\mathrm{k}}\left((\mathbf{F}_\mathrm{a})_\mathrm{B} + \left(\mathbf{F}_\mathrm{g}\right)_\mathrm{B} + (\mathbf{F}_\mathrm{t})_\mathrm{B}\right) \tag{1}$$

where $(\mathbf{v}_\mathrm{k})_\mathrm{B} \in \mathbb{R}^{3\mathrm{x}1}$ is the kinematic aircraft velocity in the B frame with components $u_\mathrm{k}, v_\mathrm{k}$ and w_k, m_k is the mass of the aircraft, $\left(\boldsymbol{\omega}^{\mathrm{OB}}\right)_\mathrm{B} \in \mathbb{R}^{3\mathrm{x}1}$ is the angular velocity vector between the B and O frame containing the roll rate p, pitch rate q as well as yaw rate r, $(\mathbf{F}_\mathrm{a})_\mathrm{B} \in \mathbb{R}^{3\mathrm{x}1}$ is the aerodynamic force, $\left(\mathbf{F}_\mathrm{g}\right)_\mathrm{B} \in \mathbb{R}^{3\mathrm{x}1}$ is the gravity force and $(\mathbf{F}_\mathrm{t})_\mathrm{B} \in \mathbb{R}^{3\mathrm{x}1}$ is the

Table 2 Actuator Parameters.

Parameters	Values	Units
Natural frequency ω_0	35	rad s^{-1}
Relative damping ζ	1	-
Max./Min. aileron deflection	±15	°
Max./Min. elevator deflection	±15	°
Max./Min. rudder deflection	± 20	°
Rate limits (all actuators)	±300	° s^{-1}

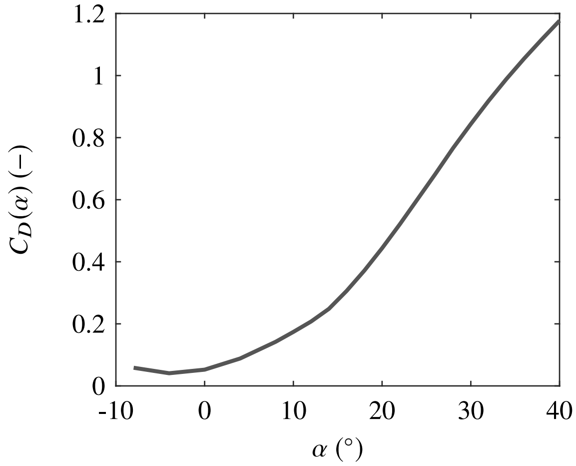

Fig. 4 Drag coefficient as a function of angle of attack.

tether force. All forces are defined with respect to the center of gravity. The aerodynamic force is modeled according to

$$(\mathbf{F}_\mathrm{a})_\mathrm{B} = \frac{1}{2}\rho V_\mathrm{a}^2 S_\mathrm{W} \mathbf{M}_\mathrm{BA} \begin{pmatrix} -C_\mathrm{D} \\ C_\mathrm{Y} \\ -C_\mathrm{L} \end{pmatrix}_\mathrm{A} \tag{2}$$

where $\rho = 1.225\,\mathrm{kg\,m^{-3}}$ is the air density and $\mathbf{M}_\mathrm{BA}$ is the transformation matrix from the aerodynamic frame A to the bodyfixed frame B [12, p. 77]. The coefficients $C_\mathrm{D}, C_\mathrm{Y}$, and C_L are nonlinear functions of the aircraft states and surface deflections. For the purpose of this paper CFD and XFLR5 was used to create lookup tables that capture the main dependencies of the coefficients on states and surface deflections. The modeled dependencies on angle of attack, sidelip angle and the control surface deflections are displayed in Fig. 4-6. Note, the contributions of the surface deflections to the drag coefficient where negligible and are therefore not displayed. Additionally, damping coefficients (see Table 3) are added which in total yields

$$\begin{aligned} C_\mathrm{D} &= C_\mathrm{D}(\alpha) \\ C_\mathrm{Y} &= C_\mathrm{Y}(\beta, \delta_\mathrm{r}) + C_\mathrm{Yp}\frac{pb}{2V_\mathrm{a}} + C_\mathrm{Yr}\frac{rb}{2V_\mathrm{a}} \\ C_\mathrm{L} &= C_\mathrm{L}(\alpha, \delta_\mathrm{e}) + C_\mathrm{Lq}\frac{q\bar{c}}{2V_\mathrm{a}} \end{aligned} \tag{3}$$

where $C_\mathrm{Yp}, C_\mathrm{Yr}$ and C_Lq are defined in Table 3.

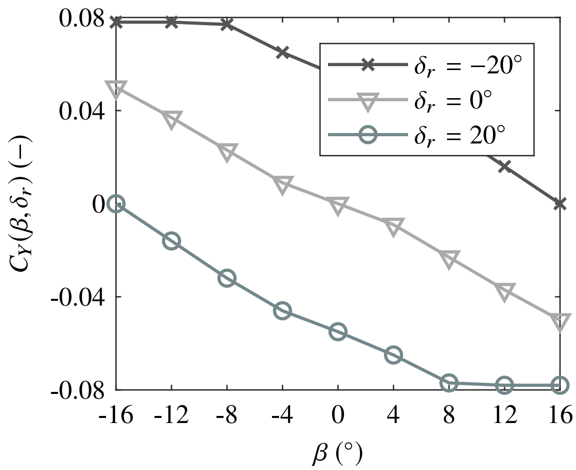

Fig. 5 Side force coefficient as a function of sideslip angle and and rudder deflection.

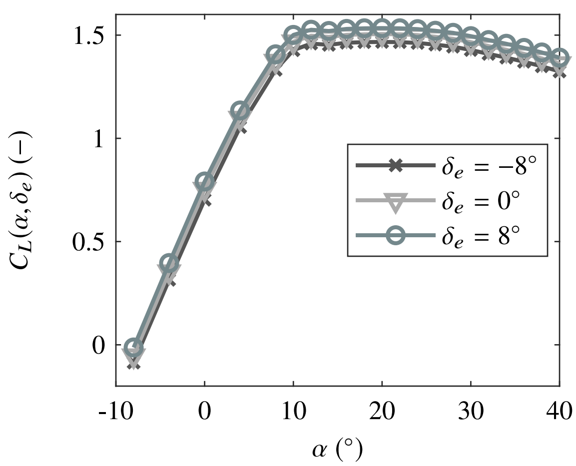

Fig. 6 Lift coefficient as a function of angle of attack and elevator deflection.

The rotational dynamics are defined as

$$\begin{aligned} \left(\dot{\omega}^{\mathrm{OB}}\right)_{\mathrm{B}} &= \begin{pmatrix} \dot{p} \\ \dot{q} \\ \dot{r} \end{pmatrix}_{\mathrm{B}} \\ &= \mathbf{J}^{-1}\left(-\left(\boldsymbol{\omega}^{\mathrm{OB}}\right)_{\mathrm{B}} \times \mathbf{J}\left(\boldsymbol{\omega}^{\mathrm{OB}}\right)_{\mathrm{B}} + (\mathbf{M}_{\mathrm{a}})_{\mathrm{B}}\right) \end{aligned} \tag{4}$$

Table 3 Rate dependencies of the force coefficients.

Coefficients	Values
C_{Yp}	-0.133
C_{Yr}	0.172
C_{Lq}	7.267

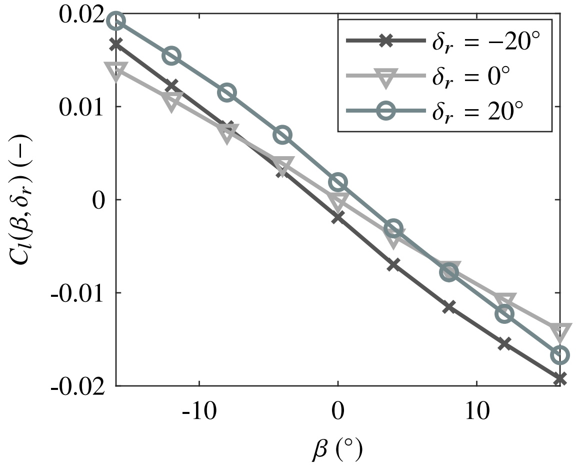

Fig. 7 Rollmoment coefficient as a function of sideslip angle and rudder deflection.

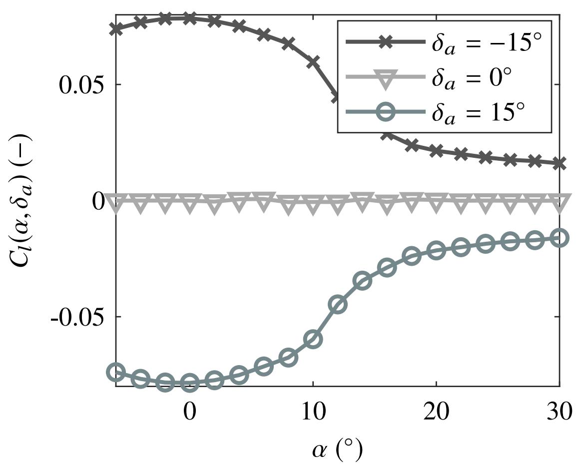

Fig. 8 Roll moment coefficient as a function of angle of attack and aileron deflection.

where $\mathbf{J} \in \mathbb{R}^{3\text{x}3}$ is the inertia tensor, and $(\mathbf{M}_\text{a})_\text{B} \in \mathbb{R}^{3\text{x}1}$ is the resulting aerodynamic moment around the center of gravity of the aircraft. Similar to the aerodynamic force the aerodynamic moment is defined using moment coefficients:

$$(\mathbf{M}_\text{a})_\text{B} = \frac{1}{2}\rho V_\text{a}^2 S_\text{W} \begin{pmatrix} bC_\text{l} \\ \bar{c}C_\text{m} \\ bC_\text{n} \end{pmatrix} \tag{5}$$

The relevant dependencies of the moment coefficients on states and surface deflections are depicted in Fig. 7-11. The damping terms are summarized in Table 4, which in total yields for the moment coefficients

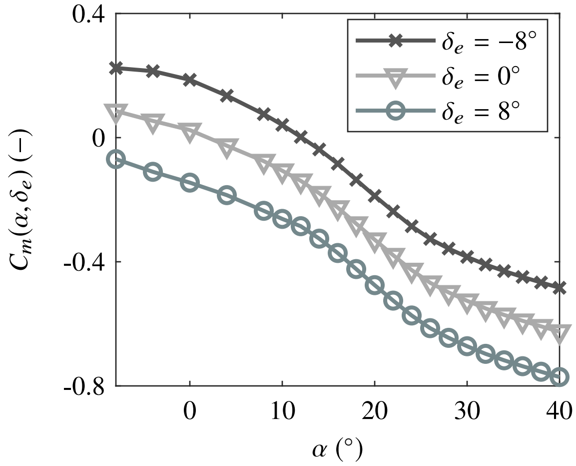

Fig. 9 Pitch moment coefficient as a function of angle of attack and elevator deflection.

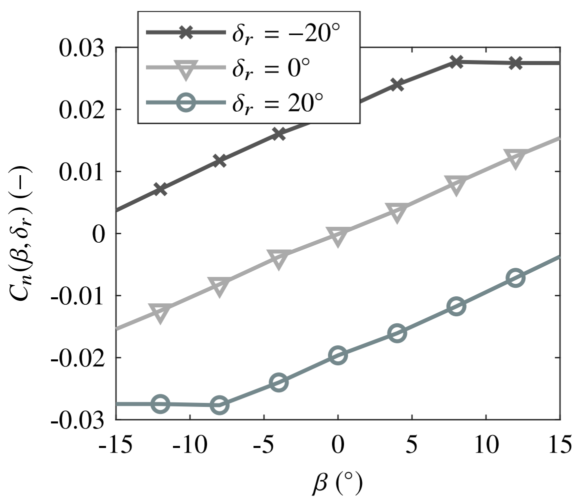

Fig. 10 Yaw moment coefficient as a function of sideslip angle and rudder deflection.

$$
\begin{aligned}
C_{\mathrm{l}} &= C_{\mathrm{l}}(\alpha, \delta_{\mathrm{a}}) + C_{\mathrm{l}}(\beta, \delta_{\mathrm{r}}) + C_{\mathrm{lp}} \frac{pb}{2V_{\mathrm{a}}} + C_{\mathrm{lr}} \frac{rb}{2V_{\mathrm{a}}} \\
C_{\mathrm{m}} &= C_{\mathrm{m}}(\alpha, \delta_{\mathrm{e}}) + C_{\mathrm{mq}} \frac{q\bar{c}}{2V_{\mathrm{a}}} \\
C_{\mathrm{n}} &= C_{\mathrm{n}}(\alpha, \delta_{\mathrm{a}}) + C_{\mathrm{n}}(\beta, \delta_{\mathrm{r}}) + C_{\mathrm{np}} \frac{pb}{2V_{\mathrm{a}}} + C_{\mathrm{nr}} \frac{rb}{2V_{\mathrm{a}}}
\end{aligned} \tag{6}
$$

Table 4 Damping coefficients.

Coefficients	Values
C_{lp}	-0.6450
C_{lr}	0.2190
C_{mq}	-16.3740
C_{np}	-0.1310
C_{nr}	-0.0335

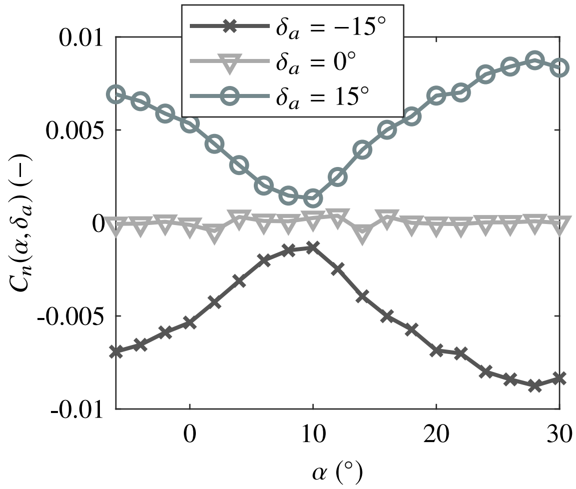

Fig. 11 Yaw moment coefficient as a function of angle of attack and aileron deflection.

