DYNAMIC STABILITY FLIGHT TESTS OF REMOTE SENSING MEASUREMENT CAPABLE AMPHIBIOUS UNMANNED AERIAL VEHICLE (A-UAV)

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ABSTRACT

Remote sensing techniques are widely used in earth sciences. Satellites and manned aircrafts are most common method for capturing remote sensing images. However, these techniques have some major disadvantages such as, high price, low image resolution, time restriction. Amphibious Unmanned Aerial Vehicle (A-UAV) is designed to integrate remote sensing measurement sensors to a Mini-UAV [Mutlu, 2012]. It can perform a 30 minutes of flight with 1 kg payload and 4 kg maximum take-off weight (MOTW). Main performance characteristics of A-UAV such as maximum velocity, rate-of-climb, and turn radius have been investigated [Mutlu, 2013]. This paper presents the dynamic stability tests of A-UAV. Sensor calibration curves are obtained before the tests via horseshoe and tower fly by methods. Dynamic stability tests focus on exciting longitudinal and lateral-directional modes -phugoid, short period; dutch roll, spiral and roll- of the A-UAV. Convergence characteristics of each mode are investigated during the tests.

AIR DATA CALIBRATION

Horseshoe Technique

Pitot-static calibration tests have been conducted to be sure about their functionalities and reliabilities. One of the methods for this purpose is “horseshoe heading technique” [David, 2001].

In this test method, the aircraft flies sequential constant headings at 90º intervals in any direction as long as they are 90° intervals (Figure 1). The method can be used at any altitude and the direction and magnitude of the wind does not need to be predetermined. During flight through these legs, aircraft heading at each leg, altitude and Indicated Air Speed are held constant and GPS data is collected. At least 3 legs should be flown to determine 3 unknowns of wind velocity component in North direction (V_N), in West direction (V_W) and magnitude of $|V_{wind}|$.

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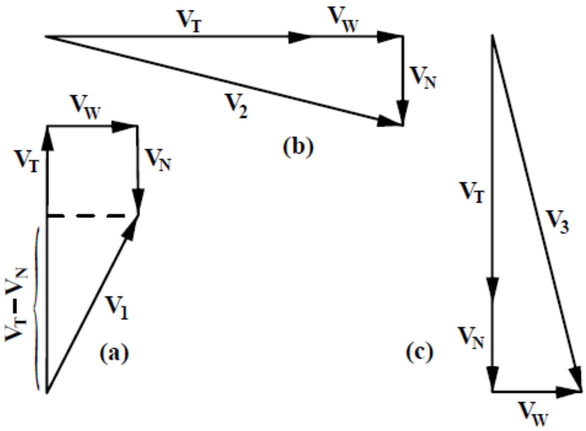

V_N: Wind velocity component in North direction
V_W: Wind velocity component in West direction
V_T: True Air Speed
V_1: Ground Velocity in Leg 1 obtained from GPS data
V_2: Ground Velocity in Leg 2 obtained from GPS data
V_3: Ground Velocity in Leg 3 obtained from GPS data

Figure 1 Horseshoe heading method velocity vectors [David, 2001]

Considering aforementioned information, flight tests are conducted and corresponding test results are shown in Table 1.

Table 1 Horseshoe Maneuver Test Results

Test #	V1 ground(kts)	V2 ground(kts)	V3 ground(kts)	Wind Angle(°)	V wind (kts)	TAS (kts)
1	56	50	39	-10	9	47

Having determined the True Air Speed (TAS) of the aircraft, thorough Equivalent Airspeed (EAS), Calibrated Airspeed (CAS) is calculated by Eq. 1 and Eq. 2, respectively.

$$EAS = \frac{TAS}{\sqrt{\frac{\rho_o}{\rho}}} \qquad Eq.1$$

$$CAS = EAS \cdot \left[1 + \frac{1}{8}\left(1 - \frac{P}{P_o}\right)M^2 + \frac{3}{640}\left(1 - 10\frac{P}{P_o} + 9\left(\frac{P}{P_o}\right)^2 M^4\right)\right] \qquad Eq.2$$

Where M is the Mach number, P is the static pressure and P_o is the total pressure read from the pitot tube.

