

Subsonic and Transonic Aerodynamic Stability Characterization of the Dragonfly Aeroshell

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NASA's Dragonfly mission, led by the Johns Hopkins University Applied Physics Laboratory, will deliver a relocatable lander to explore prebiotic chemistry and habitability of Saturn's moon, Titan. The Entry, Descent, and Landing (EDL) phase of the mission will have an Entry Vehicle (EV) system that will deliver a rotorcraft lander to an above surface release altitude. The EV system uses a supersonically deployed drogue parachute to decelerate and dynamically stabilize the aeroshell. A high-fidelity characterization of the dynamic stability of the aeroshell is needed to assure the EV system successfully completes the nearly 2-hour descent before releasing its heatshield. A 12.4%-scaled model of the aeroshell was tested in the NASA Langley Transonic Dynamics Tunnel, in air and R134a test mediums, using captive static and forced oscillation test techniques. It will be shown how the drogue design impacts test planning and execution. The objective of the test was to acquire the static and dynamic aerodynamic coefficients, at Titan flight similitude, that can be used in the Dragonfly 6-DOF Monte Carlo (MC) simulation studies. Metrics from the MC simulation studies are used to determine if the entry performance requirements are satisfied. The results show that the dynamic stability derivatives have large unstable values with nonlinear sensitivities to Mach, total angle of attack, and reduced frequency. In general, the data show that increasing the reduced frequency results in less unstable values of the dynamic stability derivatives which is beneficial to overall system performance.

Nomenclature

A	=	amplitude of oscillation
C_A	=	axial force coefficient
C_{DS}	=	parachute drag area
C_N	=	normal force coefficient
C_m	=	pitching moment coefficient
C_{m_q}	=	damping-in-pitch derivative, $\frac{\partial C_m}{\partial(q \cdot L_{ref}/2V_\infty)} + \frac{\partial C_m}{\partial(\dot{\alpha} \cdot L_{ref}/2V_\infty)}$, per radian
C_{m_α}	=	pitching moment coefficient derivative w.r.t. to angle of attack, pitch static stability
C_n	=	yawing moment coefficient
C_{n_r}	=	damping-in-yaw derivative, $\frac{\partial C_n}{\partial(r \cdot L_{ref}/2V_\infty)} - \frac{\partial C_n}{\partial(\dot{\beta} \cdot L_{ref}/2V_\infty)} \cos(\alpha)$, per radian
D	=	aeroshell maximum diameter
f	=	frequency, Hz
I_{yy}	=	mass moment of inertia about the body pitch axis
I_{zz}	=	mass moment of inertia about the body yaw axis
k	=	reduced frequency parameter, $\omega L_{ref}/2V_\infty$
L_{ref}	=	reference length
M	=	freestream Mach number
q	=	body-axis pitch rate

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$\bar{q}_\infty$	=	freestream dynamic pressure
$\hat{q}$	=	non-dimensional pitch rate, $(q + \dot{\alpha})/2 \cdot L_{ref}/(2V_\infty)$
$\hat{r}$	=	non-dimensional yaw rate, $(r - \dot{\beta})/2 \cdot L_{ref}/(2V_\infty)$
r	=	body-axis yaw rate
Re_D	=	Reynolds number based on the reference length, $V_\infty L_{ref}/\nu$
S_{ref}	=	reference area, $\pi L_{ref}^2/4$
t	=	time
V_∞	=	freestream velocity
X_r	=	distance from CG to triple bridge confluence point
α	=	angle of attack
$\dot{\alpha}$	=	time rate of change of angle of attack
α_T	=	total angle of attack
β	=	angle of sideslip
$\dot{\beta}$	=	time rate of change of angle of sideslip
ν	=	kinematic viscosity
ω	=	oscillatory frequency of the aeroshell about a body-axis ($2\pi f$), rad/s

Acronyms and Abbreviations

ADB	=	Aero Database
ADT	=	Angular Displacement Transducer
AMS	=	Angle Measurement System
BMC	=	Balance Moment Center
BR	=	Ballistic Range
CG	=	Center of Gravity
CONOPS	=	Concept of Operations
Csys	=	Coordinate System
DAS	=	Data Acquisition System
DOF	=	Degrees of Freedom
EDL	=	Entry, Descent, and Landing
EOM	=	Equation of Motion
EV	=	Entry Vehicle
FO	=	Forced Oscillation
LaRC	=	Langley Research Center
MC	=	Monte Carlo
MRC	=	Moment Reference Center
OML	=	Outer Mold Line
OTT	=	Oscillating Turn Table
POST2	=	Program to Optimize Simulated Trajectories II
PPF	=	Preparation for Powered Flight
TDT	=	Transonic Dynamics Tunnel
12FT LST	=	NASA LaRC 12FT Low-Speed Tunnel

I. Introduction

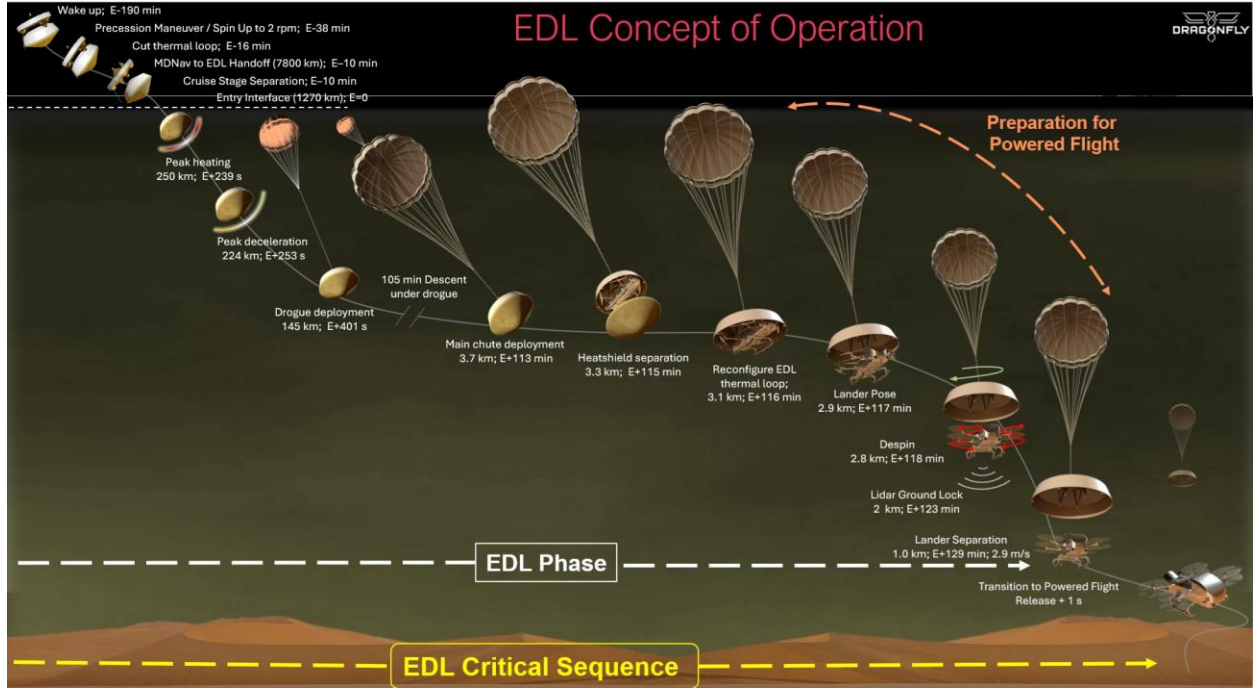


Fig. 1 Dragonfly EDL Concept of Operations on Titan.



Fig. 2 Artist concept showing near-surface and surface operations of the Dragonfly rotorcraft

NASA's Dragonfly mission will deliver a rotorcraft (Figs. 1 and 2) to Saturn's largest moon, Titan, to enable exploration and search for building blocks of life. An aeroshell will deliver the rotorcraft to an above surface release altitude [1]. Details of the Entry, Descent, and Landing (EDL) concept of operations (CONOPS) are given in Ref. [1]. Due to Titan's unusually tall (~1,300 km) and dense atmosphere the duration of the EDL is over two hours. Of those two hours, approximately 105 minutes will be spent descending under a drogue parachute at Mach numbers less than 1.5. The Dragonfly aeroshell shape and center of gravity (CG) location closely matches the Genesis sample return capsule. This capsule is known to be dynamically unstable from ballistic range tests [2] and the Earth EDL flight reconstruction [3]. The flight reconstruction estimated that after the stabilizing drogue failed to deploy at $M = 1.8$ the onset of tumble occurred at $M = 0.9$, with tumble at $M = 0.6$. Even if tumble is prevented, the successful release of the Dragonfly rotorcraft requires the aeroshell to be within attitude and angular rate limits that are

significantly more restrictive. The long flight duration allows time for dynamics, resulting from atmospheric disturbances to the aeroshell, to grow significantly if the aeroshell is not sufficiently stabilized. The drogue must be designed to provide this sufficient stability [4, 5, 6]. The need to deploy supersonically, Titan's tall atmosphere, and other design constraints specific to EDL on Titan restricts the range of drogue sizing afforded by Earth and Mars EDL. Additionally, the dynamic aero data for the Dragonfly aeroshell geometry was severely lacking for $M < 1.1$ because the Genesis capsule, for Earth EDL, did not require an extensive set of dynamic aero in this flight regime. For these reasons, a comprehensive characterization of the aerodynamic stability of the aeroshell was paramount to assure the drogue would provide the requisite dynamic stability for a successful Titan EDL. To this end, a subsonic and transonic aerodynamic stability characterization of the aeroshell was planned and executed in the NASA Langley Transonic

Dynamics Tunnel (TDT) using captive static and dynamic test techniques on a 12.42%-scaled model. The test was executed with several tunnel entries from June 2022 to February 2023. The TDT was chosen because it has the test apparatus and flow conditions necessary for obtaining static and dynamic aerodynamic data at flight similitude on Titan. The test was conducted in air and R134a test mediums. The R134a test medium was only required to match the similitude parameters for the $M = 0.2$ to heatshield release flight regime. Furthermore, to provide optimum test efficiency, the apparatus is mechanized so that the model could be set to any angle-of-attack from 0 to 360° and switched between the static and dynamic test techniques without a tunnel entry.