The attitude is parameterized using quaternions, hence the equation for the attitude propagation is given by

$$\dot{\mathbf{q}} = \begin{pmatrix} \dot{q}_1 \\ \dot{q}_2 \\ \dot{q}_3 \\ \dot{q}_4 \end{pmatrix} = \begin{pmatrix} -q_2 & -q_3 & -q_4 & q_1 \\ q_1 & -q_4 & q_3 & q_2 \\ q_4 & q_1 & -q_2 & q_3 \\ -q_3 & q_2 & q_1 & q_4 \end{pmatrix} \begin{pmatrix} p \\ q \\ r \\ 2k\kappa \end{pmatrix} \tag{7}$$

The quaternion attitude propagation equation Eq. (7) is implemented with gradient feedback as described in [15, p.64] with $\kappa = 1 - q_1^2 - q_2^2 - q_3^2 - q_4^2$ otherwise numerical inaccuracies can lead to a violation of the unity norm condition of the quaternion vector. The position of the aircraft's center of gravity $\left(\mathbf{p}^{\mathrm{G}}\right)_{\mathrm{O}}$ in the O frame will be propagated according to

$$\left(\dot{\mathbf{p}}^{\mathrm{G}}\right)_{\mathrm{O}} = \begin{pmatrix} \dot{p}_{\mathrm{x}}^{\mathrm{G}} \\ \dot{p}_{\mathrm{y}}^{\mathrm{G}} \\ \dot{p}_{\mathrm{z}}^{\mathrm{G}} \end{pmatrix}_{\mathrm{O}} = \mathbf{M}_{\mathrm{OB}} \begin{pmatrix} u_{\mathrm{k}} \\ v_{\mathrm{k}} \\ w_{\mathrm{k}} \end{pmatrix}_{\mathrm{B}} \tag{8}$$

where $\mathbf{M}_{\mathrm{OB}}$ is the transformation matrix from the B to the O frame (see [12, p. 12]).

The states of the tethered aircraft are the three kinematic velocity components in the bodyfixed frame $u_{\mathrm{k}}, v_{\mathrm{k}}$ and w_{k}, the body rates p, q, r, the quaternions q_1, q_2, q_3 and q_4, as well as the position in the O frame with components $p_{\mathrm{x}}^{\mathrm{G}}, p_{\mathrm{y}}^{\mathrm{G}}$ and $p_{\mathrm{z}}^{\mathrm{G}}$. At the moment full state feedback is assumed, and the controller requires measurements for mean wind direction on the ground ξ, position, velocity, orientation, angle of attack α, sideslip angle β, airspeed V_{a}, rotational rates as well as the total tether force F_{t} measured on the ground and at the aircraft. The reason for measuring the tether force on the aircraft as well as on the ground is that due to the tether drag and weight the force measured on the ground differs from the tether force acting on the aircraft.

C. Tether Model

The tether is modeled as a particle system where the individual particles are connected via spring-damper elements. For each particle the point mass dynamics are formulated incorporating tether drag and tether weight. During reel-out or reel-in the unstretched length of each spring-damper as well as the mass of each particle is adapted proportionally to the current change in tether length. A detailed explanation of the implemented tether model can be found in a previous work of the second author [16].

D. Ground Station

In general, the ground station consists of the generator and the winch. In this work the only relevant component for the controller development is represented by the winch which can be modeled as a scalar first order system given by

$$\dot{\omega}_{\mathrm{w}} = J_{\mathrm{w}}^{-1} \left(-\kappa_{\mathrm{w}}\omega_{\mathrm{w}} + r_{\mathrm{w}}F_{\mathrm{t}} + M_{c}\right) \tag{9}$$

where ω_{W} represents the rotational speed of the winch, r_{W} is the radius of the winch, which is assumed to be constant despite the reeling-in or -out of the tether, $\kappa_{\mathrm{W}} > \forall t$ is a viscous friction coefficient, F_{t} is the tether force and M_{c} is the motor/generator torque which represents the control input. The electrical drive system of the ground station is not modeled in this work. The utilized values for the winch are summarized in Table 5.

E. Wind Field Model

In order to test the controller in a realistic wind field, a four-dimensional velocity field is integrated into the simulation framework. The wind field data was generated by means of large-eddy simulations of a pressure-driven boundary layer. The computations were carried out using *SPWind*, a pseudo-spectral simulation code developed at KU Leuven. Information on the specification and the implementation of the flow solver can be found in [17–19]. The wind field data is available at a spatial resolution of approximately 20 m × 15 m × 7 m in x_{W}, y_{W} and z_{W} direction, respectively, for a time series of several minutes and stored in form of lookup tables. During the simulation the wind velocity vector at the

Table 5 Winch Parameters.

Parameters	Values	Units
Winch radius r_{W}	0.1	m
Inertia J_{W}	0.08	kg m^2
Viscous friction κ_{W}	0.6	kg m s^{-1}

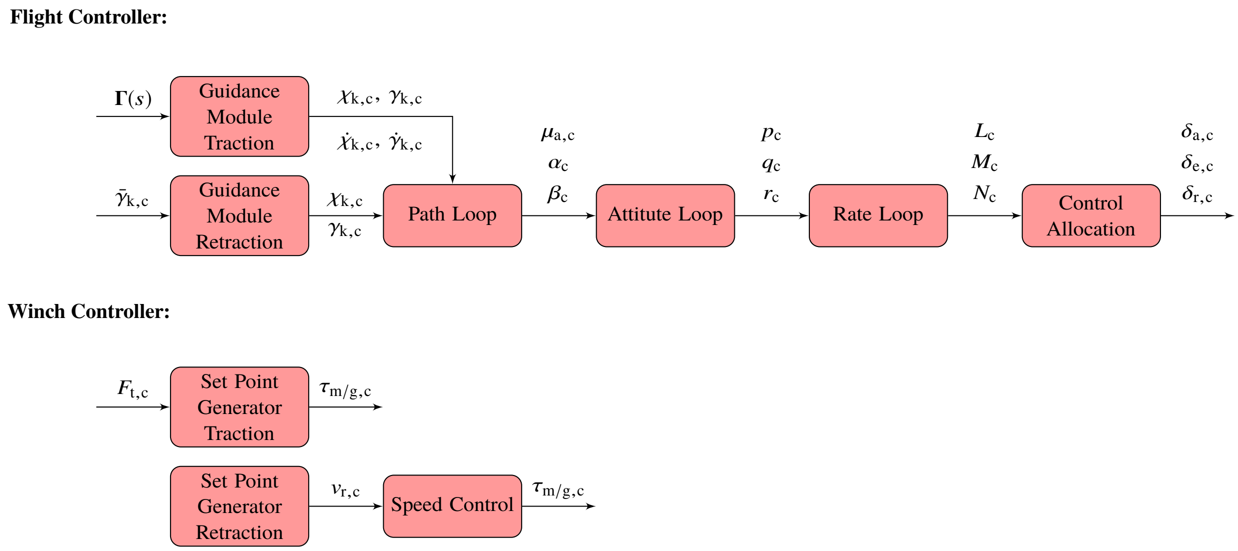

Fig. 12 Cascaded control structure of flight and winch control system for traction and retraction mode.

location of the aircraft is obtained through linear interpolation of the adjacent vertex velocity vectors.

III. Controller Development

A. Control Architecture and State Machine

The high level control architecture is displayed in Fig. 12. On the highest level the controller can be decomposed into the flight and the winch control system, represented by the upper and lower cascade in Fig. 12. The task of the flight control system is to control the tangential motion on the sphere while the radial direction is controlled by the winch. The blocks correspond to modules that will be discussed in more detail in the following sections. In general, each block has one input and one output signal corresponding to the set point that has to be tracked by the module as well as the commanded set point for the next module. Blocks with two inputs are subdivided into two submodules (not displayed), one module for the traction and one for the retraction phase. All remaining modules are the same for both traction and retraction, although different gains and filter bandwidths are used for increased performance. Based on the current state π_i, as defined in Table 6, the output from either the traction or retraction module is passed on to the next module. The flight control guidance module input of the traction phase is the path parameterization $\mathbf{\Gamma}(s) \in \mathbb{R}^{3x1}$ with $s \in (0, 2\pi)$. Within this module the required kinematic (subscript k) course $\chi_{k,c}$ and kinematic path angle $\gamma_{k,c}$ as well as the required course rate $\dot{\chi}_{k,c}$ and path angle rate $\dot{\gamma}_{k,c}$ are calculated based on the current position. The guidance module input of the retraction phase is the desired path angle $\bar{\gamma}_{k,c}$ and the output signal is the kinematic course $\chi_{k,c}$ and kinematic path angle $\gamma_{k,c}$. Note, $\bar{\gamma}_{k,c}$ and $\gamma_{k,c}$ differ from each other only in the final part of the retraction phase where the path angle $\bar{\gamma}_{k,c}$ is linearly increased to a fixed value before the transition back into the traction phase is triggered. This maneuver is used to dissipate kinetic energy of the aircraft before the turn. The path loop will track the commands from the guidance

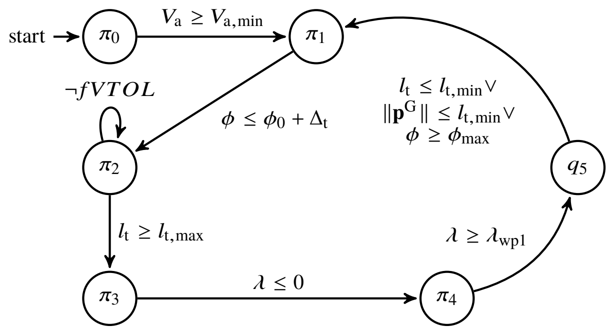

Fig. 13 State-machine for the pumping cycle mode.

module and calculates attitude commands for aerodynamic (subscript a) bank angle $\mu_{a,c}$ and angle of attack α_c. Note, α and β always refer to the aerodynamic and not kinematic angles if not indicated otherwise. The attitude loop tracks the attitude commands and transforms them into roll-, pitch-, and yaw rate commands p_c, q_c and r_c, respectively. Finally, the rate loop calculates the control moments which are then distributed among the actuators in the control allocation block, which results in an aileron command δ_a, elevator command δ_e and rudder command δ_r. The winch controller requires only a set point generator for traction and retraction phase as well as a speed controller. During the traction phase, a reference torque $\tau_{m/g,c}$ is directly calculated based on the tether force set point $F_{t,c}$. During the retraction phase a fixed reeling in speed $v_{r,c}$ is commanded that will be tracked by a speed controller, which outputs a corresponding torque command $\tau_{m/g,c}$. In both cases, the torque commands will be tracked by the electrical drive control system.

Fig. 13 shows the state-machine that is used to switch between the different control modules. The individual states are defined in Table 6. The modeled prototype of *Kitemill AS* allows vertical takeoff and landing (VTOL). A VTOL controller including the transition into pumping cycle mode is implemented in the simulation framework, however a detailed description of the VTOL controller is out off the scope of this paper and will be part of a future publication. Essentially, a similar control approach for the winch and the flight controller is adapted from [20], where a VTOL controller for a flexible kite system is presented. The interface to the pumping cycle mode is given by a transition into π_0. In this work it will be assumed that the aircraft was guided in downwind direction to the operational altitude that

Table 6 State definitions.

State	Description
π_0	Transition from take-off to aircraft mode.
π_1	Capture crosswind pattern.
π_2	Traction phase.
π_3	Intermediate state.
π_4	Transition to retraction.
π_5	Retraction phase.

fulfills the latitude condition $\phi > \phi_0 + \Delta_{\phi_0}$, where the VTOL controller keeps the aircraft in a hover state (not displayed) until π_0 is triggered. ϕ_0 is the mean latitude angle of the path and $0° \leq \Delta_{\phi_0} \leq 10°$ is a small offset. The transition from the launching state to the crosswind flight state is initiated by fast reeling in of the tether. As soon as the airspeed exceeds the minimum airspeed, here denoted with $V_{\mathrm{a,min}}$, the transition to π_1 is triggered. In this state the path-following controller is activated and the guidance law is initialized with a first guess of the closest point on the path relative to the current aircraft position. Flying towards the path decreases the elevation angle, which triggers the transition into the traction phase state π_2 if it reaches a value below $\phi_\mathrm{m} + \Delta_\mathrm{t}$ ($0° \leq \Delta_\mathrm{t} \leq 10°$) and the winch starts reeling out the tether. The intermediate state π_1 was added to start reeling out after the aircraft is sufficiently steered into the downwind direction. If the tether is reeled out immediately this could lead to a drop in tether tension during the initial turn. As long as no landing is issued by the supervisory layer ($fVTOL = 1$) the kite remains in state π_2. The transition into π_3 is triggered as soon as the specified tether length is reached. This state can be interpreted as an intermediate state which is left as soon as the aircraft flies into the negative half plane of the wind window defined by a negative longitude $\lambda < 0$. This triggers the transition to π_4. The retraction phase is initiated as soon as the aircraft flies past wp_1 which is defined as the outermost point on the path. This procedure ensures that before the reeling-in of the tether is triggered the aircraft always has to fly downwards through the center and flies towards the ground station on the same side of the wind window. Before the aircraft transitions back into the traction mode, one out of three conditions has to be satisfied: Either the tether length or the Euclidean distance $\|\mathbf{p}^\mathrm{G}\|$ of the aircraft relatively to the ground station is below a specified value, or the elevation angle of the aircraft exceeds a maximum value. The latter can be regarded as a safety mechanism that prevents the aircraft from overshooting the ground station.

B. Guidance Modules

In the existing AWE literature [6, 7, 21, 22] the kite is steered according to the tangential plane course set point $\chi_{\tau,\mathrm{c}}$. It is defined as the angle between the $\mathbf{e}_{\mathrm{x},\tau}$ axis of the tangential plane frame τ and the x-axis of the kinematic frame K as depicted in Fig. 2. This strategy is mainly motivated by the fact that a direct relationship between the steering input of a flexible kite and the tangential plane course rate can be derived [8, 23] which allows to directly calculate the steering input based on the course rate. In this work the guidance problem will be solved as well by first calculating the desired χ_τ course set point, which will be then however transformed into a corresponding set point for the course χ_k and path angle γ_k, which specify the orientation of the K frame relatively to the O frame. This approach provides an additional control degree of freedom to track the desired flight direction. Moreover, controlling course and path angle in the traction phase allows to use the same medium loop control structure for the retraction phase in which the kite is not steered on a tangential plane anymore. Furthermore, providing set points for course and path angle allows to integrate the guidance module easier into existing autopilot architectures for conventional aircraft. Hence this approach also fits better into the modular control philosophy proposed in this work.

1. Traction Phase Guidance

Separating the radial and the tangential motion of the aircraft the control objectives for the traction phase can be stated as follows: On the one hand, the radial direction needs to be controlled by the winch such that the tether force set point is tracked. Moreover, the radial direction controller needs to ensure that the maximum tether tension is not exceeded to avoid tether rupture or aircraft damage. On the other hand, for the tangential motion control the aircraft position will be projected onto the unit sphere. In that case, the flight controller needs to follow a predefined flight path on a sphere with a constant radius of one. The path on the unit sphere is adapted proportionally to the distance of the aircraft to the ground station such that the real path the aircraft traces has a constant shape during the reel-out phase. Fig. 14 depicts an example flight path, including a visualization of the aircraft and the flexible tether. Note that the depicted vectors and the aircraft model are scaled, and the physical flight path and not the path on the unit sphere that is used for the guidance is shown for visualization purposes.

