Biomimetic optic flow sensing applied to a lunar landing scenario

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Abstract— Autonomous landing on unknown extraterrestrial bodies requires fast, noise-resistant motion processing to elicit appropriate steering commands. Flying insects excellently master visual motion sensing techniques to cope with highly parallel data at a low energy cost, using dedicated motion processing circuits. Results obtained in neurophysiological, behavioural, and biorobotic studies on insect flight control were used to safely land a spacecraft on the Moon in a simulated environment. ESA’s Advanced Concepts Team has identified autonomous lunar landing as a relevant situation for testing the potential applications of innovative bio-inspired visual guidance systems to space missions. Biomimetic optic flow-based strategies for controlling automatic landing were tested in a very realistic simulated Moon environment. Visual information was provided using the PANGU software program and used to regulate the optic flow generated during the landing of a two degrees of freedom spacecraft. The results of the simulation showed that a single elementary motion detector coupled to a regulator robustly controlled the autonomous descent and the approach of the simulated moonlander. “Low gate” located approximately 10 m above the ground was reached with acceptable vertical and horizontal speeds of 4 m/s and 5 m/s, respectively. It was also established that optic flow sensing methods can be used successfully to cope with temporary sensor blinding and poor lighting conditions.

I. INTRODUCTION

Automatic landing on extraterrestrial bodies is still an extremely hazardous and challenging task, even if the lander is equipped with the most advanced navigation and guidance systems. However, landing on the Moon, Mars and celestial bodies is often a vital part of scientific space missions. Entry, descent and landing on extraterrestrial bodies is controlled in various ways. Early missions used radar to control landing (e.g. Apollo 11). Newer approaches include lidar techniques [1], [2] and visual techniques [3], [4], [5], [6], [7], [8], [9] often supported with inertial measurements. In addition, vision-based navigation plays a key role when it is required to detect an extraterrestrial target from afar. Processing video-data fast enough to extract the requisite self motion data is a technically challenging task, especially in view of the tight aerospace restrictions involved in terms of the processing power, size and payload of the embedded electronics. Alternative solutions to this problem have been suggested by the neuronal and sensory systems of flying insects, which are able to navigate swiftly in unfamiliar environments by relying heavily on the angular speed of the images that sweep backward across their view-field [10], [11], which is known as the optic flow (OF) (Fig. 2) [12]. Bees, for example, use the OF to avoid obstacles [13], [14], to control their speed [15], [16] and height [17], [18], and to cruise and land [19], [18], [20]. These insects’ motion is sensed by Elementary Motion Detectors (EMDs), which process the OF by comparing the signals collected by adjacent visual sensors [21], [22]: the artificial EMD used here was a genuine airborne OF sensor.

The use of visual cues to guide spacecrafts’ extraterrestrial soft landing performances has been recently investigated by several authors [24], [25], [3]. These systems either use visually assisted inertial navigation systems [3] or compute the optic flow by means of an optical correlator [3], [9] or extract information from a single camera [25], [26]. By contrast, the autopilot described here extends the previously described EMD-based OCTAVE-autopilot principles [27], [28] to a lunar lander.

In the present study, the optic flow regulator has been completely redesigned to stabilize the Lander: a state-space representation including a state feedback and a nonlinear observer was used to cope with lander characteristics. In addition, the validity of our neuromorphic approach to landing control was tested with lunar images, and the ability of a simplified simulated lander to arrive safely at “low gate” during the
lunar approach phase was confirmed (Fig. 1). “Low gate” position was defined as the landing height at which direct visual contact with the landing site is obscured by the dust raised by the thrusters, for instance. Visual navigation is impossible from “low gate” onwards and alternative non-visual techniques are therefore required.

In the present setting, flight was controlled by keeping the OF perceived by the EMD as close as possible to a previously set point value. The EMD output signal was used directly to adjust the engine’s thrust, and hence the horizontal and vertical speeds. The autopilot then enabled the lander to safely decrease its horizontal speed, vertical speed, and height with a rough estimation of the initial height and vertical speed at high gate to initialize the non-linear observer. This estimation may be done by applying classical equations of the celestial mechanics. In the application in question, the autopilot also has to cope with:

- the natural instability of the lander,
- the large variations in altitude and speed due to the large working domain of application,
- the presence of many non-linearities in the lander.