Currently, the dynamic aero data for supersonic Mach numbers are from Genesis Ballistic Range (BR) tests but will be replaced by Dragonfly specific BR tests. These BR tests will be conducted at Aberdeen Proving Grounds in early 2026. The final EDL segment is called the Preparation for Powered Flight (PPF) phase, for which the static aerodynamics were obtained by a combination of wind tunnel tests [7] and CFD solutions. The dynamic aero for the PPF flight phase was obtained using the forced oscillation technique in the NASA LaRC 20FT Vertical Spin Tunnel (VST) [8]. The TDT test will be discussed in this paper by describing the TDT facility and methods, the test parameter space, and data analysis to determine the functional dependence and uncertainties of the aeroshell's aerodynamic coefficients. The aero coefficients are incorporated in the Dragonfly 6-DOF Aero Database (ADB) [9]. This 6-DOF ADB is used in the Program to Optimize Simulated Trajectories II (POST2) 6-DOF Monte Carlo (MC) simulations. The simulation results are used to determine if the EDL requirements are satisfied. Examples of these results are given in Ref. [9, 10].

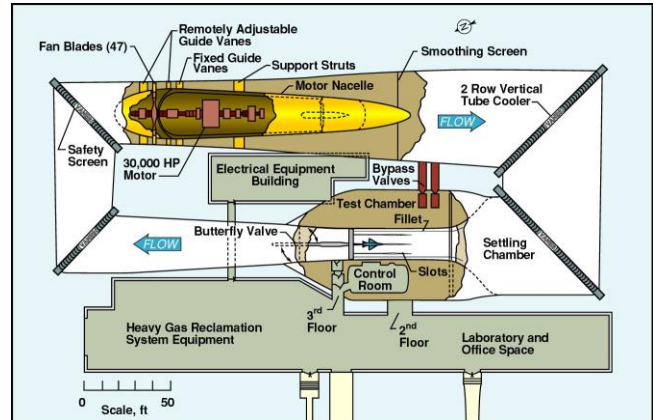


Fig. 3 Schematic showing the features of the NASA LaRC Transonic Dynamics Tunnel.

II. Test General Information

A. Test Facility and Test Hardware Description

The Dragonfly aeroshell tests were conducted in the TDT because it has the flow conditions and large test section cross area required for dynamic test techniques. Additionally, the TDT has a test apparatus that was designed to obtain rigid body static and dynamic aerodynamic data on blunt bodies.

The TDT is a closed-circuit, continuous flow, sub-atmospheric pressure wind tunnel with a 4.877-m (16-ft) square test section with cropped corners and slotted walls. The tunnel can use either air or heavy gas (R-134a) as the test medium. The tunnel can operate up to a Mach number of 1.2 and is capable of maximum Reynolds numbers of about 9.84×10^6 per meter (3×10^6 per foot) in air and 32.8×10^6 per meter (10×10^6 per foot) in R-134a. The tunnel operates at stagnation pressures from near vacuum (0.025 atmospheres) to atmospheric, and at dynamic pressures up to 15.8 kPa in air and 26.3 kPa in R-134a [11]. A schematic of the tunnel is shown in Fig. 3.

The test apparatus was designed for the forced and free-to-oscillate techniques by the NASA Orion program. The Orion Aerosciences team used these techniques to conduct dynamic stability tests [12], [13] on the Orion Crew Module and Launch Abort Vehicles. Since then, these test techniques have been used extensively on various blunt bodies, for example, the Boeing CST-100 [14], 45-, 52.5-, and 60-deg sphere-cone configurations of the NASA's Mars Sample Return Earth Entry Vehicle, Blue Origin Crew Capsule, and the SpaceX Dragon 2 capsule for the NASA Engineering and Safety Center (NESC) [15]. The rig incorporates a transverse sting mount that is driven by a hydraulic rotary actuator

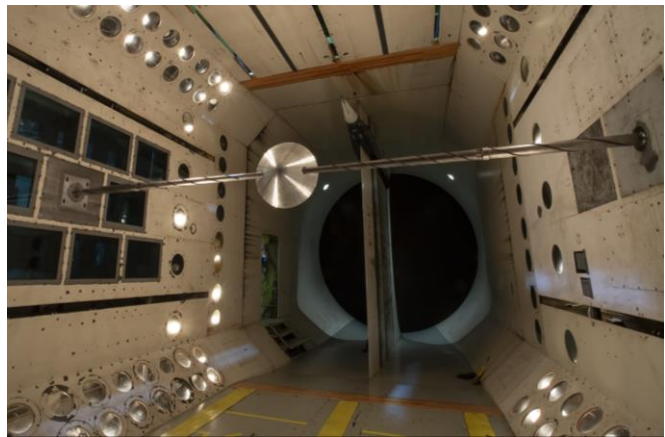


Fig. 4 The 12.42%-scale aeroshell mounted to the transverse sting in the NASA LaRC Transonic Dynamics Tunnel. View is looking downstream.

and a clutch/brake system that allows the model to be set to any angle in the 0-to-360-degree range (Fig. 4). Note, the transverse sting mount allows sweeping through only α with $\beta = 0^\circ$ or vice versa. Transition between test techniques is done seamlessly from the control room by simply changing the way the sting motion is commanded by the controller inputs to the hydraulic actuator. The controller inputs are arbitrary (e.g., pitch-pause, oscillatory, constant rate, etc.) and only limited by the angular acceleration, rate (max. ~ 320 deg/s), and position range (max. 90°) of the actuator.

B. Full-Scale Geometry, Mass Properties, Coordinate System and Reference Information

The geometry, mass properties, and aerodynamic data are defined in the body-axis coordinate system as shown in Fig. 5. The OML is axisymmetric about the X-axis. The maximum diameter, D , of the aeroshell is used as the reference length, $L_{ref} = 4.50$ m, and the area of a circle with diameter D defines the reference area, $S_{ref} = 15.90$ m². The aeroshell center-of-gravity (CG) used in the test is the Dragonfly program's reference CG location: $X_{cg}/D = 0.343$ from the nose of the heatshield, $Y_{cg}/D = 0$, $Z_{cg}/D = 0$ from the centerline. At this CG, the mass moments of inertia about the pitch and yaw body-axis are $I_{yy} = 3589$ kg-m² and $I_{zz} = 2855$ kg-m², respectively.

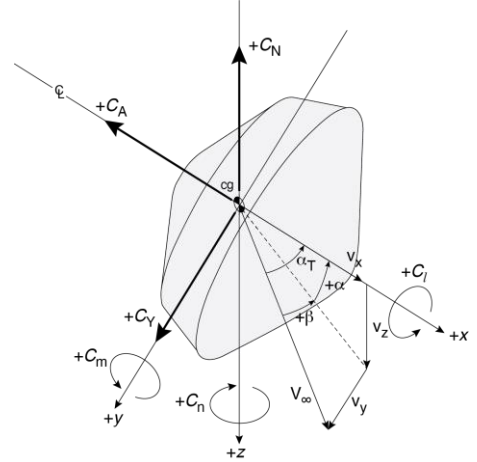


Fig. 5 Aerodynamic coordinate system (Csys).

C. Wind Tunnel Model Description

Various views of the 12.43%-scale aeroshell model mounted to the transverse sting are shown in Fig. 6. The picture on the left in Fig. 6 shows the sting entry point. The sting-to-model diameter ratio is 0.09. The picture on the right in Fig. 6 shows the model at $\alpha < 0^\circ$ with the wind from the right. A side-view drawing of the aeroshell, Fig. 7, shows the pertinent OML geometric features and location of the centerline CG. Namely, the aeroshell has a 60-deg sphere-cone heatshield and bi-conic backshell with angles of 20 and 61.6 deg.