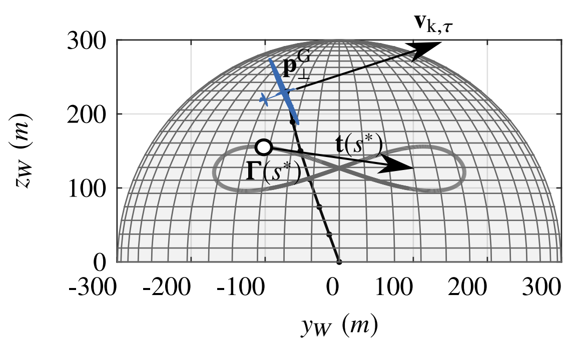

Fig. 14 Reference flight path on a sphere.

Parts of the guidance module are based on a previous work of the second author [8] where it is used to steer a flexible kite along a prescribed path. In this work some modifications are introduced such as a novel predictive part that takes the instantaneous path curvature into account in order to calculate the reference course rate. Furthermore, the interface to a rigid-wing aircraft path-following controller will be presented taking into account a generalization of the rotational rate vector $\left(\omega^{\tau\bar{K}}\right)_{\bar{K}}$ which describes the relative rotation between the rotated kinematic and the tangential plane frame. Since the terminology slightly deviates from [8] the main steps of the derivation will be presented again in addition to the novel extensions for completeness.

The objective of the guidance module is twofold. First, it needs to calculate the flight direction that leads to a reduction of the distance δ (i.e. the cross track error) as defined by the arc length between the projected aircraft position on the unit sphere $\mathbf{p}_{\perp}^{G}$ and the path $\boldsymbol{\Gamma}$. Second, for zero cross-track error the kinematic velocity vector projected onto the tangential plane $\mathbf{v}_{k,\tau}$ needs to be aligned with the path direction as defined by the tangent vector $\mathbf{t}$. For clarification, all relevant vectors are depicted in Fig. 14. The path is defined in spherical coordinates on the unit sphere, hence a point on the path is fully defined by its longitude λ_Γ and latitude ϕ_Γ. Note, all vectors are given in the W reference frame, if not indicated otherwise. In Cartesian coordinates the path is given as

$$\boldsymbol{\Gamma}(s) = \begin{pmatrix} \cos\lambda_\Gamma(s)\cos\phi_\Gamma(s) \\ \sin\lambda_\Gamma(s)\cos\phi_\Gamma(s) \\ \sin\phi_\Gamma(s) \end{pmatrix} \tag{10}$$

For subsequent calculations the tangent and its derivative need to be known. The tangent can be calculated according to

$$\mathbf{t}(s) = \frac{d\boldsymbol{\Gamma}}{ds} = \frac{\partial\boldsymbol{\Gamma}}{\partial\lambda_\Gamma}\frac{d\lambda_\Gamma}{ds} + \frac{\partial\boldsymbol{\Gamma}}{\partial\phi_\Gamma}\frac{d\phi_\Gamma}{ds} \tag{11}$$

and its derivative is given by

$$\begin{aligned}\mathbf{t}'(s) = &\frac{\partial^2\boldsymbol{\Gamma}}{\partial\lambda_\Gamma^2}\left(\frac{d\lambda_\Gamma}{ds}\right)^2 + \\ &2\frac{\partial^2\boldsymbol{\Gamma}}{\partial\phi_\Gamma\partial\lambda_\Gamma}\frac{d\phi_\Gamma}{ds}\frac{d\lambda_\Gamma}{ds} + \frac{\partial^2\boldsymbol{\Gamma}}{\partial\phi_\Gamma^2}\left(\frac{d\phi_\Gamma}{ds}\right)^2 + \frac{\partial\mathbf{t}}{\partial s}\end{aligned} \tag{12}$$

The last partial derivative is given by

$$\frac{\partial\mathbf{t}}{\partial s} = \frac{\partial\boldsymbol{\Gamma}}{\partial\lambda_\Gamma}\frac{d^2\lambda_\Gamma}{ds^2} + \frac{\partial\boldsymbol{\Gamma}}{\partial\phi_\Gamma}\frac{d^2\phi_\Gamma}{ds^2} \tag{13}$$

Furthermore, the speed of the path parameter s is denoted with $ds/dt = \dot{s}$ and is given by the projection of the velocity vector onto the path tangent:

$$\frac{ds}{dt} = \dot{s} = \frac{\mathbf{t}^\top \left(\mathbf{v}_\mathrm{k}^\mathrm{G}\right)_\mathrm{W}}{\|\mathbf{t}\|_2 \| \left(\mathbf{p}^\mathrm{G}\right)_\mathrm{W} \|_2} \tag{14}$$

The flight path can be fully described as a planar curve using scalar functions of s for longitude and latitude. The flight path in this work will be defined as a *Lemniscate of Booth*, given by

$$\begin{aligned}\lambda_\Gamma(s) &= \frac{a_\mathrm{Booth}\sin s}{1 + \left(\frac{a_\mathrm{Booth}}{b_\mathrm{Booth}}\right)^2\cos^2 s} \\ \phi_\Gamma(s) &= \frac{\frac{a_\mathrm{Booth}^2}{b_\mathrm{Booth}}\sin s\cos s}{1 + \left(\frac{a_\mathrm{Booth}}{b_\mathrm{Booth}}\right)^2\cos^2 s}\end{aligned} \tag{15}$$

which can be derived from the equation of a hyperbolic lemniscate as defined for instance in [24, p.164] with $y = x\frac{a}{b}\cos s$. a_Booth and b_Booth are parameters that define height and width of the curve. A detailed comparison with other curve parameterzations is out of the scope of this paper. Note, however, that the Lemniscate of Booth offers for a large range of width and height parameters smaller curvature peaks compared to the *Lissajous* figure parameterization utilized in

[8] which is why it is chosen in this work. Ultimately, the planar curve can be transformed into a three dimensional curve using Eq. (10).

The distance between a point on the curve and the kite position can be calculated using the definition of the arc length.

$$\delta(s) = \arccos\left(\mathbf{p}_\perp^{\mathrm{G}} \cdot \mathbf{\Gamma}(s)\right) \tag{16}$$

In order to determine the closest point (defined by s^*) requires to solve

$$\left.\frac{d\delta}{ds}\right|_{\mathrm{s=s^*}} = 0 \tag{17}$$

where the derivative is given by

$$\frac{d\delta}{ds} = -\frac{1}{\sin\delta}\frac{d\left(\mathbf{p}_\perp^{\mathrm{G}} \cdot \mathbf{\Gamma}(s)\right)}{ds} = -\frac{\mathbf{p}_\perp^{\mathrm{G}} \cdot \mathbf{t}(s)}{\sin\delta} \tag{18}$$

Eventually, the following root-finding problem needs to be solved:

$$\mathbf{p}_\perp^{\mathrm{G}} \cdot \mathbf{t}(s) = 0 \tag{19}$$

the solution can be determined using for instance *Newton*'s method. With

$$\left(\frac{d}{ds}\right)\mathbf{p}_\perp^{\mathrm{G}} \cdot \mathbf{t}(s) = \mathbf{p}_\perp^{\mathrm{G}} \cdot \frac{d\mathbf{t}(s)}{ds} \tag{20}$$

The update equation for *Newton*'s method is then

$$s^+ = s^- - \frac{\mathbf{p}_\perp^{\mathrm{G}} \cdot \mathbf{t}(s)}{\mathbf{p}_\perp^{\mathrm{G}} \cdot \mathbf{t}'(s)} \tag{21}$$

In the simulations, the method converged usually quickly after two to three iterations if the previous solution is selected as a starting point.

Knowing the closest point on the curve relative to the current aircraft position enables to calculate the desired flight direction. The vector at the current aircraft position pointing towards $\mathbf{\Gamma}(s^*)$ perpendicularly along a great circle can be expressed as

$$\mathbf{b}^{\mathrm{G}} = \frac{\mathbf{\Gamma}(s^*) - \cos\delta\mathbf{p}_\perp^{\mathrm{G}}}{\sin\delta} \tag{22}$$

This can be derived simply by looking at the normal projection of $\mathbf{\Gamma}(s^*)$ onto $\mathbf{p}_\perp^{\mathrm{G}}$ (see Fig. 15) given by

$$\mathbf{\Gamma}_{\mathrm{proj}}(s^*) = \cos\delta\mathbf{p}_\perp^{\mathrm{G}} \tag{23}$$

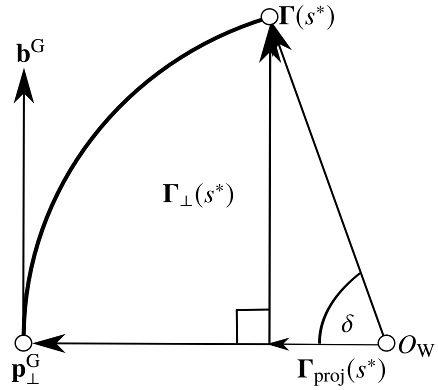

Fig. 15 A slice of the unit sphere containing a segment of the great circle that connects $\mathbf{p}_{\perp}^{\mathrm{G}}$ with $\Gamma(s^*)$.

and

$$\Gamma_{\perp}(s^*) = \Gamma(s^*) - \Gamma_{\mathrm{proj}}(s^*) \tag{24}$$

where $-\Gamma_{\perp}(s^*)$ denotes the vector of the projection direction, which is by definition perpendicular to $\mathbf{p}_{\perp}^{\mathrm{G}}$. Normalizing $\Gamma_{\perp}(s^*)$ yields:

$$\mathbf{b}^{\mathrm{G}} = \frac{\Gamma(s^*) - \Gamma_{\mathrm{proj}}(s^*)}{\|\Gamma(s^*) - \Gamma_{\mathrm{proj}}(s^*)\|_2} = \frac{\Gamma(s^*) - \cos\delta\,\mathbf{p}_{\perp}^{\mathrm{G}}}{\sin\delta} \tag{25}$$

Equation (19) can be rewritten using Eq. (25):

$$\frac{\Gamma(s^*)\cdot\mathbf{t}(s^*) - \sin\delta\left(\mathbf{b}^{\mathrm{G}}\cdot\mathbf{t}(s^*)\right)}{\cos\delta} = 0 \tag{26}$$

The first scalar product is zero, since $\Gamma(s^*)$ is perpendicular to the tangent vector, which yields

$$\tan\delta\left(\mathbf{b}^{\mathrm{G}}\cdot\mathbf{t}(s^*)\right) = 0 \tag{27}$$

If this equation is divided by $\tan\delta$ and bearing in mind that the only relevant singularity is located at $\delta = 0$ this yields for $\delta \neq 0$

$$\mathbf{b}^{\mathrm{G}}\cdot\mathbf{t}(s^*) = 0 \tag{28}$$

which proves that the direction vector pointing towards the path is indeed orthogonal to the tangent at $\Gamma(s^*)$. Hence, if the kite would fly in $\mathbf{b}^{\mathrm{G}}$ direction it would intercept the path perpendicularly. From a practical point of view it is however not desired that the aircraft intercepts the path perpendicularly. Instead, it is desirable that the commanded flight direction smoothly transitions from an orthogonal interception if the aircraft is farther away from the curve to a tangential, hence curve aligned, flight direction. If the aircraft is on the path it is desired that the path controller tracks the directional angle of the tangent vector on the curve. This behavior can be achieved as follows: If $\delta \neq 0$ the course angle $\chi_{\tau,\|}$, which can be obtained from the tangent on the path, has to be adapted such that the distance to the curve δ decreases over time. In [8] the following set point definition is proposed, which is utilized in this work as well:

$$\chi_{\tau,\mathrm{c}} = \chi_{\tau,\|} + \Delta\chi_{\tau} \tag{29}$$

with

$$\Delta\chi_{\tau} = \arctan_2\left(-\sigma(\iota)\delta, \delta_0\right) \tag{30}$$

and

$$\iota = (\mathbf{t}(s^*) \times \mathbf{\Gamma}(s^*)) \cdot \left(\mathbf{p}_{\perp}^{\mathrm{G}} - \mathbf{\Gamma}(s^*)\right) \tag{31}$$

where σ denotes the sign of ι. Depending if the aircraft is on the left or right hand side of the path, as depicted in Fig. 16, the sign of $\Delta\chi_{\tau}$ is adapted accordingly. If the course as defined in Eq. (29) is tracked by the flight control system, the relative distance δ between aircraft and path decreases over time, i.e. $\dot{\delta} < 0$ with can be proven as follows. Taking the derivative of Eq. (16) with respect to time at $s = s^*$ yields

$$\dot{\delta} = -\frac{1}{\sqrt{1-\cos^2\delta}}\left(\dot{\mathbf{p}}_{\perp}^{\mathrm{G}} \cdot \mathbf{\Gamma}(s^*) + \mathbf{p}_{\perp}^{\mathrm{G}} \cdot \dot{\mathbf{\Gamma}}(s^*)\right) \tag{32}$$

with

$$\dot{\mathbf{\Gamma}}(s^*) = \mathbf{t}(s^*)\dot{s}^* \tag{33}$$

$\mathbf{p}_{\perp}^{\mathrm{G}} \cdot \mathbf{t}(s^*)$ is zero, therefore,

$$\dot{\delta} = -\frac{1}{\sin\delta}\left(\dot{\mathbf{p}}_{\perp}^{\mathrm{G}} \cdot \mathbf{\Gamma}(s^*)\right) \tag{34}$$

With Eq. (25) the dot product can be written as

$$\dot{\mathbf{p}}_{\perp}^{\mathrm{G}} \cdot \mathbf{\Gamma}(s^*) = \dot{\mathbf{p}}_{\perp}^{\mathrm{G}} \cdot \mathbf{b}^{\mathrm{G}} \sin\delta + \dot{\mathbf{p}}_{\perp}^{\mathrm{G}} \cdot \mathbf{p}_{\perp}^{\mathrm{G}} \cos\delta \tag{35}$$

Per definition, the second scalar product on the right hand side is zero. Inserting the result into Eq. (34) yields

$$\dot{\delta} = -\dot{\mathbf{p}}_{\perp}^{\mathrm{G}} \cdot \mathbf{b}^{\mathrm{G}} \tag{36}$$

This can be further simplified to

$$\dot{\delta} = -v_{\mathrm{k},\tau}\ \cos\theta \tag{37}$$

where $v_{\mathrm{k},\tau}$ is the magnitude of $\mathbf{v}_{\mathrm{k},\tau} = \dot{\mathbf{p}}_{\perp}^{\mathrm{G}}$ and θ denotes the angle between the vector pointing perpendicularly to $\Gamma(s^*)$

and the projected aircraft velocity on the tangential plane. To calculate θ two cases have to be distinguished:

$$\theta = \begin{cases} \pi/2 - \Delta\chi_\tau + e_{\chi_\tau}, & \text{for } \sigma < 0 \\ \pi/2 + \Delta\chi_\tau - e_{\chi_\tau}, & \text{for } \sigma > 0 \end{cases} \tag{38}$$