The mechanisms involved in optic-flow processing are described in detail in the section II, and the autopilot control scheme is presented in the section III. Lastly, the results of the simulation experiments are given and discussed in the section IV.

II. EMD-BASED OPTIC FLOW MEASUREMENT USING PANGU

A. Pure translational Optic Flow (OF)

The simulated visual environment consisted of lunar surface images generated by PANGU, taking the position of the observer and that of the light source into account. The simulated lunar surface was not smooth and could include deep craters (up to 40m deep). The images generated by PANGU contained 256 gray-scale levels and had a resolution of 256 × 256 pixels; one square pixel corresponded to 0.1° × 0.1° in the field of view.

The OF sensor was mounted pointing downwards with respect to the simulated lander’s symmetry axis. This sensor consisted of two photoreceptors (i.e. two pixels) driving an Elementary Motion Detector (EMD). The visual axes of the two photoreceptors diverged by an inter-receptor angle \( \Delta \phi = 2^\circ \). The angular sensitivity of each photoreceptor obeyed a 2-D Gaussian function mimicking the angular sensitivity of a fly’s photoreceptor with an acceptance angle (the angular width at half height) \( \Delta \rho = 2^\circ \), as described by [30]. Each photoreceptor covered a field of view measuring 5° × 5°.

The photoreceptor output was computed at each simulation time step (1ms) by convolving the lunar surface image given by PANGU with the 2-D Gaussian filter. The simulated EMDs used spatio-temporal filtering steps as well as a contrast thresholding step to assess the OF [21], [22], [31].
Fig. 4. (A) Sketch of the OF-based lunar landing autopilot. The digital autopilot received the following inputs: the pitch angle \( \theta \) (given by an IMU), the measured OF \( \omega \) (given by an EMD) and the vertical acceleration \( a_{\text{lander, } z} \) (given by an accelerometer). In addition, the controller imposed the thrust level and the lander’s pitch. (B) The lander decreased or increased its forward thrust by pitching backward or forward, respectively. The vertical lift was directly affected by the pitch.

Fig. 3. EMD Input/Output characteristics resulting from the lunar image velocities generated by PANGU. Analog EMD output (in Volts) (red lines) versus the OF (i.e., an angular velocity in \(^{\circ}/\text{s}\)). This figure shows the EMD responses to several motion stimuli at various angular speeds. These data were collected using images produced by PANGU. The simulated circuit based on Franceschini’s time-of-travel EMD scheme delivered a monotonically increasing response (red plot) with respect to the angular velocity (see equation 3). In order to obtain these input/output response characteristics on the Moon, we applied ground speeds ranging from 50 to 300m/s to the lander at various altitudes (from 100 to 500m). Deviations from the theoretical functional characteristics (grey curve) obtained from (3) were mainly due to the lunar craters simulated by PANGU (which were up to 40m deep).

The working principles of the Franceschini’s time-of-travel EMD scheme used here were based on the results of studies on the fly’s eye, in which electrophysiological recordings were performed while light micro stimulation was applied to the retina [32]. The range of the EMD responses was calibrated by tuning the time constant \( \tau_{EMD} \) (see equation 3) of the decreasing exponential in order to be able to measure the exact OF generated by the lander. Depending on the speed and altitude of the lander during the landing phase, the OF therefore ranged between \( 10^{\circ}/\text{s} \) and \( 30^{\circ}/\text{s} \). Depending on the inter-receptor angle \( \Delta \phi \), the time constant \( \tau_{EMD} \) of the final low pass filter (figure 3 in [31]) was adjusted to \( \tau_{EMD} = 0.1 \)s. With these parameters, the OF sensor’s characteristics are shown in figure 3. The OF sensor’s response was a monotonically increasing function of the angular velocity with an order of magnitude ranging from \( 4.5^{\circ}/\text{s} \) to \( 45^{\circ}/\text{s} \), as described by the following equation:

\[
\omega_{\text{meas}} = k \cdot e^{-\frac{\Delta \phi}{\tau_{EMD}}}
\]

where \( k = 3.08 \)V, \( \Delta \phi = 2^{\circ} \), \( \tau_{EMD} = 0.1 \)s.