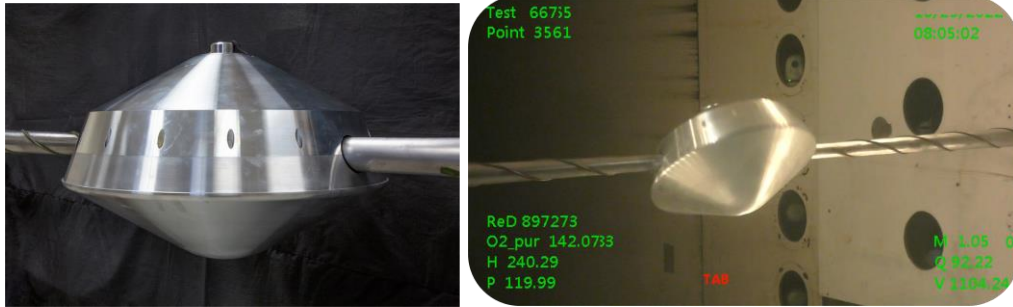


Fig. 6 Various views of the 12.43%-scale aeroshell mounted to the transverse sting along the pitch axis at $X_{cg}/D = 0.343$, $Z_{cg}/D = 0$.

D. Instrumentation and Data Acquisition System

1. Instrumentation

Forces and moments were measured with the NASA LaRC 1631B 6-component strain-gage balance mounted in the model such that the balance moment center (BMC) was coincident with the center of the rotation of the sting and the aeroshell's moment reference center shown in Fig. 5. Therefore, no moment transfers were required in the data reduction. An Angular Displacement Transducer (ADT), mounted in line with the center of rotation at the free end of the sting, was used to measure angle of attack of the model. A stand-alone, calibrated Angle Measurement System (AMS) was used as the reference for setting model zero attitude and for verifying the ADT. The ADT signal was numerically differentiated during data reduction to obtain the body-axis angular rate required for calculating the

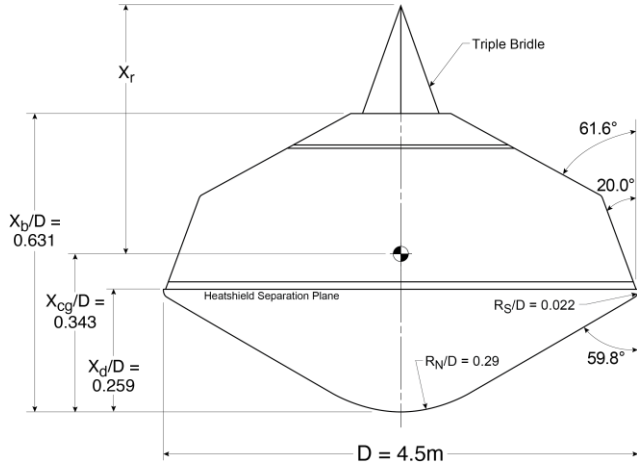


Fig. 7 Sideview drawing showing the pertinent full-scale OML geometric features including the location of the CG and Triple Bridle.

The sample time was based on test technique. The tunnel conditions like Mach, freestream velocity, etc., were acquired from the TDT DAS system then assimilated with the NEFF™ data.

III. Aerodynamic Modeling

The Dragonfly program uses statistical metrics computed from the POST2 6-DOF MC simulation results to determine if design requirements are satisfied. The flow conditions and other independent parameters, along with their ranges, were chosen to provide the requisite data, commensurate with the assumptions of the equations of motion (EOM) used in POST2, to model the aerodynamics. The EOM in POST2 assume the vehicle state can be determined by locally linear instantaneous forces and moments. The total aerodynamic forces and moments are determined from a Taylor series expansion evaluated at each time step. As an example, Eq. 1 shows the Taylor series expansion for the pitching moment coefficient truncating the series after the first order terms. Typically, only first order terms of the expansion are enough to provide an accurate aerodynamic model. The aerodynamic coefficients, shown in Eq. 1, are locally linearized, however, they can have a nonlinear functional dependence on the independent variables.

$$C_m = C_{m_0} + C_{m_\alpha} \alpha + \frac{qD}{2V} C_{m_q} + \frac{\dot{\alpha}D}{2V} C_{m_{\dot{\alpha}}} \quad (1)$$

The TDT tests obtained the aero coefficients on a geometrically scaled axisymmetric OML with a centerline CG and conditions that matched the similitude parameters for flight on Titan [17]. For $M < 1.1$, the similitude parameters for the static aero coefficients are α , β , Mach, and Reynolds number. In addition to these parameters, the oscillatory dynamic stability derivatives, C_{m_q} and C_{n_r} , must match the reduced frequency, $k = \omega L_{ref}/(2V_\infty)$, and non-dimensional angular rate parameters $\hat{q} = (q + \dot{\alpha})/2 \cdot L_{ref}/(2V_\infty)$ and $\hat{r} = (r - \dot{\beta})/2 \cdot L_{ref}/(2V_\infty)$. The TDT test matched flight similitude except for Reynolds number at $M < 0.2$. An attempt was made to ascertain the effect of Reynolds number on the static and dynamic aerodynamics at $M = 0.2$, but not enough data were collected to draw a conclusion. Also, the test support apparatus only allows setting the model at a single wind angle, either α at $\beta = 0^\circ$ or β , at $\alpha = 0^\circ$. Therefore, the static and dynamic aerodynamic coefficients are modeled as a function of the total angle of attack, α_T . This wind angle is sufficient for the static aero given the axisymmetric OML with a centerline CG. The implications to the dynamic aero will be discussed in section IV.

IV. Experimental Methods

A. Method for Static Aerodynamics

The static aerodynamic coefficients were measured using the pitch-pause and continuous sweep methods. The pitch-pause nominal sample time was 5 seconds as determined from a balance signal analysis during the first few runs.

damping derivatives. This derived angular rate was shown to have the requisite accuracy by comparison to a precision rate gyro mounted in the model during pre-test activities. The uncertainties due to instrumentation errors were calculated based on the root-sum-square (RSS) formula for combining component error in overall system-accuracy calculations [16]. Uncertainty in balance readings, angle of attack, velocity, and dynamic pressure were accounted for in these calculations. The maximum percent error of reading is on the order of 0.6%, but more typical values are less than 0.05%.

2. Data Acquisition System

The analog data were acquired from the instruments using the NEFF™ 620 system. The analog signals were conditioned using a 6-pole Butterworth low-pass analog filter with a 50 Hz cut-off frequency. All data channels were acquired using NEFF™ sample-and-hold cards and identically filtered to eliminate any phase lag between channels. The sample rate for the forced oscillation and static data was 1000 samples per second.

The aerodynamic coefficients are computed from a time average of these signals. Since the model is rotated about the pitch axis, the angle set by rotating the sting is α and, the static data are given as aerodynamic coefficients C_A , C_N , and C_m .

B. Method for Dynamic Aerodynamics

The dynamic aerodynamic coefficients were determined from the FO technique. The FO method consists of sinusoidally oscillating the model at a given frequency and amplitude about each angle of attack where the damping derivative is to be determined. The final data are conditioned using digital filters tuned to eliminate signal content from non-aerodynamic sources. For example, model specific sting structural modes, determined with in situ measurements, that are harmonics of the model oscillation frequency. Details of the data processing used in the FO technique are given in Ref. [18], but some will be discussed here for convenience and clarity.

The time history data from oscillating the model for several cycles (typically 40) are parsed into individual cycles then ensemble averaged. This average is called a mean cycle. The hysteresis of the forces and moments during the cycle represents the dynamic aero. The dynamic aero coefficients shown in this paper, and implemented in the POST2 ADB, were determined from the mean cycle using the specific point method applied at the midpoint of the cycle. This method calculates the dynamic aero derivatives as given in Eq. 2, using C_{m_q} as an example. Subscripts 1 and 2 represent value of the pitching moment signal when the model is at the minimum and maximum rate, respectively. These points coincide with the midpoint of the oscillation cycle which is the angle of attack where C_{m_q} is desired.

$$\overline{C_{m_q}} = C_{m_\alpha} + C_{m_q} = \frac{C_{m_2} - C_{m_1}}{\hat{q}_2 - \hat{q}_1} \quad (2)$$

To simplify notation, the $\overline{C_{m_q}}$ in Eq. 2 will be replaced by C_{m_q} for the remainder of this paper.

In the Aerodynamic Modeling section, it was shown that k and $\hat{q}$ are similitude parameters for aerodynamics resulting from oscillatory motion. Therefore, k and $\hat{q}$ are FO test parameters. These parameters can be varied by changing the frequency and/or amplitude of the model motion. Within the test apparatus limits, the frequency and amplitude can be changed independently of each other. To study the independent effects of k and $\hat{q}$ consideration must be given on the choice of model oscillation frequency and amplitude. For example, k can be changed simply by changing the model oscillation frequency while holding amplitude constant. For sinusoidal motion, the angular rate is given by $q = A\omega\cos(\omega t)$. This means that just changing frequency without changing amplitude also changes $\hat{q}$. Therefore, to study k independently of $\hat{q}$ requires that the amplitude must also change. The effects of $\hat{q}$ can be independently studied by varying amplitude without changing frequency.

The C_{m_q} calculated by Eq. 2 and the assumption of instantaneous aerodynamics implies C_{m_q} is temporally and spatially invariant. In fact, it has been shown [12, 19] that for certain flight conditions and blunt body geometries that C_{m_q} can depend on the aeroshell's direction of rotation (i.e. sign of $\hat{q}$). This happens when the α -range traveled during the motion transitions significant flow topology gradients, for example, when the stagnation point is oscillating near the shoulder. The pitching moment hysteresis data for each cycle can be used to ascertain these effects.