The expectation is to have considerably same CAS and IAS values. Note that, CAS is measured theoretically whereas the IAS is read directly from the pitot tube. If these speeds are not the same, pitot system needs a calibration for static port position error.

Sample calculation steps for unique airspeed (19 m/s) are tabulated in Table 2.

Table 2 Horseshoe Maneuver Test Results for IAS=19 m/s

Test #	V wind (kts)	TAS (kts)	IAS (kts)	CAS (kts)	EAS (kts)	TAS (kts)	Calibration (kts)	Calibration (m/s)
1	9	47	39	44	44	47	5	2,6
2	11	49	39,5	46	46	49	7	3,3
3	9	47	37	44	44	47	7	3,6
4	10	47	39	44	44	47	5	2,6
5	12	50	40	47	47	50	7	3,6

Test Results	
Measured by Sensor	
Theoretical Values	

Calibration (m/s)	
Mean Value	3,1
Std. Dev.	0,5

In Table 2, deviation and mean value of test results conducted at 39 KIAS (19 m/s) are indicated.

In the end, CAS and IAS values are compared and errors due to pressure distribution over static port are examined. From test results, it is convenient to assign an offset of **3.1 m/s.** For the velocity of 39 KIAS (19 m/s), required calibration is decided to be **3.1 m/s**. Horseshoe maneuver is repeated for the several operational speeds. In the end, calibration curve (Figure 2) is obtained for operational velocities and the results are also tabulated in Table 3.

Table 3 Overall Test Results

Test Speed (IAS) (m/s)	Vwind (kts)	KTAS	KIAS	KCAS	KEAS	KTAS	Position Error (kts)	Calibration (m/s)
19	10	48	37	43	43	46	6	3.08
21	10	51	41	48	48	51	7	3.6
23.5	8	57	45	54	54	57	9	4.6

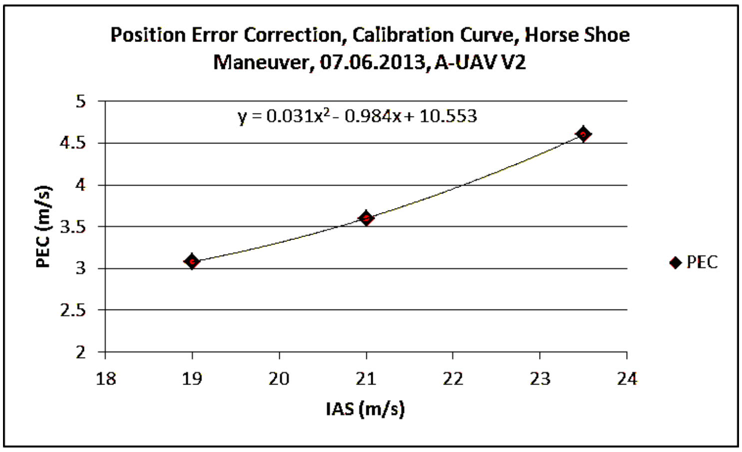

Figure 2 Calibration Curve

The calibration curve's best fit equation is given in Eq. 3:

$$\delta(x) = 0.031 \cdot IAS^2 - 0.984 \cdot IAS + 10.553 \quad (IAS\ in\ m/s) \qquad Eq.3$$

Position error of the pitot system is considerable high for the test platform. Offset value **of 3.5** m/s is required in order to have an **error of ±0.5 m/s** around **operational velocity interval (19-22 m/s).**

Tower Flyby Technique

The test technique is to fly the aircraft along a ground reference line, past the tower, in stabilized flight at a constant airspeed and at the approximate height of the tower (Figure 3) [Edward, 1995]. The primary piloting task is to maintain a constant indicated altitude during the run. The data recorded during each run are the indicated pressure altitude of the tower as it passes the tower.