III. OPTIC FLOW REGULATOR

A. Dynamic model for a lunar lander

The autopilot presented here consisted mainly of a similar OF regulator to that previously described [27], [28] operating in the vertical plane \((x, z)\), which controlled the spacecraft’s mean thruster force. To stabilize the lander, it was necessary to cope with non-linearities and the inherent instability. Since there is no atmosphere on the Moon, no friction, wind or drag forces have to be dealt with. In the present model, the heave and surge dynamics were coupled via the lander’s pitch (Fig. 4B). The mean Force \( \overline{F} \) resulting from the thrusters can be expressed in terms of the vertical lift \( L \) (see Fig. 4B), as follows:

\[
L = F \cdot \cos \theta_{\text{pitch}}
\]

The thrusters could produce only positive forces and the maximum thrust was limited to \( 100 \cdot m_{\text{lander}} \) [N]. The transfer function \( G_{\text{thruster}}(s) \) describes the thruster dynamics between the mean thruster force and the thruster’s control input signal, as follows:

\[
G_{\text{thruster}}(s) = \frac{F(s)}{\text{Thruster}_{\text{cmd}}(s)} = \frac{1/\tau_{\text{thruster}}}{1/\tau_{\text{thruster}} + s}
\]

where \( \tau_{\text{thruster}} = 0.1 \)s.
we obtain the following equation:

\[ a_{\text{induced by the thruster}} = g_{\text{transmitted to the thruster engine}} \]

where \( g_l \) is the lunar gravity constant \((g_l = 1.63 \text{m/s}^2)\), \( m_{\text{lander}} = 10^3 \text{kg} \) is the lander’s mass and \( a_{\text{lander}x} \) is the lander’s acceleration in the lunar reference frame.

From equations 6 and 5, it is possible to obtain the transfer function describing the heave dynamics, i.e., the transfer between the altitude of the lander and the control signals transmitted to the thruster engine:

\[ G_z(s) = \frac{a_{\text{lander}x}(s)}{\text{Thruster cmd}(s)} = \frac{1}{s} \left( \frac{1/\tau_{\text{thruster}} - \cos \theta_{\text{pitch}}}{1/\tau_{\text{thruster}} + g_{\text{Moon}}} - g_{\text{Moon}} \right) \]

(7)

In the lander model, the following state vector was used:

\[ X = \begin{bmatrix} h \\ V_z \\ a_{\text{thruster}z} \end{bmatrix} \]

and the following input \( u = \left[ \frac{L}{m_{\text{lander}}} \right] \). Based on the equation 6, we can write:

\[
\begin{aligned}
\dot{a}_{\text{thruster}z} &= 1/\tau_{\text{thruster}} \cdot \left[ \frac{1}{m_{\text{lander}}} - a_{\text{thruster}z} \right] \\
\dot{V}_z &= a_{\text{thruster}z} - g_{\text{Moon}} \\
\dot{h} &= V_z
\end{aligned}
\]

(8)

where \( V_z \) is the lander’s linear speed along the \( z \) axis and \( a_{\text{thruster}z} \) is the thruster’s acceleration.

The state space matrix \( A_p \), \( B_p \) and a disturbance vector \( g \) can be deduced from the equation 8 as follows:

\[
\begin{bmatrix}
\dot{h} \\
\dot{V}_z \\
\dot{a}_{\text{thruster}z}
\end{bmatrix} =
\begin{bmatrix}
0 & 1 & 0 \\
0 & 0 & \frac{1}{\tau_{\text{thruster}}} \\
\frac{1}{\tau_{\text{thruster}}} & 0 & 0
\end{bmatrix}
\begin{bmatrix}
\dot{h} \\
\dot{V}_z \\
\dot{a}_{\text{thruster}z}
\end{bmatrix} +
\begin{bmatrix}
0 \\
0 \\
\frac{L}{m_{\text{lander}}}
\end{bmatrix}
- \begin{bmatrix}
0 \\
0 \\
g_{\text{Moon}}
\end{bmatrix}
\]

(9)

The present moonlander was modelled by the thruster dynamics and a pure double integration between the acceleration and the altitude, using the state-space approach.