As mentioned in the Aerodynamic Modeling section, the test support apparatus only allows setting the model at a single wind angle, either α at $\beta = 0^\circ$ or β , at $\alpha = 0^\circ$. The ADB models the dynamic aero with damping derivatives defined with respect to the body-axis. In this test, the model was mounted to the sting on its pitch axis. Therefore, the damping derivative given in this paper is C_{m_q} as a function of α at $\beta = 0^\circ$, $C_{m_q}(\alpha, \beta = 0^\circ)$, and, theoretically speaking, only valid for motion in-plane to the pitch axis. Since the data are acquired for an axisymmetric geometry with a centerline CG ($Y_{cg} = Z_{cg} = 0$) then $C_{n_r}(\alpha = 0^\circ, \beta) = C_{m_q}(\alpha, \beta = 0^\circ)$. Unlike the static aerodynamics, the damping derivatives, in general, cannot be defined as functions of α_T , as was demonstrated by forced oscillation tests conducted in the NASA LaRC 12FT Low-Speed Tunnel (12FT LST) using a 7.78%-scale model of the Dragonfly aeroshell. The 12FT LST test rig can set α and β independently. The dynamic aerodynamics for the aeroshell were measured at α and β values in the range: $\pm 90^\circ$. Those results show that, for a given α , C_{m_q} is nearly constant for $\beta < 10^\circ$ but varies significantly for $\beta > 20^\circ$. This indicates that $C_{m_q}(\alpha, \beta = 0^\circ)$ data can be used to approximate $C_{m_q}(\alpha, \beta > 0^\circ)$, for small β and $C_{n_r}(\alpha = 0^\circ, \beta)$ is valid for small α . Unfortunately, the modeling of these effects has only been measured at incompressible flow conditions so extrapolation to compressible flow is difficult. The 12FT test results are mentioned here so the reader is aware that C_{m_q} and C_{n_r} as functions of α_T are probably not valid for significant non-planar dynamics (e.g., coning motion at a large α_T).

V. Test Parameter Space

This section describes the steps and information used to define the test matrix that provides the requisite data for the 6-DOF aero model implemented in the aero database (ADB) for the POST2 MC studies. The ADB used in POST2 constructs the aerodynamic model from a nominal and uncertainty data set. The test matrix is designed so that the nominal and dispersed values of the mass properties and flight conditions will be represented in those ADB data sets. Since these runs don't represent the final design, additional parameter space is added based on known sensitivities of the aerodynamics to these parameters. These effects can then either be modeled in the nominal data as a functional dependence or covered in the uncertainty data. At the time of the test, POST2 was modeling the static and dynamic aero coefficients with uniform uncertainties. In MC studies, accurate knowledge of uniform uncertainty bounds is extremely important since each value within the bounds is treated with equal weight. Therefore, knowing the uncertainty bounds was as important as knowing the nominal. To this end, considerable time was spent during test planning and execution to assure that test technique uncertainties and sensitivity to a dispersed set of flight conditions could be modeled. The remainder of this section will give a highlight of the details involved in determining the parameter space. Then, the results section will provide the information needed to determine the functional dependence of the nominal and uncertainties.

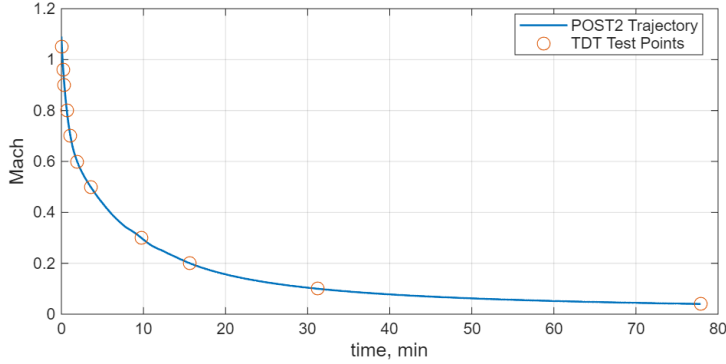


Fig. 8 Mach number time history for $M = 1.1$ to drogue release, used for test planning. POST2 nominal trajectory (circa 2022).

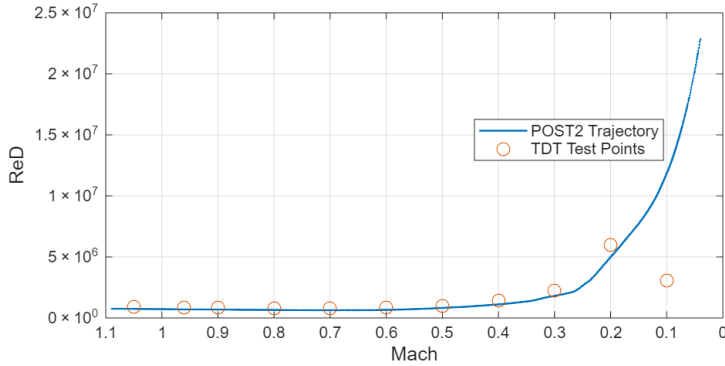


Fig. 9 Reynolds number for $M = 1.1$ to drogue release used for test planning. POST2 nominal trajectory (circa 2022).

A. Trajectory Data for Determine Flow Conditions

A nominal POST2 trajectory was used to set tunnel free-stream conditions, e.g. Mach number, total pressure, etc., that are essentially a function of altitude. The Mach time history and Re_D vs. M from a nominal trajectory are shown in Fig. 8 and Fig. 9, respectively. The aeroshell is descending under the drogue for the entire time shown. In Fig. 8, $t = 0$ secs. corresponds to $M = 1.1$ in the trajectory. shows that Mach number exponentially decreases around $M = 1$, indicative of the drag rise, and that most of the flight occurs at subsonic Mach numbers. The long flight time at subsonic Mach numbers and the exponential growth in Re_D are due to Titan's large positive density gradient near the surface.

B. Reduced Frequency Parameter, k

The reduced frequency parameter, k , is one of the important similitude requirements when conducting forced oscillation testing. Blunt bodies produce unsteady periodic flow field mechanisms (e.g. wake vortex shedding from flow separation near the shoulder). These periodic mechanisms can produce surface pressure fields that result in rigid body motion. Matching the reduced frequency parameter is required to capture these unsteady effects.

The nominal reduced frequency was determined assuming the aeroshell with drogue deployed can be modeled as a second order mass-spring-damper system [20, 21, 22]. Furthermore, it is assumed that the aeroshell with drogue deployed two-body system, as modeled in POST2, can be approximated by 1-DOF planar motion of only the aeroshell with the drogue modeled as a point force applied at distance, X_r , from the CG. The free-body diagram of the system is shown in Fig. 10. Since the system damping is unknown during the test planning phase, the undamped natural frequency is used to calculate the reduced frequency, k . The undamped natural frequency, ω , is computed from Eq. 3. This equation is derived from the 1-DOF pitching moment equation of motion as shown in Ref. [23] where C_{DS} is the drag area of the parachute calculated using the reference dimensions of the aeroshell. Also, it is assumed that the drogue experiences the same freestream dynamic pressure as the aeroshell. From Knacke [24] this is a reasonable assumption given the drogue canopy is approximately 10 body diameters downstream of the aeroshell. Then, reduced frequency is determined from $k = \omega L_{ref} / (2V_\infty)$. The reduced frequency needed for C_{n_r} will differ from the one for C_{m_q} since the mass moment of inertia for the yaw axis is different than the one for the pitch

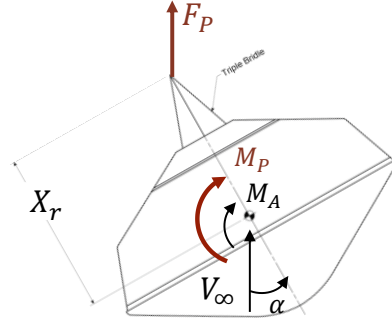


Fig. 10 Freebody diagram of the aeroshell with drogue deployed in planar motion. $M_P \equiv$ moment produced by drogue drag force, F_P . $M_A \equiv$ aeroshell aerodynamic restoring moment.

axis. These two values of k were determined during test planning. The differences between the two were small enough to forego the need to have a separate nominal data set for C_{n_r} .

