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Analysis of payload compartment venting of satellite launch vehicle

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Abstract. The problem of flow through the vent is formulated as an unsteady, nonlinear, ordinary differential equation and solved using Runge-Kutta method to obtain pressure inside payload faring. An inverse problem for prediction of the discharge coefficient is presented employing measured internal pressure of the payload fairing during the ascent phase of a satellite launch vehicle. A controlled random search method is used to estimate the discharge coefficient from the measured transient pressure history during the ascent period of the launch vehicle. The algorithm predicts the discharge coefficient stepwise with function of Mach number. The estimated values of the discharge coefficients are in good agreement with differential pressure measured during the flight of typical satellite launch vehicle.

Keywords: depressurization; discharge coefficient; heat shield; inverse problem; numerical analysis; reentry module; satellite launch vehicle; venting

1. Introduction

At the time of launch, the pressure inside the payload fairing of a satellite launch vehicle is at sea level. The ambient atmospheric pressure decreases rapidly during the ascent trajectory of the vehicle, causing a build-up of differential pressure across the heat shield. During the atmospheric flight, the heat shield compartment of a satellite launch vehicle needs venting to prevent abnormal pressure buildup, which can be detrimental to the structure. The differential pressure depends mainly on the location of the vent holes, the effective volume of air to be evacuated, and the trajectory of the launch vehicle. A compressible flow loss coefficient is essential for the mass flux calculation under polytropic process to model the venting process and to predict the differential pressure-time history reasonably to maintain permissible structural load.

The space vehicle design criteria monograph (NASA 1970) describes design criteria of compartment venting during ascent and reentry phase of space vehicles. The international reference guide (Isakowitz *et al.* 2004) to space launch gives the maximum differential pressure and rate of change of pressure inside the fairing. The problem becomes more critical for large or complex rockets (Pritchett *et al.* 2016) such as Titan launch vehicle (Rogers *et al.* 2015), Viking and Space Shuttle. More complex configurations such as the space shuttle require venting in which a quantity of small experimental package may be kept in a large payload container within a cargo

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bay and exposed to various conditions of the ascent and reentry trajectory conditions.

Experimental studies were carried out by Mironer and Regan (1983) to determine the venting design criteria for the space shuttle payload, using a nominal ascent trajectory, and payload bay pressure profile. Murri (1987) has experimentally studied venting design criteria for space shuttle payloads using worst case ascent phase trajectory, and payload bay pressure profile. They obtained a single curve that indicates the maximum differential pressure which can be expected for a given vent hole diameter. Flow areas for series and parallel compartment venting to satisfy pressure differential requirements have been studied by Kirby and Ivy (1973). Experiments were conducted by John and Jones (1974) in the 8×6 ft supersonic wind tunnel of the NASA Lewis Research Centre to find the effective discharge coefficient for the venting analysis and application of the Titan/Centaur launch vehicle. It is very difficult to obtain the discharge coefficient from experiments.

Murca (1967) has considered aerodynamically-induced loads in the design of a sounding rocket, which are due to differential pressures occurring across internal bulk heads and across the vehicle outer surface. Fay and Hengel (1993) analyzed the flow through the vent connecting the multi-compartment using a quasi-steady isentropic equation with empirical discharge coefficient. Space shuttle post flight analysis (Lutfi *et al.* 1983) has revealed discrepancies between measured and computed values of the differential pressure inside the shuttle which are attributed to external local pressure based on the subscale model of the wind tunnel test data as compared to the actual vehicle external pressure measured during the flight of the space shuttle. Most of the predictions of the discharge coefficient are based on the flight-derived vent port pressure coefficients, because the wind tunnel does not adequately define the orbital ascent pressure environment. An inverse analysis has been carried out by Mehta (2003) to estimate the discharge coefficient of the orifice in conjunction with the measured internal pressure during the flight.