This yields for $\dot{\delta}$

$$\dot{\delta} = \begin{cases} -v_{\mathrm{k},\tau} \sin\left(\Delta\chi_\tau - e_{\chi_\tau}\right), & \text{for } \sigma < 0 \\ -v_{\mathrm{k},\tau} \sin\left(-\Delta\chi_\tau + e_{\chi_\tau}\right), & \text{for } \sigma > 0 \end{cases} \tag{39}$$

with Eq. (30) it follows

$$\begin{aligned} \dot{\delta} &= -\sigma v_{\mathrm{k},\tau} \sin\left(-\Delta\chi_\tau + e_{\chi_\tau}\right) \\ &= -\sigma v_{\mathrm{k},\tau} \left(-\sin\Delta\chi_\tau \cos e_{\chi_\tau} + \cos\Delta\chi_\tau \sin e_{\chi_\tau}\right) \\ &= -\sigma v_{\mathrm{k},\tau} \left(\frac{\sigma\delta/\delta_0 \cos e_{\chi_\tau}}{\sqrt{1 + (\delta/\delta_0)^2}} + \frac{\sin e_{\chi_\tau}}{\sqrt{1 + (\delta/\delta_0)^2}}\right) \\ &= \frac{-\sigma v_{\mathrm{k},\tau}}{\sqrt{1 + (\delta/\delta_0)^2}} \left(\sigma\delta/\delta_0 \cos e_{\chi_\tau} + \sin e_{\chi_\tau}\right) \end{aligned} \tag{40}$$

where the identities

$$\begin{aligned} \sin(\arctan(x)) &= x/\sqrt{1 + x^2} \\ \cos(\arctan(x)) &= 1/\sqrt{1 + x^2} \\ \sin(x + y) &= \sin(x)\cos(y) + \cos(x)\sin(y) \end{aligned} \tag{41}$$

have been utilized. If the course error dynamics are asymptotically stable i.e. $e_{\chi_\tau} \to 0$ then

$$\dot{\delta} = -v_{\mathrm{k},\tau} \frac{\delta/\delta_0}{\sqrt{1 + (\delta/\delta_0)^2}} \tag{42}$$

where the fact that $\sigma^2 = 1$ has been exploited. Equation (42) shows that if the commanded course according to Eq. (29) is tracked, the distance δ strictly decreases over time, which concludes the proof.

The input signal to the path-following controller will be the desired course and flight path angle rates. In an inversion based control approach these rates are usually obtained by filtering the corresponding course and flight path angles. From a geometrical point of view, the reference course rate contains information about the future course angle and hence is linked to the curvature of the path that needs to be followed. If the rate of a reference filter is used only an approximation is obtained if the to be followed path is not a straight line, or a combination thereof, which results in step commands in the course reference angle that only require a course rate in the transients. If the path curvature is

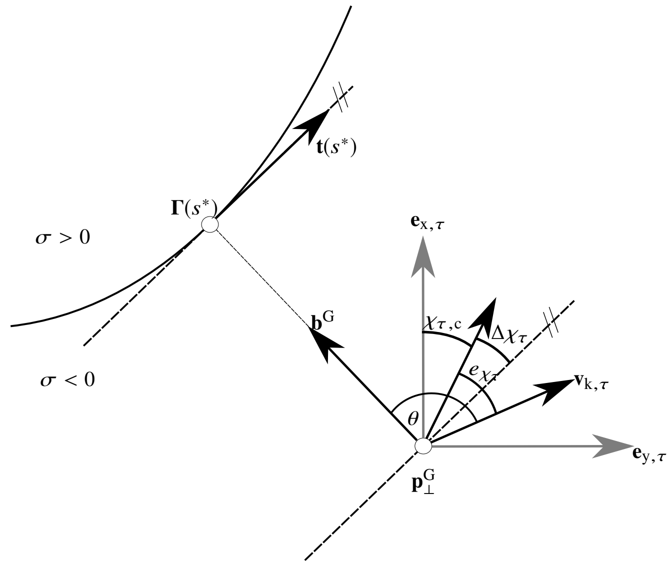

Fig. 16 Visualization of the angles utilized in Eq. (38).

not zero the approximated rate by the filter will not keep the system on the path since in general the rate of the filter does not correspond to the rate imposed by the geometry of the path. Hence, although the path-following controller would steer the aircraft towards the path, once the aircraft is on the path it would leave the path again, which can lead to unnecessary control effort and oscillations of the aircraft around the path. Theoretically, this effect can be minimized with high gain tracking error feedback, which however can lead to an unstable closed loop system. To avoid this behavior a different approach is pursued where the exact required course rate based on the path geometry will be calculated analytically instead of numerically using a filter. The commanded tangential plane course rate is given by $\chi_{\tau,\mathrm{c}}$, hence the commanded rate can be calculated by taking the derivative of the terms in Eq. (29) which yields

$$\dot{\chi}_{\tau,\mathrm{c}} = \dot{\chi}_{\tau,\parallel} + \dot{\Delta\chi_\tau} \tag{43}$$

with

$$\dot{\Delta\chi_\tau} = -\frac{\sigma/\delta_0}{1+(\delta/\delta_0)^2}\dot{\delta} \tag{44}$$

with Eq. (42) this leads to

$$\dot{\Delta\chi_\tau} = \frac{v_{\mathrm{k},\tau}\sigma/\delta_0^2}{\left(1+(\delta/\delta_0)^2\right)^{3/2}}\delta \tag{45}$$

It can be seen that for decreasing δ, hence small δ/δ_0, the contribution of $\dot{\Delta\chi_\tau}$ converges linearly to zero. Note, $\dot{\Delta\chi_\tau}$ is not linked to the path geometry directly. It improves however the path-following performance if $\delta \neq 0$. If $\dot{\Delta\chi_\tau}$ would be neglected only the course error feedback part would adapt $\dot{\chi}_{\tau,\parallel}$ such that the commanded course rate does not only contain a component that would keep the aircraft parallel to the path. Since this contribution is mainly required for

$\delta \neq 0$, a too high gain for the course tracking feedback would probably dominate also $\dot{\chi}_{\tau,\parallel}$ for $\delta = 0$. Hence, using a small gain for the course error feedback in combination with the additional feed-forward part $\Delta\dot{\chi}_\tau$ increases the performance of the path-following controller. The derivative of $\chi_{\tau,\parallel}$ is given by

$$\dot{\chi}_{\tau,\parallel} = \frac{d}{dt}\arctan\left(\frac{\mathbf{e}_{\mathrm{y},\tau}\cdot\mathbf{t}^{\mathrm{G}}}{\mathbf{e}_{\mathrm{x},\tau}\cdot\mathbf{t}^{\mathrm{G}}}\right) \tag{46}$$

$$\dot{\chi}_{\tau,\parallel} = \frac{\begin{pmatrix}\cos\left(\chi_{\tau,\parallel}\right) & -\sin\left(\chi_{\tau,\parallel}\right)\end{pmatrix}\begin{pmatrix}\left(\frac{\partial\mathbf{e}_{\mathrm{y},\tau}}{\partial\lambda}\dot{\lambda}+\frac{\partial\mathbf{e}_{\mathrm{y},\tau}}{\partial\phi}\dot{\phi}\right)\cdot\mathbf{t}^{\mathrm{G}}+\mathbf{e}_{\mathrm{y},\tau}\cdot\frac{d\mathbf{t}^{\mathrm{G}}}{ds}\dot{s}\\ \left(\frac{\partial\mathbf{e}_{\mathrm{x},\tau}}{\partial\lambda}\dot{\lambda}+\frac{\partial\mathbf{e}_{\mathrm{x},\tau}}{\partial\phi}\dot{\phi}\right)\cdot\mathbf{t}^{\mathrm{G}}+\mathbf{e}_{\mathrm{x},\tau}\cdot\frac{d\mathbf{t}^{\mathrm{G}}}{ds}\dot{s}\end{pmatrix}}{\|\mathbf{t}^{\mathrm{G}}\|} \tag{47}$$

with

$$\mathbf{e}_{\mathrm{x},\tau} = \begin{pmatrix}-\sin\phi\cos\lambda\\ -\sin\phi\sin\lambda\\ \cos\phi\end{pmatrix},\ \mathbf{e}_{\mathrm{y},\tau} = \begin{pmatrix}-\sin\lambda\\ \cos\lambda\\ 0\end{pmatrix} \tag{48}$$

and

$$\dot{\lambda} = \frac{v_{\mathrm{k}}^{\mathrm{G}}}{\|\left(\mathbf{p}^{\mathrm{G}}\right)_{\mathrm{W}}\|\cos\phi},\ \dot{\phi} = \frac{u_{\mathrm{k}}^{\mathrm{G}}}{\|\left(\mathbf{p}^{\mathrm{G}}\right)_{\mathrm{W}}\|} \tag{49}$$

where $u_{\mathrm{k}}^{\mathrm{G}}$ and $v_{\mathrm{k}}^{\mathrm{G}}$ are the x and y components of the kinematic velocity vector of the aircraft in the τ reference frame. Equation (47) defines the rate with which the angle between the tangent vector $\mathbf{t}^{\mathrm{G}}$ at the aircraft and the basis vector of the tangent plane frame $\mathbf{e}_{\mathrm{x},\tau}$ changes as a function of path geometry and aircraft velocity. It hence corresponds to the required course rate imposed by the path curvature.

Using kinematic manipulations the desired tangential plane course rate $\dot{\chi}_{\tau,\mathrm{c}}$ can be converted into the corresponding rates for the course and flight path angle $\dot{\chi}_{\mathrm{k,c}}$ and $\dot{\gamma}_{\mathrm{k,c}}$, respectively. The tangential plane course rate occurs in the angular velocity vector between the τ and the $\bar{K}$ frame for instance given in the rotated kinematic frame $\bar{K}$:

$$\left(\boldsymbol{\omega}^{\tau\bar{\mathrm{K}}}\right)_{\bar{\mathrm{K}}}^{\top} = \begin{pmatrix}-\dot{\chi}_\tau\sin\gamma_\tau & \dot{\gamma}_\tau & \dot{\chi}_\tau\cos\gamma_\tau\end{pmatrix}_{\bar{\mathrm{K}}} \tag{50}$$

The $\bar{K}$ frame is obatained by rotating the kinematic frame around the x_{K} axis by μ_{k} such that the $y_{\bar{\mathrm{K}}}$ axis is in the tangential plane. Note, in [8] it is assumed that $\gamma_\tau \approx 0$ which is only justified if the reeling-out speed is small compared to the onto the tangential plane projected kinematic velocity vector. Hence, Eq. (50) generalizes the result in [8]. Furthermore, Eq. (50) offers through $\dot{\gamma}_\tau$ another control degree of freedom that can be used to assist the winch controller in the radial direction motion control. In this work this has not been further investigated, hence $\dot{\gamma}_{\tau,\mathrm{c}}$ is set to zero.

$\left(\boldsymbol{\omega}^{\tau\bar{\mathrm{K}}}\right)_{\bar{\mathrm{K}}}$ can be converted into the angular velocity vector between the O and $\bar{K}$ frame, denoted with $\left(\boldsymbol{\omega}^{\mathrm{O}\bar{\mathrm{K}}}\right)_{\bar{\mathrm{K}}}$

according to

$$
\begin{aligned}
\left(\boldsymbol{\omega}^{\mathrm{O}\bar{\mathrm{K}}}\right)_{\bar{\mathrm{K}}} &= \\
&= \mathbf{M}_{\bar{\mathrm{K}}\mathrm{O}}\left(\left(\boldsymbol{\omega}^{\mathrm{OW}}\right)_{\mathrm{O}} + \mathbf{M}_{\mathrm{OW}}\left(\boldsymbol{\omega}^{\mathrm{W}\tau}\right)_{\mathrm{W}}\right) + \left(\boldsymbol{\omega}^{\tau\bar{\mathrm{K}}}\right)_{\bar{\mathrm{K}}}
\end{aligned} \tag{51}
$$

It is reasonable to assume that the mean wind direction changes much slower than the transport rate $\left(\boldsymbol{\omega}^{\mathrm{W}\tau}\right)_{\mathrm{W}}$ and the course rate vector $\left(\boldsymbol{\omega}^{\tau\bar{\mathrm{K}}}\right)_{\bar{\mathrm{K}}}$ hence $\left(\boldsymbol{\omega}^{\mathrm{OW}}\right)_{\mathrm{K}}$ can be set to zero. This yields

$$
\begin{aligned}
\left(\boldsymbol{\omega}^{\mathrm{O}\bar{\mathrm{K}}}\right)_{\bar{\mathrm{K}}} &= \mathbf{M}_{\bar{\mathrm{K}}\mathrm{O}}\mathbf{M}_{\mathrm{OW}}\left(\boldsymbol{\omega}^{\mathrm{W}\tau}\right)_{\mathrm{W}} + \left(\boldsymbol{\omega}^{\tau\bar{\mathrm{K}}}\right)_{\bar{\mathrm{K}}} \\
&= \begin{pmatrix} \dot{\mu}_{\mathrm{k}} - \dot{\chi}_{\mathrm{k,c}}\sin\gamma_{\mathrm{k}} \\ \dot{\gamma}_{\mathrm{k,c}}\cos\mu_{\mathrm{k}} + \dot{\chi}_{\mathrm{k,c}}\sin\mu_{\mathrm{k}}\cos\gamma_{\mathrm{k}} \\ -\dot{\gamma}_{\mathrm{k,c}}\sin\mu_{\mathrm{k}} + \dot{\chi}_{\mathrm{k,c}}\cos\mu_{\mathrm{k}}\cos\gamma_{\mathrm{k}} \end{pmatrix}_{\bar{\mathrm{K}}}
\end{aligned} \tag{52}
$$

with

$$
\left(\boldsymbol{\omega}^{\mathrm{W}\tau}\right)_{\mathrm{W}}^{\top} = \begin{pmatrix} \dot{\phi}\sin\lambda & -\dot{\phi}\cos\lambda & \dot{\lambda} \end{pmatrix}_{\mathrm{W}} \tag{53}
$$