The reduced frequency as a function of Mach number along the nominal trajectory at the time of the test is shown in by the blue line in Fig. 11. As Eq. 3 shows, the frequency is dependent on mass properties, static stability and the drogue design. These quantities will change during design maturation. Therefore, the test matrix included runs at

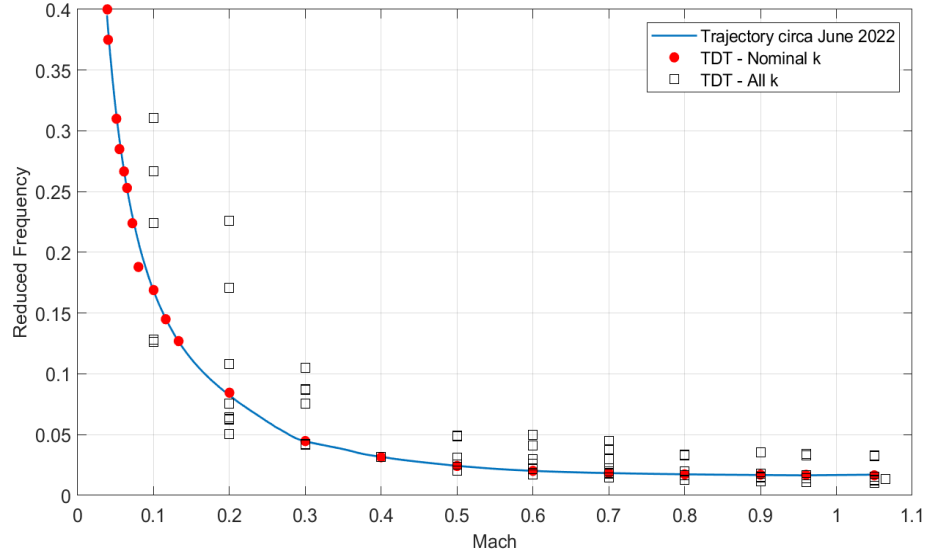


Fig. 11 Reduced frequency as a function of Mach number along a nominal trajectory. The square symbols are off-nominal k used for sensitivity studies.

off nominal k values to provide dynamic aero to account for these types of design changes. After the TDT test data were implemented in the ADB and run in POST2, the reduced frequencies from the simulation results were compared to the ones used in the TDT tests and were found to be within the range of k shown in Fig. 11.

$$\omega = \sqrt{\frac{-C_{m_\alpha} \bar{q} S_{ref} L_{ref} + (C_D S)_{chute} \bar{q} X_r}{I_{yy}}} \quad (3)$$

C. Non-dimensional Angular Rate, $\hat{q}$

When conducting dynamic aero testing, matching the non-dimensional angular rate similitude parameter is used to assure that the induced flow field caused by the rate is accurately modeled. Therefore, C_{mq} from the forced oscillation technique given in this paper can be referred to as a rate or amplitude effect. As a large range of $\hat{q}$ can occur in flight, the test matrix covered a $\hat{q}$ range limited only by the FO apparatus. The maximum amplitude provided by the FO rig varies with reduced frequency. In general, the largest amplitude achieved while matching nominal flight reduced frequency was 20° . The FO rig can provide up to 40° amplitude motion for reduced frequencies less than the nominal. The large amplitude effects were studied at these off-nominal reduced frequencies. Since $\hat{q}$ always varies, only one amplitude is chosen for developing the nominal C_{mq} values for which to build the uncertainties. The amount of test time required to do otherwise is prohibitive. If the test indicates a functional dependence on $\hat{q}$, then it can easily be modeled in the ADB for the nominal with the uncertainty model based on a single amplitude. An amplitude of 15° was chosen as it provides a signal-to-noise ratio in the strain gauge balance signals that helps reduce instrumentation error.

D. Repeatability and Reproducibility

Confidence in the nominal data is determined from the ability to repeat it and reproduce it. To this end, multiple runs using the nominal value of reduced frequency commensurate with Mach number (Fig. 11), at identical test conditions, were planned. The actual number was dependent on daily test schedule constraints. Also, the runs were conducted as α sweeps with equal negative and positive ranges. Since the aeroshell is axisymmetric with a centerline CG, symmetry rules were applied to the data so that negative α values could be averaged with positive values. This effectively doubles the number of data points to quantify the uncertainty of the nominal. The repeat data were obtained by returning to the test conditions typically after at least one calendar day. Although the TDT test was planned as a single entry, four entries spaced apart by a month or more were required to complete the test matrix given the tunnel scheduling and availability. The multiple entries, with a complete removal and reinstall of the hardware, provided the unique opportunity to include reproducibility data as a part of the uncertainty.

E. Support System Interference Effects

Support system interference effects are common in wind tunnel testing, and the effect can be significant. Quantifying the interference effects on the static aerodynamics is challenging. The effects on dynamic aero are more challenging. The sting enters the model from both sides just downstream of the maximum diameter affecting the flow over the model behind the sting and wake separation. Sting interference effects using CFD was conducted by the NESC [25]. The NESC study was a considerable effort, spanning over a year, using multiple CFD codes, and requiring a maturation of CFD to model blunt body dynamic aerodynamics. The results of the study are currently under investigation.

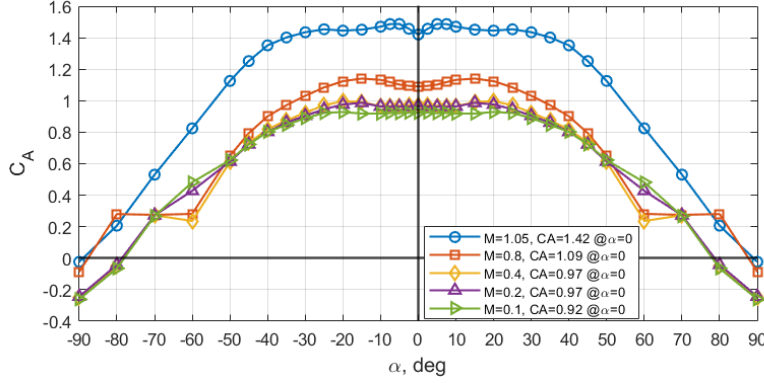


Fig. 12 Effect of Mach on C_A vs. α .

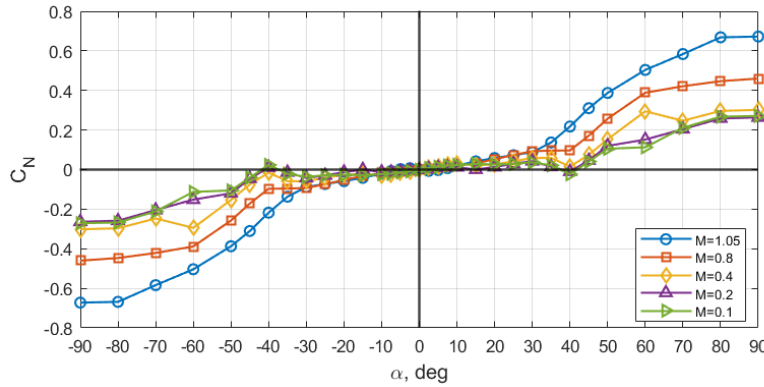


Fig. 13 Effect of Mach on C_N vs. α .

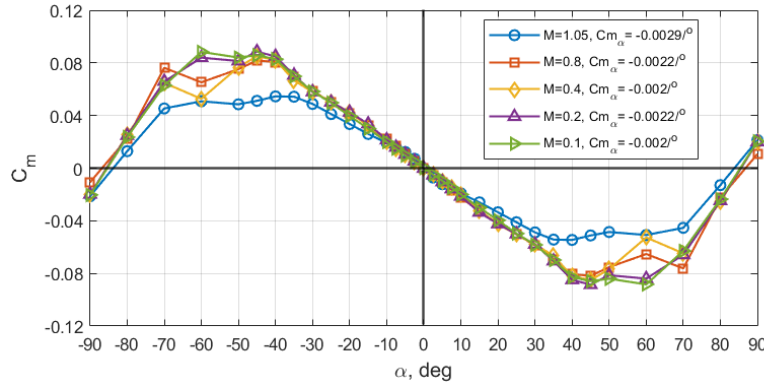


Fig. 14 Effect of Mach on C_m vs. α .

chute except Reynolds number for $M < 0.2$. The test obtained enough data to provide a physically realizable uncertainty model.

The effect of Mach on the static and dynamic aero coefficients is shown in Figs. 13-15 for α from -90° to $+90^\circ$ at a subset of the Mach numbers (0.1, 0.2, 0.4, 0.8, and 1.05) for clarity. The Mach effect on axial force coefficient is shown in Fig. 12. The value for C_A at $\alpha = 0^\circ$ is given in the legend. These values at $\alpha = 0^\circ$ shows the drag is nearly constant for low subsonic Mach numbers where compressibility effects are small. The as-expected large drag rise occurs as Mach number increases through the speed of sound. In contrast to other blunt bodies, C_A has nearly a constant value for 10 degrees of α around trim. The nonlinearity in the C_A values near $\alpha = 0^\circ$ at $M = 1.05$ was unexpected as the CFD solutions for this test [25] do not show this nonlinearity. The anomaly is still under investigation. The normal force coefficient, C_N vs. α is shown in Fig. 13. The data shows C_N is, in general, linear with almost no effect with

VI. Results and Discussion

Using the assumptions and methods described in section V, this section will present the test data, along with discussion, that shows how the results were used to determine the functional dependence of the subsonic and transonic aerodynamics. The data are discussed using plots of the static aero coefficients, C_A , C_N , C_m and the dynamic aero coefficient, C_{mq} as they vary with α , at $\beta = 0^\circ$. As previously mentioned, the TDT apparatus has only 1-DOF and cannot set α and β independently. The symbols in the plots indicate at which α value the data were recorded. The straight lines connecting the symbols are provided as a visual aid for detecting trends and represents the linear interpolation that the ADB uses to provide the aero coefficients to POST 2. Since similitude was satisfied for $M \geq 0.2$, subsection A shows the nominal aero coefficients as the aeroshell decelerates from $M = 1.1$ to 0.1. Then in subsection B, the uncertainty model is constructed and shown by plotting the maximum positive and negative bounds along with the nominal from subsection A. Section VI concludes by showing the dynamic stability sensitivity to the similitude parameters, reduced frequency and non-dimensional angular rate, in subsections C and D, respectively.