An analytical approach of the discharge process of a compartment into a decreasing time-dependent pressure environment has been published by Sanz-Andres *et al.* (1997). A closed-form expression for the isothermal (Mehta 1999) and isentropic (Mehta 2002) venting has been presented using small time-steps during the short time of depressurization. Dykhuizen *et al.* (2012) have derived analytical solutions to calculate the internal pressure of vented enclosure during launch.

The effective discharge coefficient for multi-row vent-holes on the payload fairing of Titan IV launch vehicle has been numerically obtained using three-dimensional Computational Fluid Dynamics (CFD) technique by Huseman and Chern (1997). The discharge coefficient of vent holes has been obtained for a range of external flow Mach numbers and internal-to-external pressure ratios applicable to Titan IV flight trajectory. Brower (2006) has reported the internal payload fairing compartment pressure inside the Titan launch vehicle. Analytical solutions to compute the decompression of pressurized aircraft cabin are derived by Pagani and Carrera (2016) under constant ambient pressure conditions and used for computing structural loads on hinged panes.

Many closed-form solutions for isothermal, polytropic, and isentropic decompression were derived for re-pressurization for aircraft and spacecraft by Mavriplis (1963). The discharge coefficients were obtained through measurements, CFD analyses (Breard *et al.* 2004) or sensitive analyses (Daidzic and Simones 2010). Venting analysis of a Boeing 747 aircraft fuel tank has been carried out by Jensen (2000). It is worth to mention here the major difference between the depressurization process of aircraft and launch vehicle is that in the case of aircraft the back pressure is equal to the atmospheric pressure of ambient air and remains constant during the process of decompression but in the case of space launch vehicles the ambient pressure falls

is the air velocity at the exit of the hole. The discharge process is modeled as isentropic. This is a reasonable assumption, as the outflow is very rapid and the orifice is relatively short. Here, C_D is the discharge coefficient of the vent orifice. A is the area of the vent hole. The discharge coefficient is defined as the ratio between the actual diabatic irreversible outflow and the theoretical maximum possible or isentropic outflow. The discharge coefficients used should be applicable in terms of orifice Reynolds number and pressure ratio across the orifice. The local external flow conditions of Mach number and boundary layer thickness and profile should be accounted for in orifice flow analysis.

Using the first law of thermodynamics for an isentropic process and employing the pressure-density-temperature relationship in the form of perfect gas law, one obtains isentropic relationship (Liepmann and Roshko 2007) as a function of the internal pressure level as

$$\rho_a = \rho_c \left(\frac{p_a}{p_c} \right)^{\frac{1}{\gamma}} \tag{2}$$

where p is pressure and subscripts a and c represent the conditions at ambient and inside the compartment, respectively and γ is ratio of specific heats. The velocity u is a function of pressure and density (Shapiro 1953) and can be written as

$$u = \sqrt{\frac{2\gamma}{\gamma - 1} RT_c \left\{ 1 - \left(\frac{p_a}{p_c} \right)^{\frac{(\gamma-1)}{\gamma}} \right\}} \tag{3}$$

Substituting Eq. (3) and introducing the perfect gas law into Eq. (1) and yields (Haber and Glamann 1953)

$$\dot{m}_c = \frac{C_D A p_c}{\sqrt{RT_c}} \sqrt{\frac{2\gamma}{\gamma - 1} \left(\frac{p_a}{p_c} \right)^{\frac{2}{\gamma}} \left\{ 1 - \left(\frac{p_a}{p_c} \right)^{\frac{(\gamma-1)}{\gamma}} \right\}} \tag{4}$$

The maximum mass flow rate equation (Shapiro 1953) for choked flow is

$$\dot{m}_{max} = \frac{C_D A P_c \sqrt{\gamma}}{\sqrt{RT_c}} \left(\frac{2}{\gamma + 1} \right)^{-\frac{(\gamma+1)}{2(\gamma-1)}} \tag{5}$$

Considering the speed of sound a in the compartment is

$$a = \sqrt{\gamma R_c T_c} \tag{6}$$

The maximum flow equation can be written as

$$\dot{m}_{max} = C_D a \rho_c A \left(\frac{2}{\gamma + 1} \right)^{-\frac{(\gamma+1)}{2(\gamma-1)}} \tag{7}$$

and the compartment air pressure versus atmospheric pressure ratio decrease to a critical pressure ratio (Shapiro 1953) as