The autopilot, which operated on the basis of a single OF measurement (that of the ventral OF), consisted of a visuomotor feedback loop driving the mean thruster force. The vertical lift and the forward thrust were coupled and the loop therefore controlled both the heave and surge axes. The pitch angle \( \theta_{\text{pitch}} \) was controlled by an external system: the lander pitched backwards from -60° to -30° while landing. The autopilot (Fig. 5) was composed of (i) a precompensation gain, (ii) a non linear state observer, and (iii) a state feedback gain. The non linear state observer estimated the state vector \( X \) on the basis of the ventral OF, \( \omega_{\text{meas}} \), the lander acceleration, \( a_{\text{lander}x} \), and the lander pitch, \( \theta_{\text{Pitch}} \). The complete regulator combined the estimated states with the full state feedback control loop.

B. Control law based on full state feedback

The autopilot kept the ventral OF of the simulated lander at the set point \( \omega_{\text{set}} \). This set point was compared with the product of the estimated state vector \( \hat{X} \) (Eq.11) and the state feedback gain \( L_{sf} \) to generate the thruster command. The
state feedback gain was calculated using the minimization criterion in the lqr method (Linear quadratic regulator), using the following matrix: \( A_{sf} = A_p, B_{sf} = B_p \) and \( C_{sf} = \begin{bmatrix} K_{lin} & 0 & 0 \\ 7.8 \times 10^{-4} & 0 & 0 \\ 0 & 0 & 0 \\ 0 & 0 & 0 \end{bmatrix} \) and the state-cost matrix \( Q_c = \begin{bmatrix} 1 \\ 0 & 0 & 0 \end{bmatrix} \) and \( R_c = [1] \). To compute the \( C_{sf} \) matrix, we linearized the expression for the OF near a set point. Here the set point was \( h_{lin} = 200m, V_{zi,n} = 50m/s \) and \( \omega = 14.3^\circ/s \). The OF was defined as an inverse function of \( h \). We therefore used the slope of the tangent to linearize the expression as follows:

\[
K_{lin} = \frac{V_{zi,n}}{\frac{d}{dh}(\frac{1}{h})}_{h=h_{lin}} = -\frac{V_{zi,n}}{h_{lin}^2}. \quad (10)
\]

C. Non linear state observer

Since the system is observable, a state observer for \( \hat{X} \) can be formulated as follows:

\[
\hat{X} = A_o \cdot \hat{X} + B_o \cdot u + K_o \cdot (y - \hat{y})
\]

\[
\hat{y} = C_o \cdot \hat{X} + D_o \cdot u \quad (12)
\]

where \( A_o = A_{sf}, B_o = B_{sf}, C_o = \begin{bmatrix} 0 & C_{sf} \\ 0 & 0 & 1 \end{bmatrix}, D_o = [0] \) and \( K_o \) (Observer gain) was also computed with the lqr method, using the \( A_o \) and the \( C_o \) matrix. As shown in figure 5, the estimator requires the value of the lander’s acceleration \( a_{lander_z} \).

To achieve an integral control, the augmented state vector \( \hat{X} \) was thus defined: \( \hat{X}_e = \begin{bmatrix} \hat{X} \\ d \end{bmatrix} \). The new state matrix could therefore be written as follows:

\[
A_{oe} = \begin{bmatrix} A_o & B_o \\ 0 & 0 & 0 & 0 \end{bmatrix},
\]

\[
B_{oe} = \begin{bmatrix} B_o \\ 0 \end{bmatrix},
\]

\[
C_{oe} = \begin{bmatrix} C_o & 0 \\ 0 \end{bmatrix}
\]

The new state feedback gain \( L_{sf} \) was equal to:

\[
L_{sf} = \begin{bmatrix} L_{sf} & 1 \end{bmatrix}
\]

D. Automatic landing example

To ensure a soft landing, the lander had to reach a distance of approximately 10 meters from the ground (i.e., the “low gate”) at a residual velocity of one meter per second in both the horizontal and vertical directions. Thanks to the biomimetic autopilot, the lander reached “low gate” with greatly reduced horizontal and vertical speeds approximately equal to the required values (Fig. 6). The lunar surface perceived by the lander consisted of gray-scale images generated by PANGU. In our simulation, we adopted an initial altitude \( Z_0 = 500m \), an initial ground speed \( V_{z0} = 150m/s \) and an initial vertical speed \( V_{z0} = -50m/s \). The pitch angle \( \theta_{pitch} \) decreased exponentially from \(-60^\circ\) to \(-30^\circ\), and the forward speed therefore decreased quasi-exponentially (Figure 6C), as did the vertical speed (Figure 6C), since its integral \( h \) was reduced quasi-exponentially to hold the measured OF \( \omega_{meas} = v_z / h \) around the set point value \( \omega_{set} \) (Figure 6D).