Note, the second equality in Eq. (52) is a generic expression which can be obtained from the literature for instance from [14, p. 75]. The transformation matrix $\mathbf{M}_{\bar{\mathrm{K}}\mathrm{O}}$ can be calculated using the knowledge of course and path angle as well as the position of the aircraft in the W frame. With

$$
\begin{aligned}
\mathbf{e}_{\mathrm{x},\bar{\mathrm{K}},\mathrm{O}} &= \begin{pmatrix} \cos\chi_{\mathrm{k}}\cos\gamma_{\mathrm{k}} \\ \sin\chi_{\mathrm{k}}\cos\gamma_{\mathrm{k}} \\ -\sin\gamma_{\mathrm{k}} \end{pmatrix} \\
\mathbf{e}_{\mathrm{y},\bar{\mathrm{K}},\mathrm{O}} &= \frac{-\mathbf{M}_{\mathrm{OW}}\left(\mathbf{p}^{\mathrm{G}}\right)_{\mathrm{W}} \times \mathbf{e}_{\mathrm{x},\bar{\mathrm{K}},\mathrm{O}}}{\| -\mathbf{M}_{\mathrm{OW}}\left(\mathbf{p}^{\mathrm{G}}\right)_{\mathrm{W}} \times \mathbf{e}_{\mathrm{x},\bar{\mathrm{K}},\mathrm{O}} \|} \\
\mathbf{e}_{\mathrm{z},\bar{\mathrm{K}},\mathrm{O}} &= \mathbf{e}_{\mathrm{x},\bar{\mathrm{K}},\mathrm{O}} \times \mathbf{e}_{\mathrm{y},\bar{\mathrm{K}},\mathrm{O}}
\end{aligned} \tag{54}
$$

this yields

$$
\mathbf{M}_{\bar{\mathrm{K}}\mathrm{O}} = \begin{pmatrix} \mathbf{e}_{\mathrm{x},\bar{\mathrm{K}},\mathrm{O}}^{\top} \\ \mathbf{e}_{\mathrm{y},\bar{\mathrm{K}},\mathrm{O}}^{\top} \\ \mathbf{e}_{\mathrm{z},\bar{\mathrm{K}},\mathrm{O}}^{\top} \end{pmatrix} \tag{55}
$$

Ultimately, the desired course and path angle rates can be calculated according to

$$\begin{aligned}\dot{\chi}_{\mathrm{k,c}} &= \frac{\omega^{\mathrm{O}\bar{\mathrm{K}}}_{\mathrm{y},\bar{\mathrm{K}}} \sin\mu_{\mathrm{k}} + \omega^{\mathrm{O}\bar{\mathrm{K}}}_{\mathrm{z},\bar{\mathrm{K}}} \cos\mu_{\mathrm{k}}}{\cos\gamma_{\mathrm{k}}} \\ \dot{\gamma}_{\mathrm{k,c}} &= \omega^{\mathrm{O}\bar{\mathrm{K}}}_{\mathrm{y},\bar{\mathrm{K}}} \cos\mu_{\mathrm{k}} - \omega^{\mathrm{O}\bar{\mathrm{K}}}_{\mathrm{z},\bar{\mathrm{K}}} \sin\mu_{\mathrm{k}}\end{aligned} \tag{56}$$

with

$$\mu_{\mathrm{k}} = \arctan\left(\frac{M_{\bar{\mathrm{K}}\mathrm{O},23}}{M_{\bar{\mathrm{K}}\mathrm{O},33}}\right) \tag{57}$$

2. Retraction Phase Guidance

The retraction phase guidance module is separated from the traction phase module. The supervisory logic switches to the retraction phase according to the high-level state machine status. The outputs of the retraction guidance module are again course and path angle commands. In contrast to the traction phase the aircraft will not follow a prescribed path but directly flies towards the zenith position with a predefined path angle. The path angle set point is given by a fixed descend angle which is chosen manually. The course angle is calculated based on the relative position of the aircraft and the waypoint which is located at the zenith position of the small earth. The choice of this waypoint seems naturally because reeling in the tether will automatically pull the aircraft towards the zenith position. Additionally, in order to achieve a smoother transition back into the traction phase a *flare-like* maneuver is commanded that increases the descent rate linearly leading to a slight pull-up maneuver before the aircraft goes back into cross wind flight. The flare is initiated as a function of the aircraft latitude:

$$\begin{aligned}\gamma_{\mathrm{k,c}} &= \frac{\gamma_{\mathrm{f}} - \gamma_{\mathrm{i}}}{\phi_{\mathrm{max}} - \phi_0}\left(\phi - \phi_0\right) + \gamma_{\mathrm{i}} \\ \bar{\gamma}_{\mathrm{k,c}} &= \max(\min(\gamma_{\mathrm{k,c}}, \gamma_{\mathrm{f}}), \gamma_{\mathrm{i}})\end{aligned} \tag{58}$$

with $\phi_0 = \phi_{\mathrm{max}} - \Delta\phi$. The parameters $\Delta\phi, \gamma_{\mathrm{f}}, \gamma_{\mathrm{i}}$ are chosen manually by the operator and characterize the length of the flare, in terms of elevation angle, as well as the final and initial descent angle. The desired course angle is calculated based on the relative position of the aircraft and the origin of the wind frame:

$$\mathbf{b}_{\mathrm{O}}^{\top} = -\begin{pmatrix} p_{\mathrm{O,x}} & p_{\mathrm{O,y}} & 0 \end{pmatrix} \tag{59}$$

The course set point is then given by

$$\chi_{\mathrm{k,c}} = \arctan_2\left(b_{\mathrm{O,y}}, b_{\mathrm{O,x}}\right) \tag{60}$$

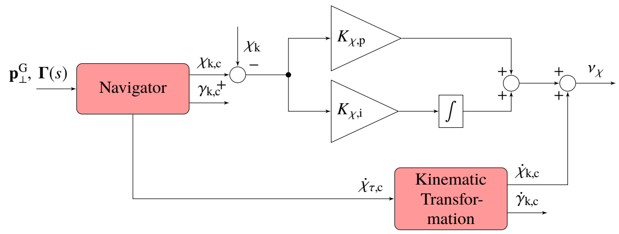

Fig. 17 Course controller block diagram.

C. Path Loop

1. Traction Phase

In the path loop the commanded course and path angle as well as their corresponding rates (output of guidance module) are used to calculate the set points for the attitude loop. The overall pseudo control inputs are given by

$$\begin{aligned}
\nu_\chi &= \dot{\chi}_{k,c} + k_{p,\chi}\left(\chi_{k,c} - \chi_k\right) + k_{i,\chi}\int_0^t \left(\chi_{k,c} - \chi_k\right) d\tau \\
\nu_\gamma &= \dot{\gamma}_{k,c} + k_{p,\gamma}\left(\gamma_{k,c} - \gamma_k\right) + k_{i,\gamma}\int_0^t \left(\gamma_{k,c} - \gamma_k\right) d\tau
\end{aligned} \tag{61}$$

The set points of the attitude controller will be derived using a model for the path dynamics. The total acceleration of the aircraft in the kinematic frame is given by:

$$\begin{aligned}
(\dot{\mathbf{v}}_k)_K^O &= \begin{pmatrix} \dot{v}_k \\ 0 \\ 0 \end{pmatrix}_K + \left(\omega^{OK}\right)_K \times \begin{pmatrix} v_k \\ 0 \\ 0 \end{pmatrix}_K = \begin{pmatrix} \dot{v}_k \\ \dot{\chi}_k \cos\gamma_k v_k \\ -\dot{\gamma}_k v_k \end{pmatrix}_K \\
&= \begin{pmatrix} a_{x,K} \\ a_{y,K} \\ a_{z,K} \end{pmatrix}_K
\end{aligned} \tag{62}$$

The path dynamic are then defined according to

$$m \begin{pmatrix} a_{x,K} \\ a_{y,K} \\ a_{z,K} \end{pmatrix}_K = (\mathbf{F}_a)_K + \left(\mathbf{F}_g\right)_K + (\mathbf{F}_t)_K \tag{63}$$

involving the aerodynamic force $(\mathbf{F}_\mathrm{a})_\mathrm{K} \in \mathbb{R}^{3\mathrm{x}1}$, gravitational force $\left(\mathbf{F}_\mathrm{g}\right)_\mathrm{K} \in \mathbb{R}^{3\mathrm{x}1}$ as well as the tether force $(\mathbf{F}_\mathrm{t})_\mathrm{K} \in \mathbb{R}^{3\mathrm{x}1}$ in the K frame, where gravity and tether force are calculated with

$$\left(\mathbf{F}_\mathrm{g}\right)_\mathrm{K}^\top = \begin{pmatrix} -\sin\gamma_\mathrm{k} m_\mathrm{k} g & 0 & \cos\gamma_\mathrm{k} m_\mathrm{k} g \end{pmatrix} \tag{64}$$

and

$$(\mathbf{F}_\mathrm{t})_\mathrm{K} = -\mathbf{M}_\mathrm{KO} \frac{(\mathbf{p})_\mathrm{O}}{\left\|(\mathbf{p})_\mathrm{O}\right\|_2} F_\mathrm{t} \tag{65}$$

Solving for the aerodynamic force yields

$$\begin{pmatrix} f_\mathrm{x,a,K} \\ f_\mathrm{y,a,K} \\ f_\mathrm{z,a,K} \end{pmatrix}_\mathrm{K} = m \begin{pmatrix} a_\mathrm{x,K} \\ a_\mathrm{y,K} \\ a_\mathrm{z,K} \end{pmatrix}_\mathrm{K} - \left(\mathbf{F}_\mathrm{g}\right)_\mathrm{K} - (\mathbf{F}_\mathrm{t})_\mathrm{K} = (\mathbf{F}_\mathrm{a})_\mathrm{K} \tag{66}$$

The last two rows can be written as

$$\begin{aligned} f_\mathrm{y,a,K} &= \cos\mu_\mathrm{k} f_{\mathrm{a,y,\bar{K}}} - \sin\mu_\mathrm{k} f_{\mathrm{a,z,\bar{K}}} \\ f_\mathrm{z,a,K} &= \sin\mu_\mathrm{k} f_{\mathrm{a,y,\bar{K}}} + \cos\mu_\mathrm{k} f_{\mathrm{a,z,\bar{K}}} \end{aligned} \tag{67}$$

where μ_k is the kinematic bank angle, i.e. the rollangle around the kinematic velocity vector and

$$\begin{aligned} f_{\mathrm{a,y,\bar{K}}} &= -\cos\alpha_\mathrm{k} \sin\beta_\mathrm{k} f_\mathrm{a,x,B} \\ &\quad + \cos\beta_\mathrm{k} f_\mathrm{a,y,B} - \sin\alpha_\mathrm{k} \sin\beta_\mathrm{k} f_\mathrm{a,z,B} \\ f_{\mathrm{a,z,\bar{K}}} &= -\sin\alpha_\mathrm{k} f_\mathrm{a,x,B} + \cos\alpha_\mathrm{k} f_\mathrm{a,z,B} \end{aligned} \tag{68}$$

Note that α_k and β_k are the kinematic angle of attack and kinematics sideslip angle. Since the inner loop controller actively controls the sideslip angle β, i.e. the aerodynamic sideslip angle, the aerodynamic side force $f_\mathrm{a,y,B}$ is approximately zero. Contrarily, the kinematic sideslip angle β_k is in presence of wind not zero. Hence,

$$\begin{aligned} f_{\mathrm{a,y,\bar{K}}} &= -\cos\alpha_\mathrm{k} \sin\beta_\mathrm{k} f_\mathrm{a,x,B} - \sin\alpha_\mathrm{k} \sin\beta_\mathrm{k} f_\mathrm{a,z,B} \\ f_{\mathrm{a,z,\bar{K}}} &= -\sin\alpha_\mathrm{k} f_\mathrm{a,x,B} + \cos\alpha_\mathrm{k} f_\mathrm{a,z,B} \end{aligned} \tag{69}$$

The set point for the kinematic bank angle based on the required course and path angle rate is calculated by solving Eq. (67) for μ_k and inserting the pseudo control signals for the course and path angle rates:

$$\begin{aligned}\mu_{\mathrm{k,c}} &= \operatorname{arctan}_2\left(\frac{m_\mathrm{k} \nu_\chi \cos\gamma_\mathrm{k} v_\mathrm{k} - f_{\mathrm{t,y,K}}}{m_\mathrm{k} \nu_\gamma v_\mathrm{k} + m_\mathrm{k} g \cos\gamma_\mathrm{k} + f_{\mathrm{t,z,K}}}\right)\\ &+ \arctan\left(\frac{f_{\mathrm{a,y,\bar{K}}}}{f_{\mathrm{a,z,\bar{K}}}}\right)\end{aligned} \tag{70}$$

which requires estimates for the aerodynamic forces $f_{\mathrm{a,y,\bar{K}}}$ and $f_{\mathrm{a,z,\bar{K}}}$.

Based on the set point for the kinematic banking angle the corresponding Euler roll angle can be calculated according to

$$\Phi_\mathrm{c} = \arcsin\left(\frac{\cos\gamma_\mathrm{k}\cos\beta_\mathrm{k}\left(\sin\mu_{\mathrm{k,c}} - \tan\gamma_\mathrm{k}\tan\beta_\mathrm{k}\right)}{\cos\Theta}\right) \tag{71}$$

Equation (71) can be obtained by comparing the relevant coefficients of $\mathbf{M}_{\mathrm{B}\tau} = \mathbf{M}_{\mathrm{BO}}\mathbf{M}_{\mathrm{OW}}\mathbf{M}_{\mathrm{W}\tau}$. The matrix $\mathbf{M}_{\mathrm{BO}}$ is obtained for instance from [12, p. 12]. The matrix $\mathbf{M}_{\mathrm{W}\tau}$ is equivalent to the transformation from the Earth-Centered-Earth-Fixed (E) frame into the O frame (see [12, p. 31]) where the E frame corresponds to the W frame and the O frame corresponds to the τ frame. $\mathbf{M}_{\mathrm{OW}}$ is given by

$$\mathbf{M}_{\mathrm{OW}} = \begin{pmatrix}\cos\xi & \sin\xi & 0\\ \sin\xi & -\cos\xi & 0\\ 0 & 0 & -1\end{pmatrix} \tag{72}$$

where ξ denotes the wind direction measured from the north direction. Note, the structure of $\mathbf{M}_{\mathrm{B}\tau}$ is equivalent to the structure of $\mathbf{M}_{\mathrm{BO}}$. Φ_c can then be transformed into an aerodynamic banking angle command $\mu_{\mathrm{a,c}}$ using Eq. (73).

$$\mu_{\mathrm{a,c}} = \arcsin\left(\frac{\cos\Theta\sin\Phi_\mathrm{c}}{\cos\gamma_\mathrm{a}\cos\beta} + \tan\gamma_\mathrm{a}\tan\beta\right) \tag{73}$$

The required aerodynamic path angle can be calculated using Eq. (74), which has been derived in [13, p. 20-23].

$$\begin{aligned}\gamma_\mathrm{a} &= \arcsin\left(\frac{v_\mathrm{k}\sin\gamma_\mathrm{k} + v_{\mathrm{w,O,z}}}{v_\mathrm{a}}\right)\\ &\approx \arcsin\left(\frac{v_\mathrm{k}\sin\gamma_\mathrm{k}}{v_\mathrm{a}}\right)\end{aligned} \tag{74}$$