Pressure altitude values are recorded to the aircraft data computer. Pressure at the level of flyby line is measured with flight test instruments. Aircraft constant altitude from the flyby line is measured by taking photos from the tower. At the end, two altitude values are compared.

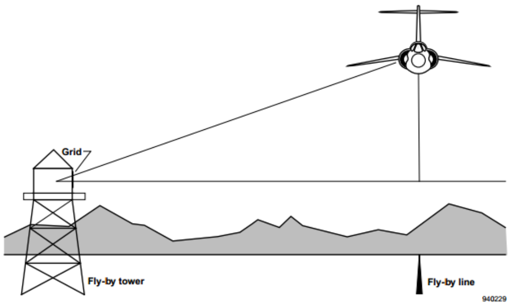

Figure 3 Tower Flyby Test [Edward, 1995]

Data reduction process is explained given in Eq. 4-Eq. 6 [Edward, 1995];

$$\Delta H = H_{exalt} - H_{calc}$$

$$\mathrm{H_{calc}} = \left(\Delta P \cdot 2{,}95 \cdot 10^{-4} + P_{ground}\right) \cdot 952 \qquad Eq.4$$

$$\Delta P = \rho_o \cdot g \cdot H_{pxl} \cdot \sigma \qquad Eq.5$$

where $\rho = \rho_o \cdot \sigma$

$$\sigma = \frac{P_{ground}/P_o}{T_{ground}/T_o} \qquad Eq.6$$

Multiply geometric altitude (h_{pxl}) with temperature ratio to obtain barometric altitude(H_{pxl}) .

$$H_{pxl} = h_{pxl} \cdot \frac{T_{std}}{T_{test}}$$

$$h_{pxl} = \left(\frac{Height\ in\ Pxl}{Aircraft\ Length\ in\ Pxl}\right) \cdot (Aircraft\ Length\ in\ Ft)$$

$H_{exalt}: Uncorrected\ Barometric\ Altitude\ in\ ft$
$H_{calc}: Barometric\ Altitude\ (in\ ft)\ Calculated\ wrt\ Standard\ Sea\ Level\ Pressure\ (29.92\ inHg)$
$\Delta P: Pressure\ difference\ between\ ground\ level\ and\ flight\ level\ r\ Pascal$
$P_{ground}: Pressure\ at\ ground\ level\ in\ inHg$
$\rho_o: sea\ level\ air\ density, 1.225\frac{kg}{m^3}$
$g: gravitational\ acceleration, 9.81\frac{m}{s^2}$
$H_{pxl}: barometric\ altitude\ calculated\ from\ photos\ via\ pixels$
$\sigma: density\ ratio$
$\rho: density\ at\ Flight\ level$
$P_o: Sea\ level\ pressure, 29.92\ inHg$
$T_o: Sea\ level\ temperature, 288.15\ {}^oK$
$T_{ground}: Ground\ level\ temperature\ in\ Kelvin$
$T_{std}: Standard\ day\ temperature\ at\ flight\ level, in\ Kelvin$
$T_{test}: Test\ day\ temperature\ at\ flight\ level, in\ Kelvin$
$h_{pxl}: geometric\ altitude\ calculated\ from\ photos\ via\ pixels$[5]

Flight tests for this maneuver have been completed. Following the data reduction process, explained previously, results are obtained and are shown in Figure 4.