The spacecraft’s simulated approach took 58.4s, where \( t_l \) is the time required to reach “low gate”. The lander reached “low gate” at a final ground speed \( V_{zi} = 5m/s \) and a final vertical speed \( V_{zi} = -4m/s \); the distance travelled by the lander during the landing was 2600 meters. The final horizontal and vertical speeds are slightly higher than expected to strictly satisfy the speeds’ criterion at low gate (1m/s) in a near future.
IV. RESULTS

A. Influence of initial conditions

![Automatic landing from various initial altitudes. The simulation was initiated at altitudes of 750 m (green), 500 m (blue), and 250 m (red) under regular lighting conditions (initial ground speed 150 m/s, initial vertical speed −50 m/s). The lander is plotted every 20 s until it reached low gate. With an initial altitude of 250 m, the lander will probably crash, since “low gate” was reached after 8 s at a high ground speed of 77.4 m/s and a high vertical speed of −12.7 m/s. With an initial altitude of 500 m, the lander reached “low gate” in 60 s with a ground speed of 7.25 m/s and a vertical speed of −3.69 m/s. With an initial altitude of 750 m, the lander safely reached “low gate” in 70.7 s with a ground speed of 3.36 m/s and a vertical speed of −1 m/s. The influence of the initial altitude (with a given initial ground speed and vertical speed, \( V_{x0} = 150 \text{ m/s} \) and \( V_{z0} = −50 \text{ m/s} \), respectively) is shown in Figure 7. Since pitching reduced the forward speed, the lander automatically adjusted its thrust to maintain the ventral OF at the set point throughout the landing. With initial altitudes of 500 m and 750 m, the lander successfully reached “low gate” with an acceptable ground speed and vertical speed. In the case of a very low starting altitude (here: 250 m), the ventral OF was initially too high for the autopilot. It therefore reached “low gate” with a high ground speed \( V_{x1} = 77.4 \text{ m/s} \) and a high vertical speed \( V_{z1} = −12.7 \text{ m/s} \); these values were too high for safe landing to be possible.]

The fuel consumption criterion was defined as follows:

\[
\text{Consumption} = \int_0^{t_\text{f}} \text{Thruster}_{\text{cmd}}(t) \cdot m_{\text{Lander}} \text{dt}
\] (15)

Table I shows that a slow linear decrease in pitch (0.25 or 0.125 °/s) resulted in faster landing and an appropriate decrease in the vertical and horizontal speeds that reached a few meters per second at “low gate”. We are considering to introduce an additional breaking phase in order to reach final speeds of 1 m/s.

C. Low solar elevation

The landing trajectory was compared between two lighting conditions:

- nominal light condition with the sun elevation equals to 15°,
- low light condition with the sun elevation equals to 1.5° (south pole).

In our PANGU-based simulations, we observed that even under challenging lighting conditions (e.g. with a sun elevation of 1.5°), EMD-based OF processing results in successful landing (Fig. 8). It was concluded that even lighting conditions such as those encountered during lunar southpole landing might be compatible with neuromorphic technology. The \( \Omega_{\text{meas}} \) values recorded were similar to those obtained under nominal lighting conditions (Figure 8C and F). However, the EMD updating rate depends on the presence of contrast within the sensor’s range of sensitivity. Under very weak lighting conditions, the OF sensor output would not be updated because the contrast would not be detected by the sensor.

D. Temporary absence of OF measurements

According to our simulations, this system resists temporary blinding of the sensor (Fig. 9). Blinding may occur during the approach as the result of crossing a zone which does not provide any visual contrast, e.g. a valley, a crater or the shaded side of a hill. As long as this blinding is of limited duration (up to 6 s in this setup), the plant will successfully reach “low gate”; when it encounters a sensor blinding level of 9 s, the plant arrives at “low gate” at a speed which is too high to be able to land safely.