Notice that, the calculation of γ_a requires the knowledge of the wind component in z_O direction $v_{\mathrm{w,O,z}}$ which is however usually negligibly small compared to the horizontal components. The angle of attack set point can be calculated similarly to the approach presented in [25] with

$$L_{\mathrm{req}} \approx \sqrt{\bar{f}_{\mathrm{y,K}}^2 + \bar{f}_{\mathrm{z,K}}^2} \tag{75}$$

Note, due to the wind influence this is only an approximation which is neglected in [25]. However, since the available traction force needs to be maximized it makes sense to choose a fixed set point during the traction phase close to the maximum angle of attack. Setting the angle of attack to a fixed value is similar to the case where the angle of attack saturates. This can lead to a windup of the integrators in the path loop. One approach so mitigate the windup is to adapt the reference model by the control deficit that results from the saturation (i.e. pseudo control hedging, PCH). However, for the traction phase controller the reference course rate is directly calculated based on the path geometry, as discussed in the previous section. This prevents a standard implementation of PCH, since no reference filter is used. Instead, an anti-windup scheme based on *back-calculation* is used, where the feedback part corresponds to the deficit between for instance the commanded course rate $\nu_{\chi,\mathrm{k,c}}$ and the expected course rate $\hat{\dot{\chi}}_{\mathrm{k,c}}$. The hedge signal is in this case defined by

$$\nu_{\mathrm{h},\chi} = k_{\mathrm{bc}} \left(\nu_{\chi,\mathrm{k,c}} - \hat{\dot{\chi}}_{\mathrm{k,c}} \right) \tag{76}$$

The gain k_{bc} is chosen to be smaller than the integrator gain, as recommended in [26, pp. 79-80]. The feedback law for the pseudo control input is then adapted according to

$$\begin{aligned} \nu_{\chi} = \dot{\chi}_{\mathrm{k,c}} &+ k_{\mathrm{p},\chi} \left(\chi_{\mathrm{k,c}} - \chi_{\mathrm{k}} \right) \\ &+ k_{\mathrm{i},\chi} \int_0^{\mathrm{t}} \left(\chi_{\mathrm{k,c}} - \chi_{\mathrm{k}} - \nu_{\mathrm{h},\chi} \right) d\tau \end{aligned} \tag{77}$$

The adaption of the flight path rate channel follows analogously.

2. Retraction Phase

For the retraction phase the course and path angle controller are designed similarly to the traction phase controller, the only difference consists of the calculation of the course and path angle rate commands. Since in the retraction phase no defined path needs to be followed, the rate commands are generated with second order reference filters. Although first order filters would be sufficient second order filters lead to an additional smoothing of the derivatives [27]. Instead of using a back-calculation anti-windup scheme a conventional PCH approach is chosen using estimates for the feasible course and path angle rates. With the hedging signal ν_{h} the equations of the second order filter, here displayed for the course filter, are defined by

$$\begin{aligned} \dot{\nu}_{\mathrm{r},\chi} &= -2\zeta\omega_0\nu_{\mathrm{r},\chi} + \omega_0^2 \left(\chi_{\mathrm{k,c}} - \chi_{\mathrm{k,r}} \right) \\ \dot{\chi}_{\mathrm{k,r}} &= \nu_{\mathrm{r},\chi} - \nu_{\mathrm{h}} \end{aligned} \tag{78}$$

and an equivalent pseudo control law with PI controller as for the traction phase is used (see Eq. (61)). Note, in contrast to a fixed value for the angle of attack set point, the approximate expression of the required lift in Eq. (75) is

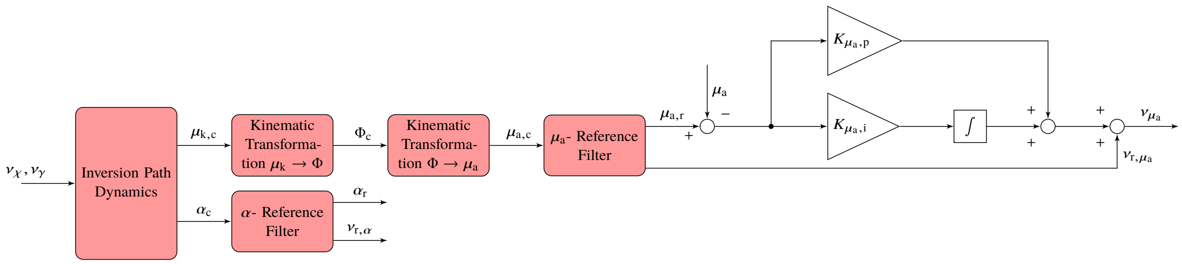

Fig. 18 Path and attitude loop block diagram.

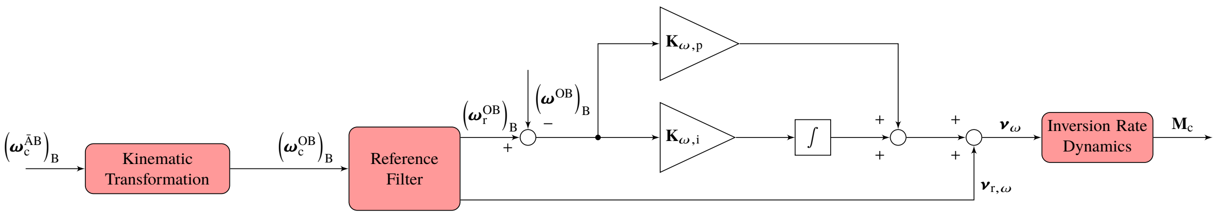

Fig. 19 Rate loop block diagram.

used to determine the corresponding lift coefficient and by inversion of the lift coefficient the angle of attack set point α_c for the attitude loop is determined.

D. Attitude Loop

The pseudo-control inputs for the attitude to rate inversion are given by

$$\begin{aligned}
\nu_{\mu_a} &= \nu_{r,\mu_a} + K_{\mu,p}\left(\mu_{a,r} - \mu_a\right) + K_{\mu,i}\int_0^t \left(\mu_{a,r} - \mu_a\right) d\tau \\
\nu_\alpha &= \nu_{r,\alpha} + K_{\alpha,p}\left(\alpha_r - \alpha\right) + K_{\alpha,i}\int_0^t \left(\alpha_r - \alpha\right) d\tau \\
\nu_\beta &= K_{\beta,p}\left(\beta_r - \beta\right) + K_{\beta,i}\int_0^t \left(\beta_r - \beta\right) d\tau
\end{aligned} \tag{79}$$

where ν_{r,μ_a} and $\nu_{r,\alpha}$ are calculated with an equivalent reference filter as defined for the course angle in Eq. (78). The inversion of the attitude to rate dynamics is purely kinematic and given by

$$\left(\boldsymbol{\omega}_c^{OB}\right)_B = \mathbf{M}_{B\bar{A}} \begin{pmatrix} -\dot{\chi}_a \sin\gamma_a \\ \dot{\gamma}_a \\ \dot{\chi}_a \cos\gamma_a \end{pmatrix}_{\bar{A}} + \left(\boldsymbol{\omega}^{\bar{A}B}\right)_B \tag{80}$$

with

$$\left(\boldsymbol{\omega}^{\bar{A}B}\right)_B = \begin{pmatrix} \cos\alpha\cos\beta\nu_\mu + \nu_\beta \sin\alpha \\ \sin\beta\nu_\mu + \nu_\alpha \\ \sin\alpha\cos\beta\nu_\mu - \cos\alpha\nu_\beta \end{pmatrix}_B \tag{81}$$

The matrix $\mathbf{M}_{\bar{A}B}$ is defined for instance in [14, p. 62]. $\dot{\chi}_a$ and $\dot{\gamma}_a$ are estimated by filtering Eq. (74) and Eq. (82), as derived in [13, p. 23]

$$\chi_a = \chi_k + \beta - \arcsin\left(\frac{1}{V_a \cos \gamma_a}\left(v_{w,O,y} \cos \chi_{k,c} - v_{w,O,x} \sin \chi_{k,c}\right)\right) \tag{82}$$

using a washout-filter, as proposed in [28]:

$$G(s) = \frac{s\omega_f^2}{s^2 + 2\omega_f s + \omega_f^2} \tag{83}$$

where $\omega_f = 90\,\mathrm{rad\,s^{-1}}$ is the chosen filter bandwidth. Note, a better accuracy could be achieved by calculating $\dot{\chi}_k$ and $\dot{\gamma}_k$ analytically using the model of the course and path rate dynamics as defined in Eq. (63) and only filter the remaining terms. Alternatively, a model can be used to estimate $\dot{\chi}_a$ and $\dot{\gamma}_a$ which requires to write down the path dynamics with respect to the aerodynamic frame assuming a stationary wind field.

E. Rate Loop

Note, since it is assumed that the tether is connected close to the center of gravity of the aircraft the rate loop of the tethered aircraft can be implemented analogously to the rate loop of a conventional aircraft. In the literature there exists an ample amount of different approaches to control the rate dynamics of aircraft, in this work a conventional first order dynamic inversion controller with second order reference filters and an incremental control allocation as presented in [25] is used. Note, the incremental approach is necessary since in general the relationship between actuator inputs and aerodynamic moments is nonlinear and not globally invertible. Since up to now and in the future extensive effort is and will be put into the modeling and identification of the AWE system, a model-based inversion is chosen over a sensor-based inversion as for instance presented in [29].

The commanded attitude rates as calculated by Eq. (80) are filtered and the resulting rate accelerations are added to a PI control part analogously to Eq. (79) yielding the pseudo-control input $\boldsymbol{\nu}_\omega$ for the inversion of the rate dynamics as defined in Eq. (4). From the resulting moment the current acting moment on the aircraft, estimated using a model, is subtracted yielding the required moment increment to track the commanded rates:

$$\begin{aligned}\begin{pmatrix}\Delta L\\ \Delta M\\ \Delta N\end{pmatrix} &= \begin{pmatrix}L_c\\ M_c\\ N_c\end{pmatrix} - \begin{pmatrix}L_0\\ M_0\\ N_0\end{pmatrix}\\ &= \mathbf{J}\boldsymbol{\nu}_\omega + \left(\boldsymbol{\omega}^{OB}\right)_B \times \mathbf{J}\left(\boldsymbol{\omega}^{OB}\right)_B - \begin{pmatrix}L_0\\ M_0\\ N_0\end{pmatrix}\end{aligned} \tag{84}$$

F. Control Allocation

Eventually, the moment increments are mapped to a surface deflection increment that is added to the current surface deflection resulting in the final actuator command:

$$\begin{aligned}\begin{pmatrix}\delta_{a,c}\\ \delta_{e,c}\\ \delta_{r,c}\end{pmatrix} &= \begin{pmatrix}\delta_{a,0}\\ \delta_{e,0}\\ \delta_{r,0}\end{pmatrix} + \begin{pmatrix}\Delta\delta_a\\ \Delta\delta_e\\ \Delta\delta_r\end{pmatrix}\\ &= \begin{pmatrix}\delta_{a,0}\\ \delta_{e,0}\\ \delta_{r,0}\end{pmatrix} + \begin{pmatrix}C_{l\delta_a} & 0 & C_{l\delta_r}\\ 0 & C_{m\delta_e} & 0\\ C_{n\delta_a} & 0 & C_{n\delta_r}\end{pmatrix}^{-1} \begin{pmatrix}\Delta L\\ \Delta M\\ \Delta N\end{pmatrix}\end{aligned} \tag{85}$$

where the $C_{i,j}$ coefficients represent roll- (L), pitch- (M) or yaw-moment (N) control derivatives that are obtained by linearizing the aerodynamic moment model with respect to the control surface deflections.

G. Winch Controller

The winch controller is derived based on the model defined in Eq. (9) without explicitly taking into account the aircraft dynamics as presented for instance in [30]. The reason is that if the aircraft dynamics are taken into account, the full state vector of the aircraft needs to be available to the winch controller including a tether model with measurable states. So far no reliable information about the communication between the aircraft and the ground station is available and feedback of tether states is not practical. Hence, it is decided to control the winch only based on the measured tether force on the ground. In AWE, two high level control objectives for the winch controller can be formulated. First, the net power output has to be maximized by controlling the radial motion of the aircraft in an optimal way, second, the winch controller needs to prevent too high tension in the tether, for instance as a result of sudden wind speed changes, which would lead to a tether rupture or damage of the aircraft. In this work, the focus is on the second control objective, since it is more critical for the reliable operation of the AWE system.

Note, from the perspective of the winch, the dynamics of the aircraft and the tether represent a disturbance that the winch controller needs to regulate in order to track a force set point. If a tether force measurement on the ground is available, which is usually the case in this application, a complex disturbance model is not necessary because all relevant information is condensed in the force measurement. Note that this approach assumes implicitly that the difference between the tether force measured on the ground and the tether force measured at the aircraft is negligible. Simulation results show that this assumption is valid during the traction phase. The set point for the reeling speed can be derived as follows. The aircraft dynamics in the tangential plane, or spherical coordinates, are given by

$$\left(\dot{\mathbf{v}}^{\mathrm{G}}\right)_{\tau} + (\omega)_{\tau}^{\mathrm{W}\tau} \times \left(\mathbf{v}^{\mathrm{G}}\right)_{\tau} = \frac{\left(\mathbf{F}_{\mathrm{g}}\right)_{\tau} + (\mathbf{F}_{\mathrm{a}})_{\tau} + (\mathbf{F}_{\mathrm{t}})_{\tau}}{m_{\mathrm{k}}} \tag{86}$$

Assuming a straight tether only the third row is relevant which is given by

$$\dot{v}_{\mathrm{z},\tau} = -\omega_{\mathrm{x}} v_{\mathrm{y},\tau} + \omega_{\mathrm{y}} v_{\mathrm{x},\tau} + \frac{F_{\mathrm{g,z},\tau} + F_{\mathrm{a,z},\tau} + F_{\mathrm{t}}}{m_{\mathrm{k}}} \tag{87}$$

This can be written more compactly as

$$\dot{v}_{\mathrm{z},\tau} = \frac{F_{\mathrm{aircraft}} + F_t}{m_{\mathrm{k}}} \tag{88}$$

with

$$F_{\mathrm{aircraft}} = m_{\mathrm{k}} \left(-\omega_{\mathrm{x}} v_{\mathrm{y},\tau} + \omega_{\mathrm{y}} v_{\mathrm{x},\tau}\right) + F_{\mathrm{g,z},\tau} + F_{\mathrm{a,z},\tau} \tag{89}$$

Note, F_{aircraft} requires the knowledge of the full aerodynamic model of the aircraft as well as the relevant measured states if used for the set point calculation. However, instead of an estimation of F_{aircraft} the measured tether force on the ground can be used, if it is assumed that $F_{\mathrm{aircraft}} \approx -F_{\mathrm{t,m}}$. If the tether is straight, the reeling speed v_{r} is equal to $-v_{\mathrm{z},\tau}$, hence

$$\dot{v}_{\mathrm{r}} = \frac{F_{\mathrm{t,m}} - F_{\mathrm{t}}}{m_{\mathrm{k}}} \tag{90}$$