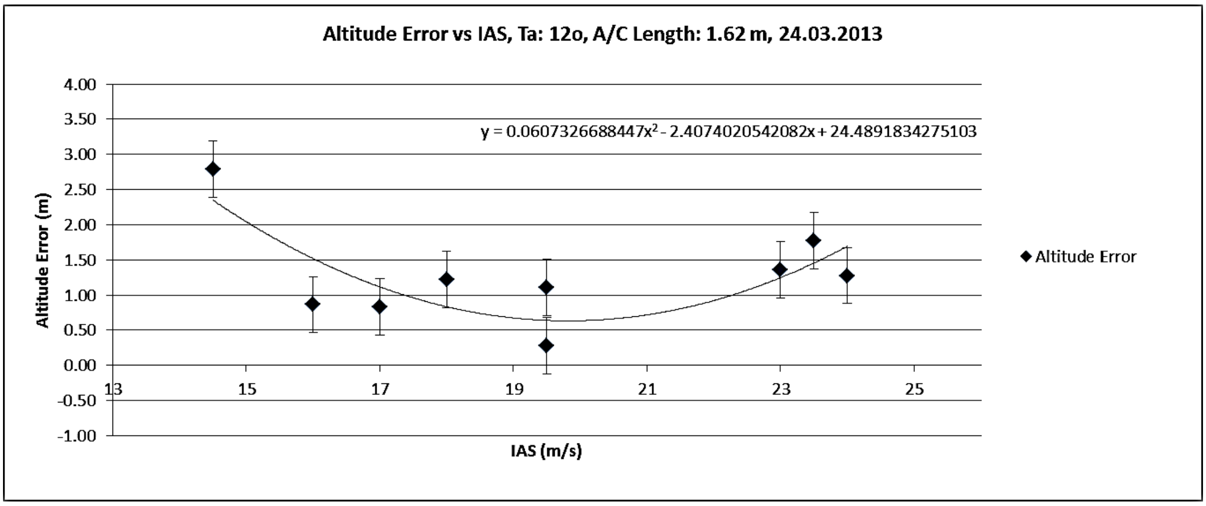

Figure 4 Tower Flyby Test Results, Calibration Curve

Altitude error of the test platform is almost negligible. Expectation is to have higher altitude errors at lower and higher speeds. Since pressure distribution over the aircraft is affected by higher angle of attack in lower airspeeds, altitude error increases. At higher airspeeds, turbulent effects and increased induced velocity result in higher altitude errors. Offset value of **1 meter** can be applied to the data reduction process in Autopilot system.

DYNAMIC STABILITY TESTS

The aim of the dynamic stability tests is to acquire information about dynamic behavior of the modes of the aircraft. In other words, for each mode, damping ratio and undamped natural frequency or time constant is to be determined.

Type of Test Inputs

The random excitations due to the atmospheric disturbances are simulated with the aid of the three basic test inputs (Figure 5).

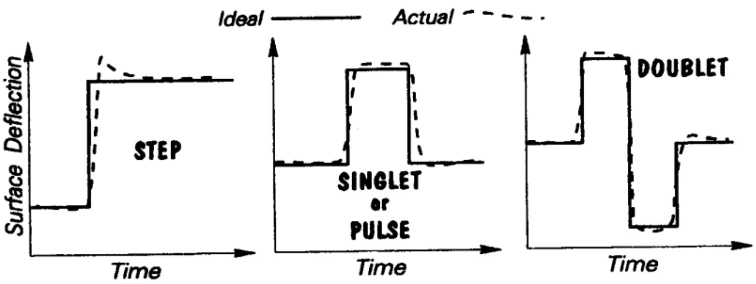

Figure 5 Control Inputs for Dynamic Testing [Edward, 1995]

Since control systems have inertia, ideal input is not achievable. Therefore, the surface deflection dynamics is like actual case in Figure 5.

Step Input: Control surfaces are deflected to the desired position and they are held that position.

Pulse Input: Pulse input is composed by two step inputs. First and second step input have same amplitude with opposite signs and application time difference.

Doublet Input: Doublet input is composed by two pulse inputs. Second pulse input is in the opposite sense of the first one.

Dynamic Longitudinal Tests

Phugoid Test Method: In this test, the aircraft is deviated from the equilibrium condition with varying the elevator angle provided that indicated airspeed does not increase or decrease as 5% of the trim airspeed. Then, elevator is returned to its original equilibrium position. In other words, pulse input is imitated in this way. Then, the transient characteristics of the pitch attitude, pitch rate, and airspeed are recorded without applying any longitudinal inputs. Test result is shown in Figure 6.