V. CONCLUSIONS

Here we have presented the first simulations in which neuromorphic principles have been applied to monitoring and processing the optic flow in an autonomous visual-based extraterrestrial landing scenario. In the present autopilot, biological principles such as motion extraction and OF regulation were used to land a simulated spacecraft in a lunar
TABLE I

COMPARISON BETWEEN LANDING PERFORMANCES WITH VARIOUS PITCHING LAWS.

<table>
<thead>
<tr>
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<td>τ = 20s</td>
<td>τ = 15s</td>
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<td>58.4</td>
<td>74.7</td>
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<td>4.8</td>
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<tr>
<td>Final Vz1 [m/s]</td>
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<td>-3.69</td>
<td>-6.43</td>
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<tr>
<td>Consumption/mlander</td>
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<td>200</td>
<td>222.7</td>
</tr>
<tr>
<td>Pitch at touch down [°]</td>
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<td>Typical</td>
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</table>

Fig. 8. Comparison between automatic landing under nominal and south-pole lighting conditions. (A) Lander trajectory under nominal lighting conditions and with an initial ground speed $V_{g0} = 150 m/s$, an initial vertical speed $V_{z0} = -50 m/s$ and an initial altitude $Z_0 = 500 m$ (cf. Fig. 6) (B) PANGU-generated lunar surface under nominal lighting conditions. (C) Output $\omega_{m \alpha \zeta}$ of the OF sensor monitored during the landing under nominal lighting conditions. The OF was relatively constant throughout, $\omega_{m \alpha \zeta} = 1 V(0.3 rad/s = 17.2°/s)$. (D) Landing trajectory obtained using the same initial settings but under lighting conditions resembling those pertaining at the lunar south pole, i.e. sun elevation at 1.5°. (E) PANGU generated lunar surface with sun elevation at 1.5°. The terrain was the same as that presented in Fig. 8C. (F) The OF measured $\omega_{m \alpha \zeta}$ remained the same as in (C) despite the challenging lighting conditions, which were similar to those pertaining at the south pole of the Moon. $\omega_{m \alpha \zeta} = 1 V(0.3 rad/s = 17.2°/s)$.

The main differences with previous visual landing methods are the low optical resolution, the lightweight optics consisting of only two pixels and the high temporal resolution of the visual processing system presented here. Thanks to the biomimetic electronic circuits such as EMD used here, all the computations were performed at a sampling frequency of 1kHz. In addition, the present autopilot monitored only the optic flow and thus regulated the spacecraft’s flight without any need to specify the speed and distance, and hence without any need for bulky, power-consuming sensors. The simulated lander navigated on the basis of only a few parameters: the pitch, the vertical acceleration, the ventral optic flow with a rough estimation of the initial height and vertical speed at high gate to initialize the non-linear observer. It was established that the OF regulator held the perceived OF close to a previously chosen set point by acting on the mean thrustor force. As a result, the lander’s ground speed and vertical speed decreased automatically during the landing phase due to the coupling between the heave and surge dynamics, and the lander therefore reached “low gate” at low speed.

The performances of the present controller were found to be robust despite the presence of various disturbances. Safe and soft landing occurred even at a low sun elevation of 1.5°, which is typical that occurring at landing sites near the Moon’s south pole. Temporary sensor blinding can be compensated for to a certain extent at this early stage of development. Variations in the initial flight conditions occurring during the trajectory can also be compensated for successfully (up to a point) by the system presented here. During all the simulation experiments, it is worth noting that PANGU generated a highly realistic Moon terrain, including valleys and craters with depths of up to ~40m.

As these first simulations nicely demonstrated the high potential of our bio-inspired approach, the next steps will focus on reducing again the approach speeds to “low gate” below 1m/s, and we will probably introduce an additional braking phase. Next to piloting the spacecraft, guidance and navigation issues need to be addressed. Therefore we suggest...
to enlarge the sensory field of view and address hazard detection and avoidance mechanisms based on EMD-technology. Further vibration tests should determine the robustness of a gimbal mounted OF sensor made of Franceschini’s time-of-travel EMD scheme. In addition, we suggest simulations with three degrees of freedom including a second feedback loop to control the lander’s pitch through optic flow. Finally we suggest to implement high level target localization features for point-to-point landing capability.

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REFERENCES