If F_{t} is replaced by the desired traction force $F_{\mathrm{t,c}}$ the resulting acceleration can be interpreted as a reference acceleration proportional to the tether force tracking error. With $\dot{\omega}_{\mathrm{w}} = \dot{v}_{\mathrm{r}}/r_{\mathrm{w}}$ this expression can be substituted into the winch model in Eq. (9) and solved for the reference torque:

$$M_{\mathrm{c}} = \left(\frac{J_{\mathrm{w}}}{r_{\mathrm{w}} m_{\mathrm{k}}} - r_{\mathrm{w}}\right) F_{\mathrm{t,m}} - \frac{J_{\mathrm{w}}}{r_{\mathrm{w}} m_{\mathrm{k}}} F_{\mathrm{t,c}} \tag{91}$$

Substituting this expression back into the winch model yields the closed loop winch model

$$\dot{\omega}_{\mathrm{w}} = \frac{1}{r_{\mathrm{w}} m_{\mathrm{k}}} \left(F_{\mathrm{t,m}} - F_{\mathrm{t,c}}\right) + \Delta_{\mathrm{w}} \tag{92}$$

where Δ_{w} is the model mismatch as a result of an imperfect inversion of the plant dynamics. Note, if the measured tether force deviates from the set point the winch will reel out faster or slower. Although simple, this approach proved to be highly effective in dealing with varying wind conditions and wind gusts as will be shown in section IV, while being independent of any aircraft state. In order to get rid of steady state errors an integrator term $k_{\mathrm{i}} \int_0^{\mathrm{t}} F_{\mathrm{m}} - F_{\mathrm{t,s}} d\tau$ can be added to Eq. (91). For the stability of Eq. (92) only a qualitative but intuitive stability proof is given. If the tether force becomes larger than the set point force the winch will start to accelerate according to Eq. (92). Of course this is strictly only true if $\frac{1}{r_{\mathrm{w}} m_{\mathrm{k}}} \left(F_{\mathrm{m}} - F_{\mathrm{t,s}}\right) > \Delta_{\mathrm{w}}$. However, in the opposite case the acceleration will only be delayed, since if the winch further decelerates the tension in the tether would increase steadily until the tracking error contribution will be larger than Δ_{w}. If the winch accelerates the kinematic radial speed of the aircraft will increase which decreases the apparent wind speed. As a consequence the lift force will drop, which decreases the tension in the tether and therefore decreases the tether force tracking error. The causal chain holds of course for the opposite case as well, where the tether force is smaller than the force set point.

During the retraction phase the reeling-in speed is set to a fixed value, usually the maximum reeling-in speed that the winch can achieve is chosen in order to minimize the retraction time. For the tracking task of the speed controller a dynamic model based feed-forward controller (see [31, pp. 324-328]) for fast tracking is combined with a linear quadratic feedback regulator with servomechanism [32, pp. 51-62]. The prefilter is used to create smooth transitions between set point changes. Additionally, a feed-forward disturbance compensation is added since from the perspective of the speed controller the tether force represents a measurable disturbance.

IV. Results

In this section two different simulation campaigns are used to investigate the robustness of the control system. First, the robustness with respect to modest changes in the wind speed due to turbulence and wind shear is assessed. In the second part, the effect on the control performance due to sudden and significant wind speed changes caused by gusts is analyzed.

A. Consecutive Pumping Cycles in a Turbulent Wind Field

Fig. 20 and Fig. 21 show the resulting flight paths projected into the $x_{\mathrm{W}} z_{\mathrm{W}}$ and $x_{\mathrm{W}} y_{\mathrm{W}}$ plane, respectively. Fig. 22 depicts the path projected into the tangential plane at $\lambda = 0°$ and $\phi = \phi_0$ (center of the figure of eight). Despite the turbulent wind field, shown in Fig. 23 and Fig. 24, the control system is able to guide the aircraft along the defined flight path reliably. The visible deviations between the reference path (light grey curve in Fig. 22) and the real flight path are acceptable and are caused by the limited bandwidth of the control system. This limitation results in a repetitive non-zero cross track error during the turns. The results display roughly three consecutive pumping cycles. The reoccurring flight pattern demonstrates the robustness of the closed loop system towards modest changes in wind speed caused by wind shear and turbulence.

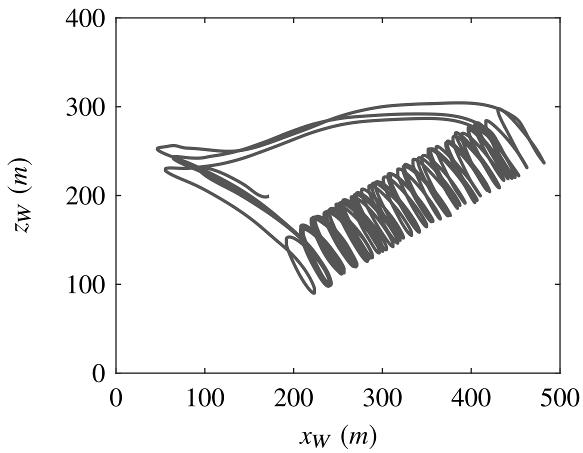

Fig. 20 Flight path in $x_W z_W$-plane).

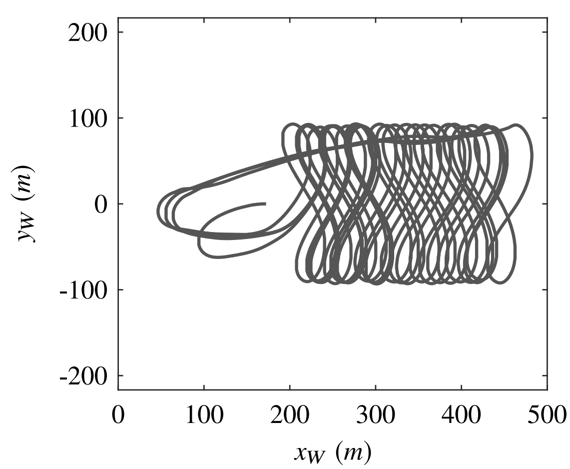

Fig. 21 Flight path in $x_W y_W$-plane.

As described in section III, reference filters are used to generate the course and path angle rates during the retraction phase. This allows to implement PCH to adapt the reference filters in case of saturation of the control signal. From the point of view of the path loop, the control signals are the bank angle command $\mu_{a,c}$ as well as the angle of attack command α_c. In Fig. 25 it can be observed that during a significant part of the retraction phase, e.g. for instance between

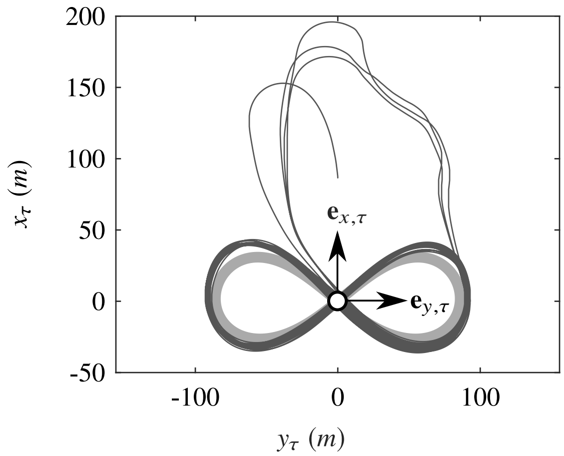

Fig. 22 Figure of eight flight path projected into the tangential plane.

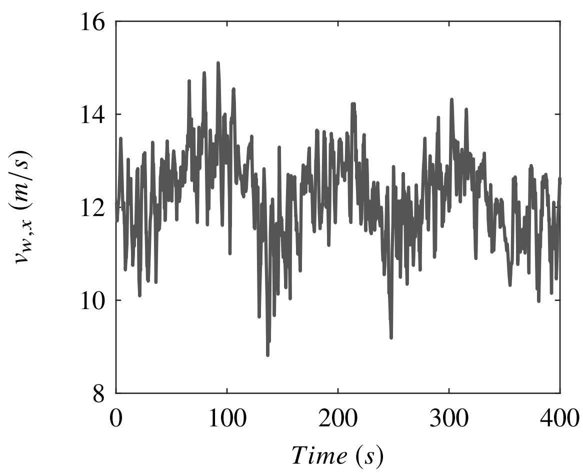

Fig. 23 x component of the wind velocity vector in the *W* frame.

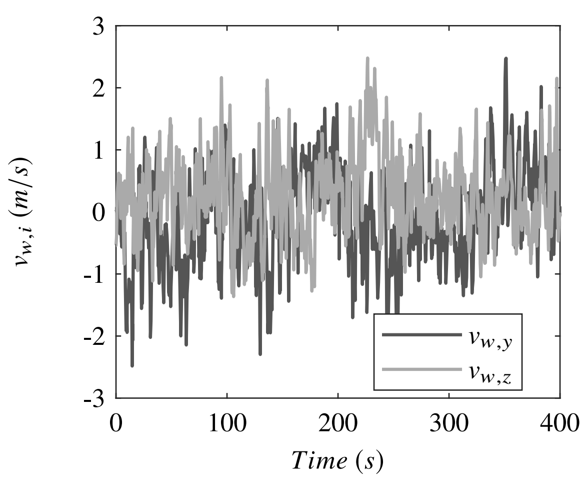

Fig. 24 y and z components of the wind velocity vector in the *W* frame.

226 s and 234 s, the angle of attack is saturating. In this case the commanded pseudo-control inputs ν_γ and ν_χ will

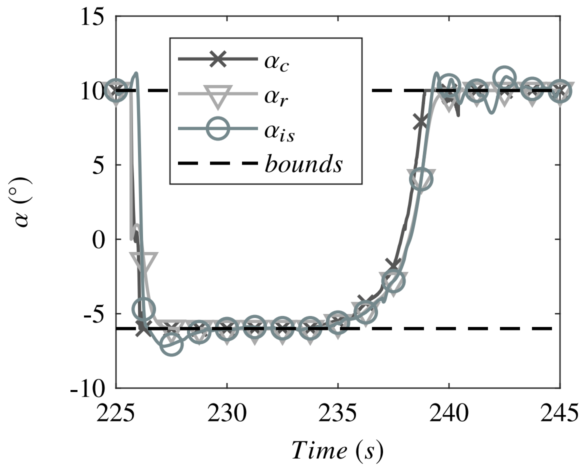

Fig. 25 Angle of attack tracking.

deviate from the actual plant responses. The adaptation of the course and path angle reference filters can be observed in Fig. 26 and Fig. 27. The effect is especially visible for the path angle whose primary control variable is the angle of attack. As the angle of attack is saturating the reference path angle increases (e.g. at ≈ 226 s) as a result of the hedge signal before it decreases again and eventually converges towards the negative commanded set point $\gamma_{\mathrm{k,c}}$.

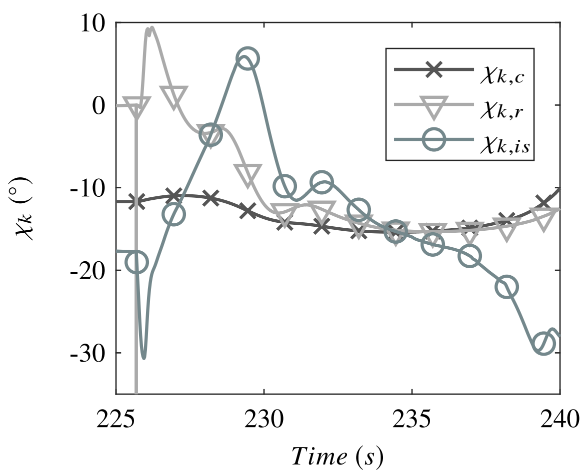

Fig. 26 Course angle tracking during retraction.

During the pumping cycles the sideslip angle varies most of the time between −2° to +2°. Larger sideslip angles occur during the transition phases from traction into retraction and vice verse as can be seen in Fig. 28. The evolution of the aircraft control surface deflections is depicted in Fig. 29. It can be observed that the highest control effort is required in the transition phases where the control surfaces partially saturate. During the traction phases the aileron δ_{a} and rudder δ_{r} inputs vary in a repetitive manner between −5° to +5° while the elevator deflection δ_{e} remains almost

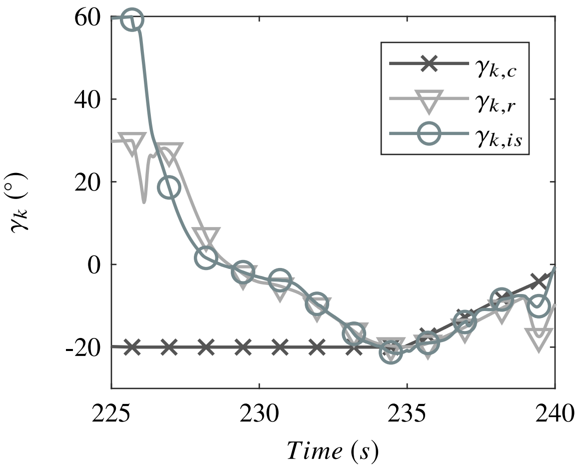

Fig. 27 Flight path angle tracking during retraction.

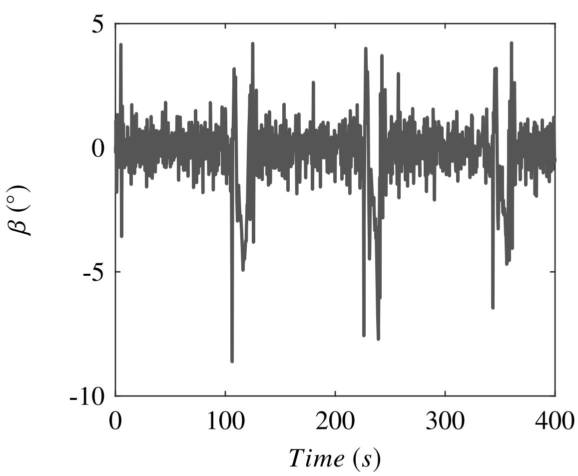

Fig. 28 Sideslip angle regulation.

constant at around $-9°$ as a results of the fixed angle of attack set point during the traction phase.