A. Effect of Mach

The TDT test collected an extensive aerodynamic data set for the Dragonfly aeroshell at $M = 0.1, 0.2, 0.3, 0.4, 0.5, 0.6, 0.7, 0.8, 0.9, 0.97, 1.05$ and α from -90° to $+90^\circ$. This data satisfies all similitude requirements for flight under drogue

Mach number for α near trim. As shown in Fig. 14, with a plot of C_m vs. α , the aeroshell has linear static stability for $\alpha < 35^\circ$ and $M \leq 1.05$. In the legend for this plot the static stability, C_{m_α} , near trim is given for each Mach number. For subsonic Mach numbers C_{m_α} is constant with a value of -0.0021 per degree. Static stability increases in the transonic flight regime. For $M = 1.05$, C_{m_α} has increased to -0.0029 per degree. In contrast to the aeroshell's linear static stability, the dynamic stability derivative, C_{mq} is highly nonlinear and unstable for $M \geq 0.2$, as shown in Fig. 15, by observing the entire α range for a given Mach number, C_{mq} can have a large variation in magnitude and whether it is positive (destabilizing) or negative (stabilizing). For $M < 0.2$ and the reduced frequencies commensurate with the drogue deployed, C_{mq} has stable values near trim. The significant effects of compressibility on C_{mq} are shown by the distinct change in the shape of the C_{mq} vs α curve for $M > 0.4$. For $M < 0.4$, C_{mq} is almost constant for $-20^\circ < \alpha < 20^\circ$ but transitions to an $-\alpha^2$ dependence at $M = 1.05$.

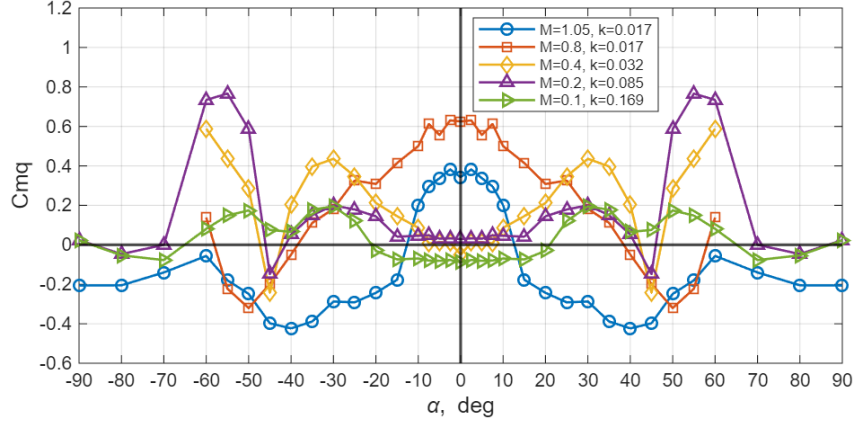


Fig. 15 Effect of Mach on C_{mq} vs α .

B. Uncertainty – Repeatability, Reproducibility, Instrumentation

This section will discuss the uncertainties due to repeatability, reproducibility, and instrumentation error. For brevity, repeatability and reproducibility will be called repeatability. These uncertainties are calculated using Eq. 4 for each aero coefficient, C_x .

$$\hat{U}_{Cx} = h_1 \sqrt{2 * (\hat{\sigma}_{Cx})_{repeat, reprod}^2 + (\hat{\sigma}_{Cx})_{instrumentation}^2} \quad (4)$$

The root-sum-squared terms in Eq. 4 are uncorrelated random variables. To give full coverage for uniform uncertainties the value of h_1 is $\sqrt{3}$. The (repeat, reprod) term is calculated from data at identical test conditions within

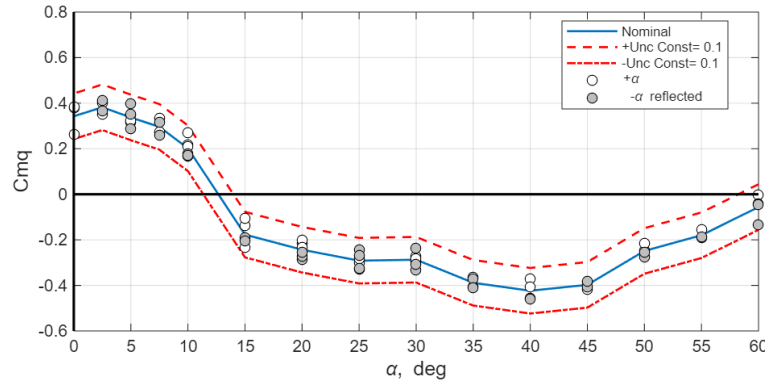


Fig. 16 Nominal and uncertainty bounds along with the experimental data used in Eq. 4. $M = 1.05$, $k = 0.017$, $\hat{q} = 0.0044$.

an entry and across entries. As mentioned previously, the data taken at $\alpha < 0^\circ$ are reflected about $\alpha = 0^\circ$ by applying symmetry. The reflected data, with α values that match a positive α value, are considered a repeat point. The instrumentation uncertainty term is at least an order of a magnitude less than the repeatability variations. Therefore, the uncertainty bounds shown in the following plots are primarily due to repeatability and reproducibility. The uncertainty increment, $\hat{U}_{Cx}$ in Eq. 4 is determined at each α with 4 or more repeat points with most α having 6 points. The test data show that the uncertainty increment is a function of α . The uncertainty increment as a function of α

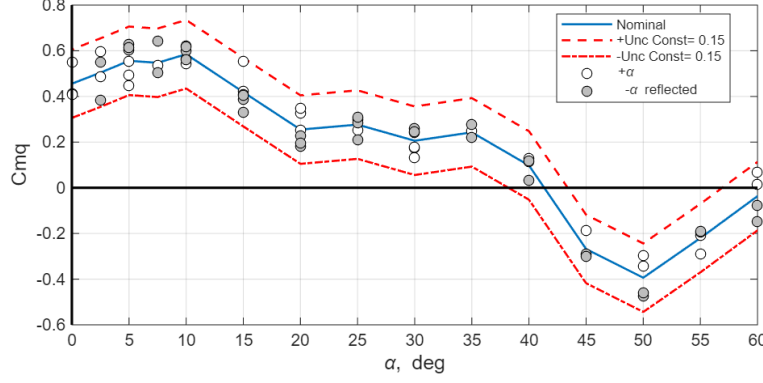


Fig. 17 Nominal and uncertainty bounds along with the experimental data used in Eq. 4. $M = 0.7$, $k = 0.019$, $\hat{q} = 0.0049$.

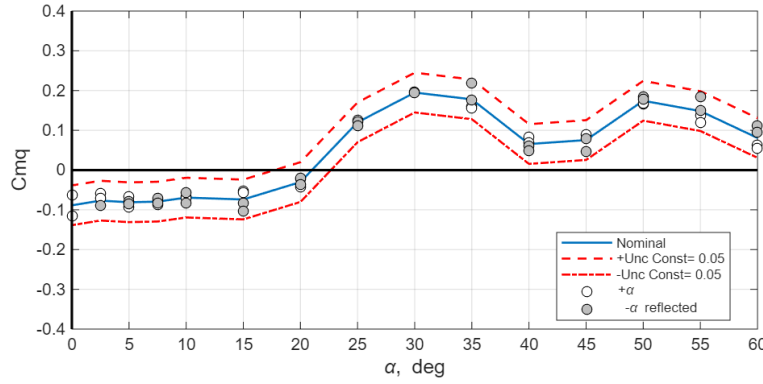


Fig. 18 Nominal and uncertainty bounds along with the experimental data used in Eq. 4. $M = 0.1$, $k = 0.169$, $\hat{q} = 0.0441$.

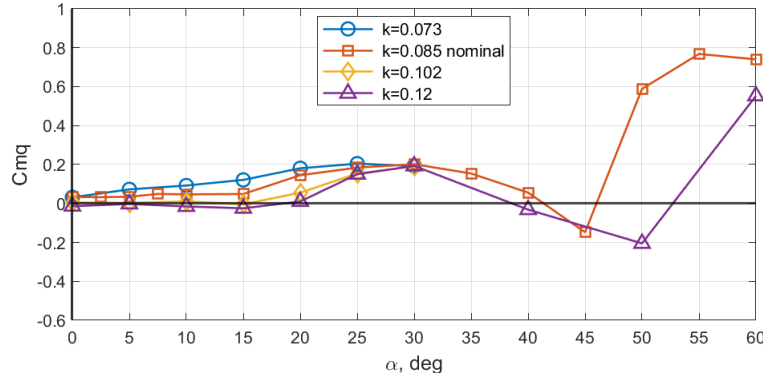


Fig. 19 Sensitivity of C_{mq} to k at $M = 0.2$ and $\hat{q} = 0.022$.

data in the plots of Figs. 19-22 show that as k increases dynamic stability increases. Figure 24 shows the C_{mq} data taken at $M = 0.1$, but at different reduced frequencies. The Mach numbers given in the legend are shown as a map

was replaced by a value that is constant with α . This constant increment is greater than or equal to the maximum for all α in that run. Therefore, this increment represents a conservative uncertainty on the nominal. The plots show the nominal, solid blue line, which is the average of all the repeat/reprod points, and uncertainty bounds, along with the experimental data used in Eq. 4. A representative set of data for C_{mq} to show how the uncertainty bounds cover the data at $M = 1.05$, 0.7 and 0.1 are given in Figs. 16-18, respectively. In these figures, the red dash lines in the plots shows the uncertainty increment used in the ADB. In the subsequent sections, the uncertainty bounds shown in the plots were determined as explained in this section. The uncertainty bounds will be used as one way to quantify the effect of the parameter on the aerodynamic coefficient.