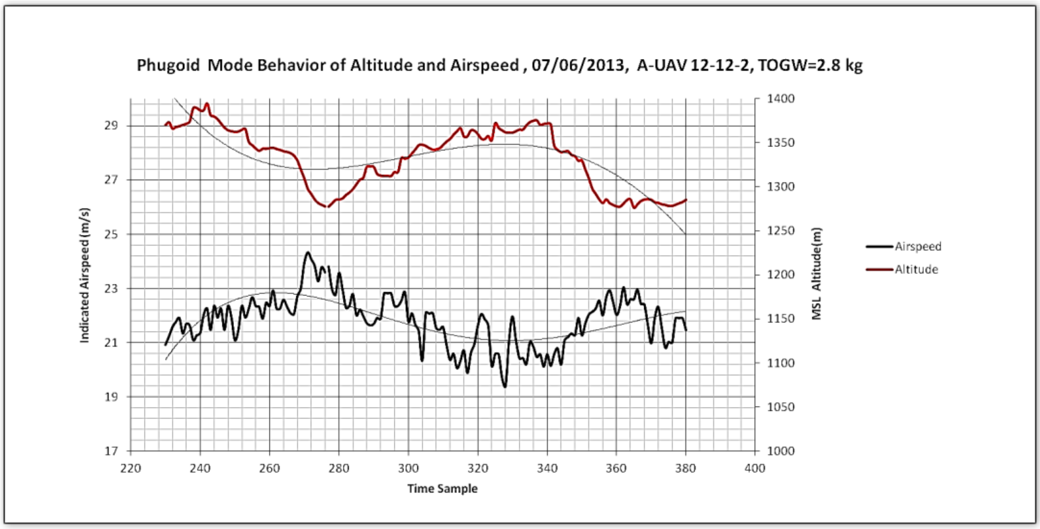

Figure 6 Phugoid Mode Behavior

In transient peak method, transient peak ratios are obtained as shown in Figure 7.

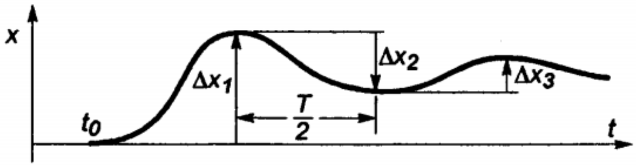

Figure 7 Transient Peak Method [Donald & Straganac, 2001]

Period of the phugoid mode (T) is measured as 7 sec.

Table 4 Transient Peak Ratios

$\Delta x_2/\Delta x_1$	$\Delta x_3/\Delta x_2$	Mean of the Ratios
0.75	0.67	0.71

From the mean of TPRs (table 4), damping ratio is obtained by using the figure 8.

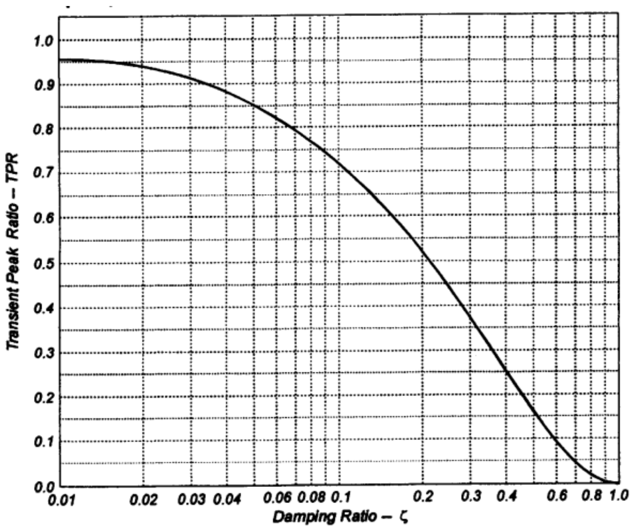

Figure 8 TPR vs. Damping Ratio [Donald & Straganac, 2001]

Natural frequency is obtained by using the following equation.