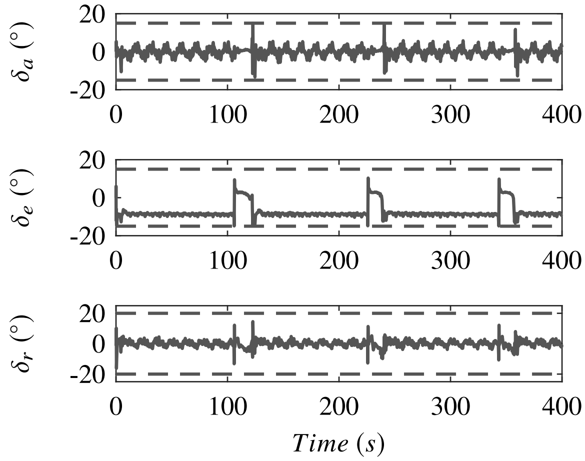

Fig. 29 Control surface deflections with limits (dashed lines).

Besides the analysis of the flight control performance the winch control performance needs to be assessed. Fig. 30 shows the evolution of the tether force as measured on the ground. During the conducted simulations a tether force set point of 1000 N is chosen, which is well beyond the structural limitations of around 1500 N. The tether force oscillates around the set point with an amplitude of around 50 N to 100 N. The oscillations are a result of the continues acceleration and deceleration of the aircraft while flying down and upwards during the figure of eight flight patterns. To further reduce these oscillations an improved feed-forward winch controller could be implemented in the future that systematically reels out slower during upward and faster during downward flight. At the moment this is partially achieved via feedback control of the tether force. Furthermore, the resulting variations in the reeling speed depicted in Fig. 31 should be reduced in the future since variations in reeling speed would lead to large oscillations in the mechanical power output in combination with a constant tether force. One option to tackle this problem would be to use the pitch channel of the aircraft to control the airspeed, which is out of the scope of this paper.

B. Robustness towards Wind Gusts

In this section the robustness of the control system towards rapid changes in the mean wind speed will be analyzed. For that purpose a *mexican hat* gust as defined in [33] is implemented and activated during the simulation at a specified instant in time. In this work only the response of the aircraft towards gusts in up- and downwind direction as depicted in Fig. 32 and Fig. 33 is analyzed. In both cases the gust leads to a significant increase or decrease in airspeed and therefore tether force (see Fig. 34 and Fig. 35). In order to keep the tether force around the set point the winch controller has to adapt the reeling out speed according to Eq. (91) (see Fig. 36 and Fig. 37). It can be observed that the reeling speed change follows the shape of the gust proportionally. The adaptation of the reeling speed has a direct effect on the

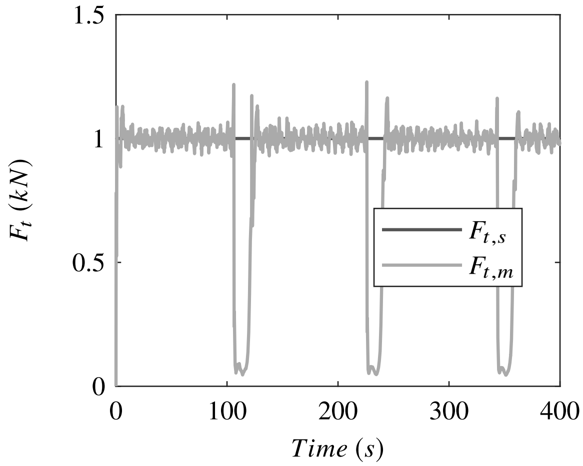

Fig. 30   Tether force tracking.

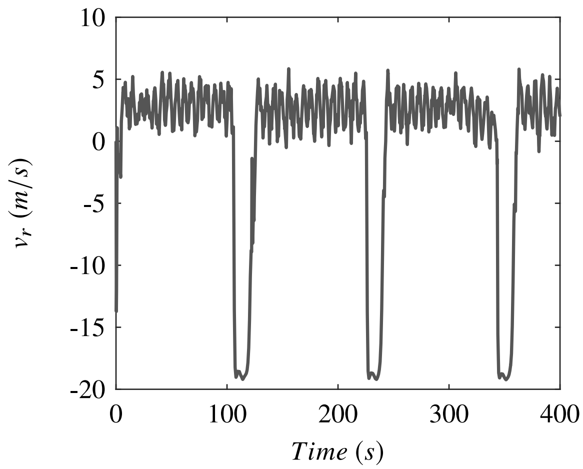

Fig. 31   Reeling speed.

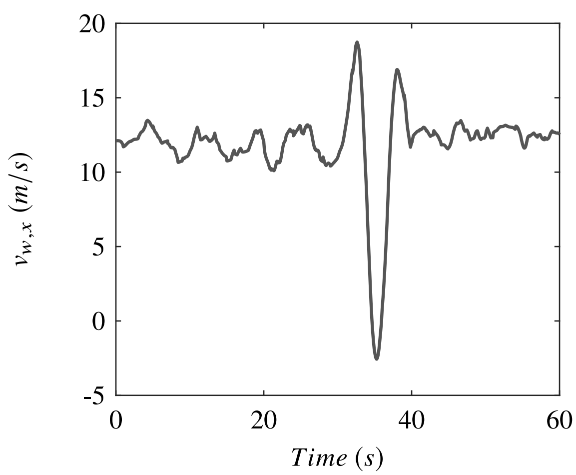

Fig. 32   Gust in upwind direction.

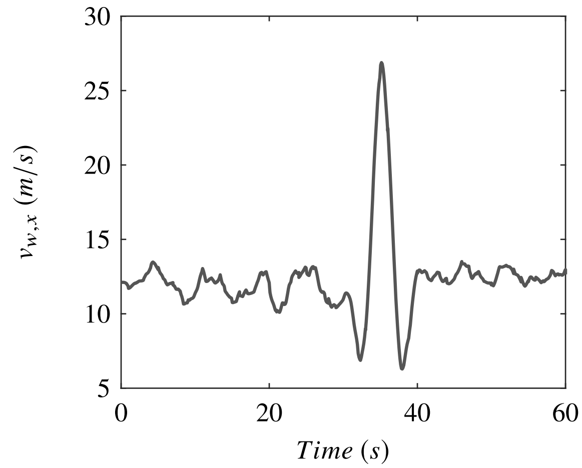

Fig. 33   Gust in downwind direction

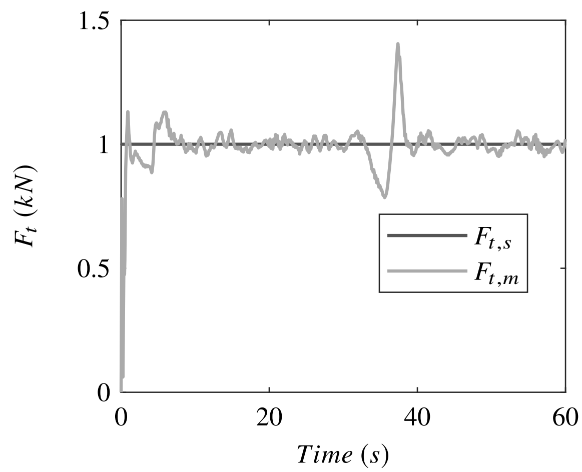

Fig. 34   Tether force with gust in upwind direction.

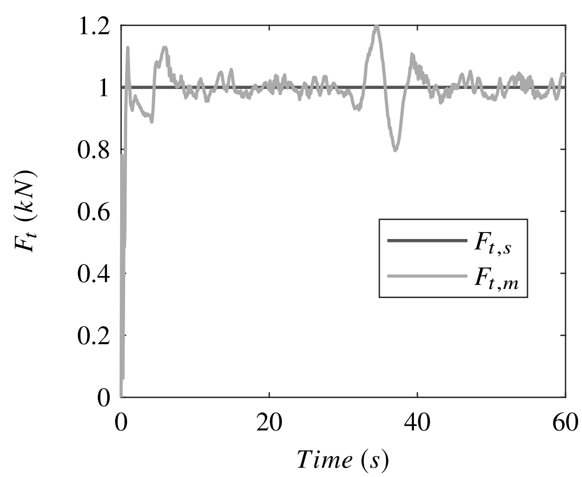

Fig. 35   Tether force with gust in downwind direction.

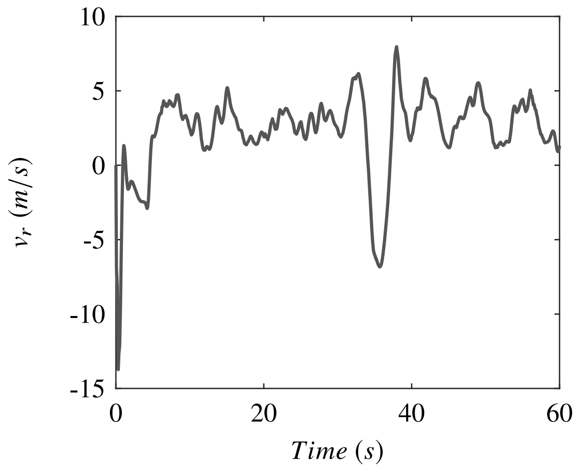

Fig. 36 v_r **with gust in upwind direction.**

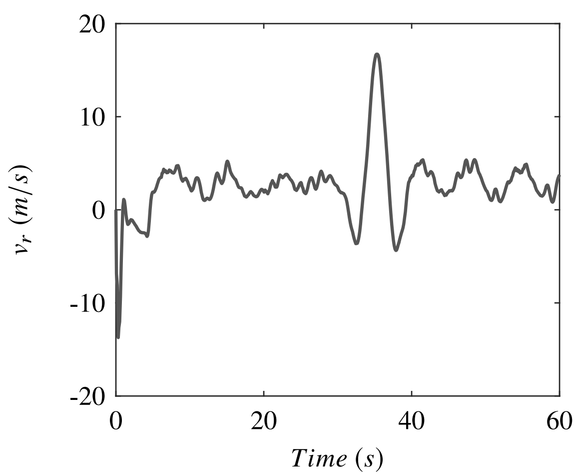

Fig. 37 v_r **with gust in downwind direction.**

flight path in radial direction. The flight path gets either compressed (Fig. 38) or stretched (Fig. 39) depending on the gust direction as a result of the increasing or decreasing reeling out velocity. Contrarily, Fig. 40 and Fig. 41 show that

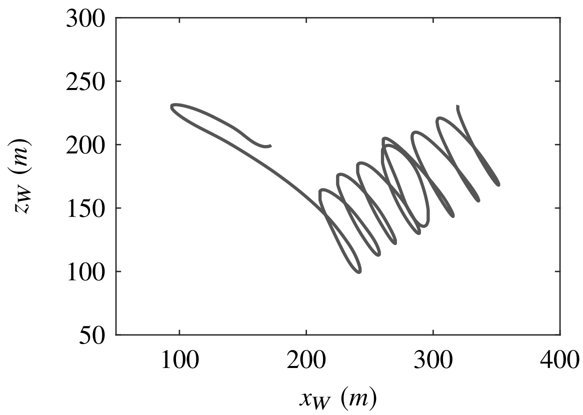

Fig. 38 Flight path with gust in upwind direction.

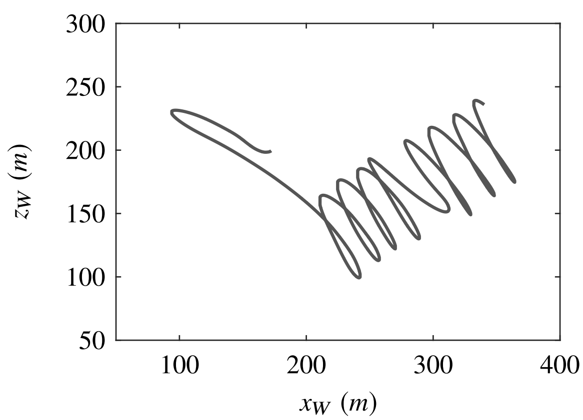

Fig. 39 Fight path with gust in downwind direction.

the adaptation of the reeling speed has only a small effect on the path-following performance in the tangential plane.

V. Conclusion

In this paper a novel cascaded model based control architecture for rigid-wing airborne wind energy systems operated in pumping-cycle mode has been presented. The proposed control approach leads to a robust control performance while flying in a realistic turbulent wind field. The extended geometric path-following approach guided the aircraft along a three dimensional curve reliably. State and input constraints are systematically handled using pseudo control hedging, which turns out to be beneficial especially during the retraction phase where the commanded flight path is adapted

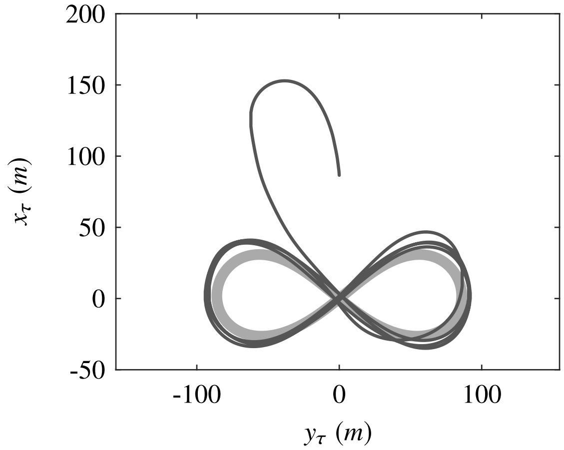

Fig. 40 Flight path with gust in upwind direction.

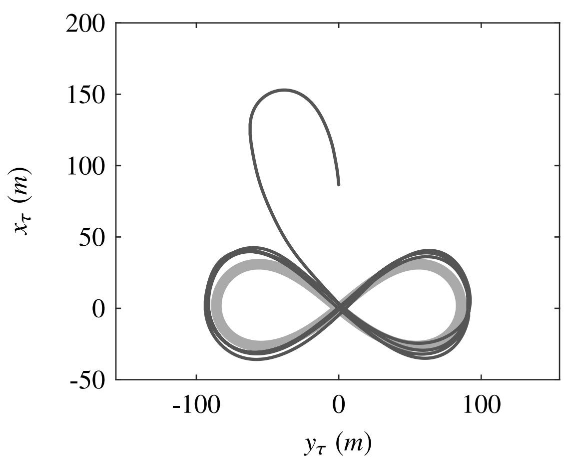

Fig. 41 Fight path with gust in downwind direction.

automatically in case of angle of attack saturation. Challenging phases during the pumping cycle are the transitions from the traction to the retraction phase and vice versa. Due to the rapid tether force changes in these phases, overshoots in sideslip angle and angle of attack are present although these peaks occurred only for a short period of time and the resulting tracking errors could be regulated back to the set point by the respective feedback controller. Moreover, the results show that the tether force set point can be tracked effectively by directly calculating a torque command as a function of the force tracking error. However, the excellent tether force tracking performance leads to a high variance in the reeling speed and therefore to oscillations in the mechanical power output. This effect could be reduced in the future by additionally using the pitch angle of the aircraft to control the airspeed. In return, this would lead to a less aggressive reeling speed adaption and hence a reduced variance of the mechanical power. In addition to the ability of tracking a constant tether force the proposed winch controller can react to sudden wind speed changes, such as gusts, through adaption of the reeling-out speed, effectively, ensuring the structural integrity of the aircraft.

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