C. Effect of Reduced Frequency Parameter, k

The reduced frequencies used in the test and how they were determined was discussed in Section V. As a reminder, the nominal k values shown are for the aeroshell with the drogue deployed. These values of k are much higher than the ones for the aeroshell in free flight. The effect of reduced frequency on C_{mq} for the subsonic Mach numbers $M = 0.2$, 0.3 , 0.5 , and 0.6 are shown in Figs. 19-22, respectively. The effect of reduced frequency on C_{mq} is nonlinear with α except for $\alpha = 30^\circ$ where there is no effect of k . The windward side of the forebody cone is normal to the oncoming flow with the stagnation point near the shoulder for this α (Fig. 23). This result indicates the frequency effects are associated with attached flow on the heatshield then separating at the shoulder. In general, the

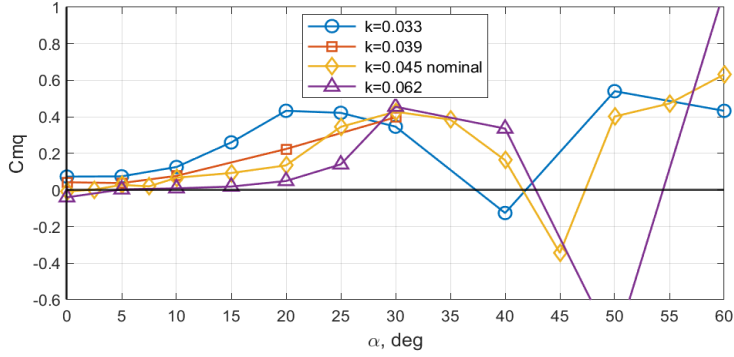


Fig. 20 Sensitivity of C_{mq} to k at $M = 0.3$ and $\hat{q} = 0.012$.

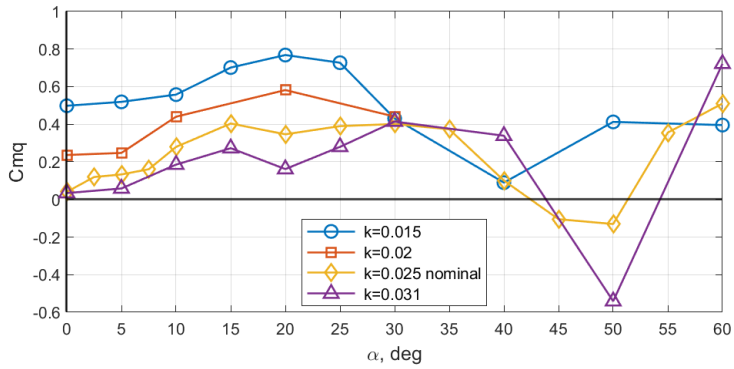


Fig. 21 Sensitivity of C_{mq} to k at $M = 0.5$, $\hat{q} = 0.007$.

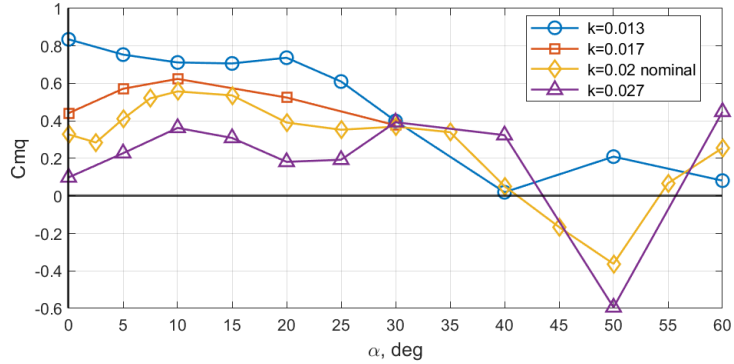


Fig. 22 Sensitivity of C_{mq} to k at $M = 0.6$, $\hat{q} = 0.005$

back to the trajectory. The aeroshell continues to decelerate until drogue release. For these incompressible Mach numbers, k continues to increase dramatically. Therefore, the change in C_{mq} in this flight regime is a function of k . The highest value of k shown in Fig. 24 is the value at drogue release. For $\alpha < 20^\circ$ the stabilizing increment in C_{mq} , by increasing k , exponentially decays to an asymptotic value as k approaches a value of 0.4.

An effect of the drogue that is rarely discussed or published is its impact on the damping derivatives of the aeroshell. In Fig. 24 the C_{mq} values for $k = 0.1$ (blue line with circle symbols) are the ones if the aeroshell was in free flight at $M = 0.1$. These C_{mq} values are unstable, a well-known characteristic of an aeroshell with an OML geometry and X_{cg} location as the Genesis capsule. The red line with square symbols is the data for flight under drogues at $M = 0.1$. The k value for this data is 0.2 (twice the k value for free flight). The data show a large stabilizing delta in C_{mq} for $\alpha < 30^\circ$. Therefore, the drogue stabilizes the aeroshell both statically and dynamically.

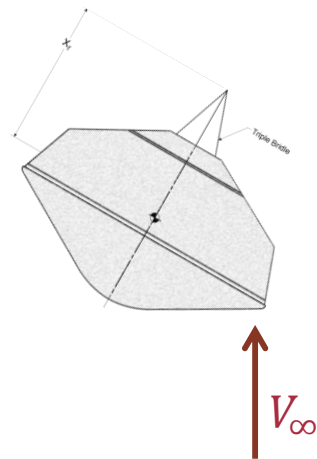


Fig. 23 Stagnation point is near the shoulder when oscillating about $\alpha = 30^\circ$.

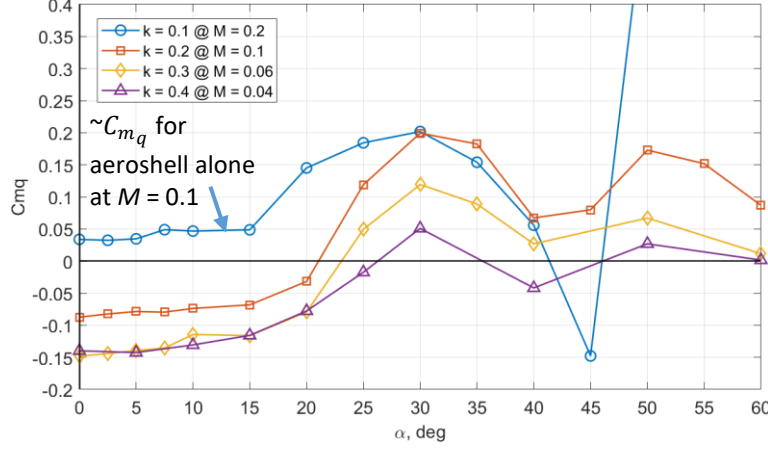


Fig. 24 C_{m_q} for k values in the incompressible Mach regime. Note, the data for $k = 0.4$ is at drogue release.

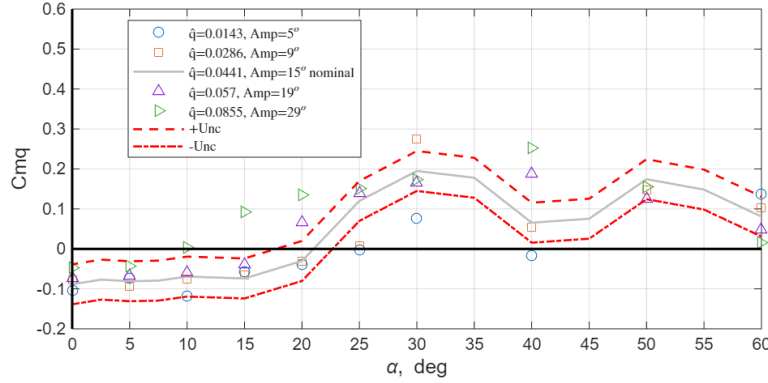


Fig. 25 Sensitivity of C_{m_q} to $\dot{q}$ at $M = 0.1$ and $k = 0.164$.

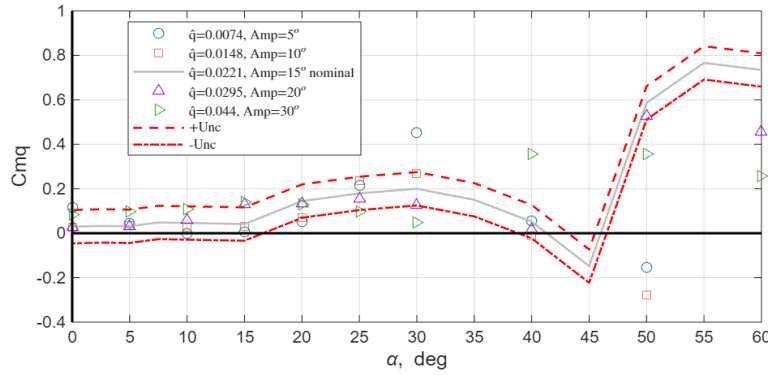


Fig. 26 Sensitivity of C_{m_q} to $\dot{q}$ at $M = 0.2$ and $k = 0.085$.