$$w_{n_p} = \frac{2\pi}{T\sqrt{1-\zeta_p^2}} \qquad [Donald\ \&\ Straganac, 2001]$$

Table 5 Damping and Natural Frequency of Phugoid Mode

ζ_p	w_{n_p}
0.1	0.9

The phugoid mode natural frequency and damping are not very low as shown in Table 5. This is due to the small inertia of the aircraft. Aircraft approaches its trimmed attitude very fast. This test requires very smooth weather. Therefore, repeating the test needs too much time. Since phugoid mode occurs fast, observing short period mode and roll mode seems not possible. Therefore, these tests will not be conducted.

Short Period Test Method: In this test, elevator is deflected so as to create doublet input. Elevator input is applied along about 2 seconds between -50 and +50 perturbations. After elevator is returned to trim point, longitudinal controls are released. Then, the transient responses of pitch attitude, pitch rate, and airspeed are recorded.

Considering the results obtained from the test, exciting phugoid mode which is quite slow compared to short period mode, it is considered that short period response of the aircraft cannot be observed. Therefore, this test is not applicable for the aircraft A-UAV.

Dynamic Lateral-Directional Tests

Dutch Roll Test Method: In this test, rudder input is applied along about 2 seconds between -50 and 50 perturbations. Like short period test method, doublet test signal is imitated. After rudder comes back to its equilibrium position, the lateral-directional controls are released. Then, the transient response of yaw rate is recorded.

Spiral Test Method: In this test, after the aircraft is trimmed at the test condition, the aircraft is rolled to reach a bank angle of approximately 100 and stabilized at that case. For free or fixed rudder and aileron case, whether the aircraft tends to roll out of the turn or not will be observed and roll rate is recorded.

Roll Test Method: In this test, roll rate is recorded by rolling the aircraft from 300 bank angle to -300 bank angle by applying constant aileron deflection.

Response to exciting phugoid mode indicates that the modes are too fast to be observed in such a low-inertia aircraft. Therefore, dynamic stability characteristics are determined theoretically.

Theoretical Results

Longitudinal Modes

Table 6

	Phugoid Mode	Short Period Mode
Poles	$-0.019 \pm 0.244j$	$-21.138 \pm -27.194j$
ω_n(rad/sec)	0.245	34.443
ζ	0.078	0.614
ω_d(rad/sec)	0.244	27.194
T(sec)	25.751	0.231

Both modes are oscillatory. As it is seen in Table 6, phugoid mode with low frequency is lightly damped. On the other hand, short-period mode with high frequency is heavily damped. Note that short period mode is much faster than phugoid mode.

Lateral Modes

Table 7

	Spiral Mode	Dutch Roll Mode	Roll Mode
Poles	0.288	$-0.456 \pm 2.326j$	-93.645
ω_n(rad/sec)	-	2.370	-
ζ	-	0.192	-
ω_d(rad/sec)	-	2.326	-
T(sec)	-	2.701	-
τ (sec)	-3.472	-	0.011

Only Dutch Roll mode is oscillatory among the lateral modes. Roll mode is the fastest; however, spiral mode is the slowest mode. Moreover, spiral mode is on the unstable region. In spite of this, pilot can recover because the time constant is sufficient.

CONCLUSION

Flight tests play important role to verify the capabilities of the aircraft in all manner; including its performance characteristics and stability characteristics. In this purpose, flight test program is applied to the Amphibious UAV as it consists of two main campaigns. First one is to verify the instruments and to make sure that the measurements are satisfactorily valid so that they can be used in analyses. Following these calibration process, stability tests are conducted. However, only exciting one of the slowest modes, phugoid mode, gives legitimate results. Considering required times to approach trim of all other modes are much lower than phugoid mode, even phugoid approaches trim quite fast, it is understood that stability tests apart from exciting phugoid mode are not applicable for such an aircraft. Main reason is apparently low inertia of the A-UAV. In this case, aircraft approaches to trim very fast even in slowest mode.

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