D. Effect of Non-Dimensional Pitch Rate, $\dot{q}$

As discussed in the Aerodynamic Modeling section (Section III), $\dot{q}$ is a similitude requirement for C_{m_q} . Matching non-dimensional rate assures that the induced flow field caused by angular rate is reproduced in the sub-scale wind tunnel test. The effect of $\dot{q}$ on C_{m_q} for Mach = 0.1, 0.2, 0.5, and 0.9 is shown in Figs. 25-28, respectively. Prior to the test, some POST2 MC results had shown large angular rates associated with 60° amplitude motion. Due to limitations of the rotary actuator, OTT, it was not always possible to match the amplitude and angular rates seen in the MC results. The data shown represent the largest practical range of $\dot{q}$ and represent the greatest sensitivity to $\dot{q}$ for subsonic and transonic Mach numbers. The nominal with minimum and maximum uncertainty bounds, as determined in the section on uncertainties (Section VI B), are co-plotted with the data. Recall, using this nominal and uncertainty in the ADB represents a C_{m_q} that is constant with $\dot{q}$.

Two factors are used to determine if C_{m_q} needs to be modeled as a function of $\dot{q}$: 1.) it must have a discernable trend relative to the nominal and 2.) must exceed the uncertainty bounds for most α or at least the ones near trim ($\alpha = 0^\circ$) since these C_{m_q} values are the most impactful for oscillatory dynamics. Recall, the uncertainties on C_{m_q} are dispersed uniformly. Therefore, if the C_{m_q} data for most α are contained within the uncertainty bounds then the MC simulation results will cover these values. The plots in Figs. 25-28 show no discernable trend and only $M = 0.1$ and 0.2 showed exceedance of the uncertainty bounds. Even so, these exceedances occurred at α away from trim. Therefore,

the nominal for the damping derivatives will not have a functional dependence on non-dimensional angular rate. The uncertainties shown here cover the variance measured in the test.

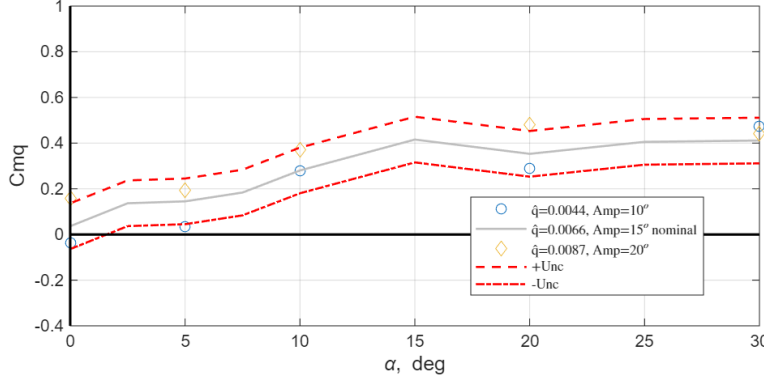


Fig. 27 Sensitivity of C_{m_q} to $\hat{q}$ at $M = 0.5$ and $k = 0.025$.

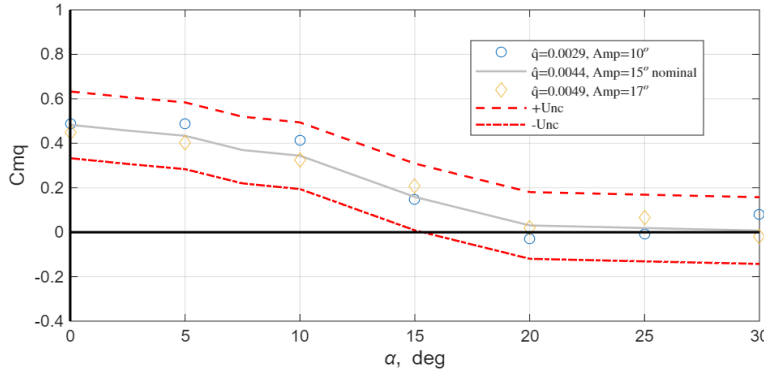


Fig. 28 Sensitivity of C_{m_q} to $\hat{q}$ at $M = 0.9$ and $k = 0.025$.

for $M < 0.2$. The extensive data set provided an uncertainty model with realistic bounds early in the design phase.

The static aerodynamics was presented with the static aero force and moment coefficients, C_A , C_N and C_m as a function of Mach and α . The effect of Mach and α on C_A are similar to other blunt-body capsules with an interesting difference that it has almost a constant value within 10° of trim. Also, within 10° of trim, C_N and C_m are linear with α and nearly constant with Mach number. Additionally, the aeroshell was shown to be statically stable ($C_{m_\alpha} < 0$) for a large α -range ($\sim \pm 40^\circ$) from trim. In contrast to the static stability, C_{m_q} is highly nonlinear and unstable near trim for $M \geq 0.2$. The effects of compressibility were shown as a change in C_{m_q} from being nearly a constant value for $\alpha < 20^\circ$ to an $-\alpha^2$ dependence at $M = 1.05$.

The dynamic aerodynamics were presented with the dynamic stability derivative, C_{m_q} , as a function of Mach and α at conditions provided by a nominal Titan EDL trajectory. The sensitivity of C_{m_q} to the dynamic similitude parameters reduced frequency, k , and nondimensional angular rate, $\hat{q}$, was evaluated.

The effect of k on C_{m_q} for $M < 0.8$ is nonlinear with α except for $\alpha = 30^\circ$ where there is no effect of k . It was pointed out that this is the α that matches the heatshield half-cone angle, and the flow is normal to the heatshield surface. This result indicates the frequency effects are associated with attached flow on the heatshield then separating at the shoulder. In general, as k increases, dynamic stability increases. This stabilizing effect on C_{m_q} exponentially decays to an asymptotic value as k approaches a value of 0.4. The drogue size and attach point location has a large impact on the aeroshell's reduced frequency. For example, the aeroshell in free flight at $M = 0.1$ and $k = 0.07$ has *unstable* values of C_{m_q} . The aeroshell with the drogue deployed at $M = 0.1$ and $k = 0.17$ has large *stable* values of C_{m_q} . Therefore, an increase in the drogue size and/or attach point location has a two-fold beneficial effect on the aeroshell's stability. It produces a larger restoring moment and more stable values for C_{m_q} .

The effect of $\hat{q}$ on C_{m_q} was investigated to determine if the damping derivative model in the ADB would require a functional dependence on $\hat{q}$. At the nominal reduced frequency values, the data showed that C_{m_q} varied with $\hat{q}$ but did not show a discernable trend. Except at high α for $M = 0.1$ and 0.2 , the variances in C_{m_q} due to $\hat{q}$ are within the

VII. Summary and Conclusions

Dynamic stability is one of the major concerns for meeting Dragonfly EDL performance requirements exacerbated by a long flight time, greater than 2 hours, with most of this time spent at subsonic Mach numbers. The Dragonfly program uses 6-DOF POST2 Monte Carlo assessments of the EDL trajectory data to make critical design decisions and accurate risk assessments. The Monte Carlo simulation assessments require a 6-DOF aero database of static and dynamic aerodynamic coefficients that cover a large parameter space with physically realizable uncertainties. To provide this data for $M < 1.1$, a test in the NASA LaRC Transonic Dynamics Tunnel, using the static and forced oscillation test techniques, was planned and executed to match flight similitude for the aeroshell flying with a drogue parachute deployed. The TDT test collected an extensive aerodynamic data set for the Dragonfly aeroshell at $M = 0.1, 0.2, 0.3, 0.4, 0.5, 0.6, 0.7, 0.8, 0.9, 0.97, 1.05$ and α from -90° to $+90^\circ$. This data satisfies all similitude requirements except Reynolds number

uncertainty bounds. Therefore, for the purposes of the Dragonfly aeroshell ADB, the dynamic aero coefficients will not have a functional dependence on non-dimensional angular rate.

In summary, the TDT test provided a comprehensive data set that satisfies flight similitude for the Dragonfly aeroshell during the $M < 1.1$ phase of EDL. Even so, this paper pointed to a few areas that should be addressed in future tests: 1.) conduct a comprehensive Reynolds number study and 2.) design and implement a forced oscillation apparatus that can set α and β independently to determine the validity of modeling C_{mq} and C_{nr} with only α_T in the subsonic and transonic flight regime.

VIII. Acknowledgements

The above and beyond contributions from the dedicated TDT technicians and engineers made a large impact on the success of the Dragonfly aeroshell dynamic stability test. The test would not have been possible without the support of the following NASA organizations in no particular order: the Dragonfly EDL Working Group, Flight Dynamics Branch, Aeroelasticity Branch, and Atmospheric Flight & Entry Systems Branch. Additionally, the authors would like to acknowledge the insight provided by the results from the NESC CFD sting interference studies.

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