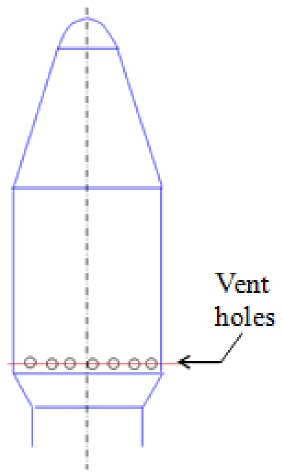

Fig. 1 Typical payload fairing with vent holes

$$\Delta p = p_c - p_a \tag{12}$$

3. Flowfield analysis over the payload fairing

3.1 Payload fairing and vent holes

The volume of air to be evacuated in the heat shield of a typical launch vehicle is about 42 m^3 and the vent area is about 0.0472 m^2. The vent area is distributed as a number of circular holes. The Reynolds number based on the velocity at the exit of the vent hole and based on the orifice diameter varies in the range of 3×10^4–1.2×10^5. The ambient pressure p_a is interpolated using the atmospheric table (Anathasayanam *et al.* 1987) corresponding to the instantaneous vehicle altitude. To evaluate the performance of the depressurization process during the ascent phase of the satellite launch vehicle, two pressure transducers have been used to measure the heat shield compartment pressure and outside pressure in the vicinity of the vent. To find the most appropriate location for the venting orifice a criteria was established (NASA 1970) which recommends that the difference between the static pressure at the vent location over the payload fairing surface and the ambient pressure close to zero. In the next section a numerical analysis is carried out and described.

3.2 Flow field and pressure distribution over the payload fairing

A flow solver code is developed by Mehta (1997) to solve the turbulent axisymmetric Reynolds-averaged Navier-Stokes equations using finite volume method in conjunction with three-stage Runge-Kutta time-stepping scheme with Baldwin-Lomax turbulence model. Numerical simulation has been performed to obtain pressure distribution over a bulbous payload shroud at zero angle of attack in the Mach number range of 0.8-3.0 and Reynolds number range of 3.314×10^7/m–4.682×10^7/m. Fig. 1 shows the typical heat shield of satellite launch vehicle and Fig.

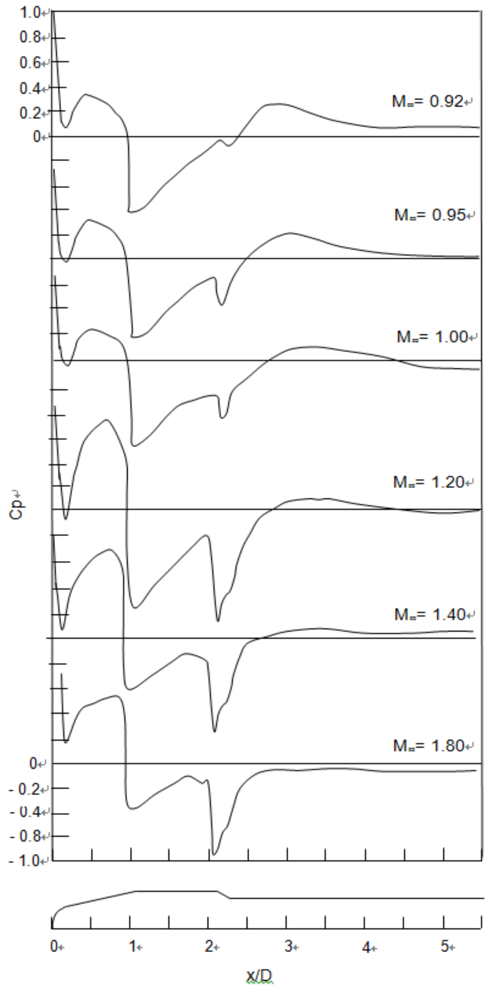

Fig. 4 Pressure distribution along the launch vehicle

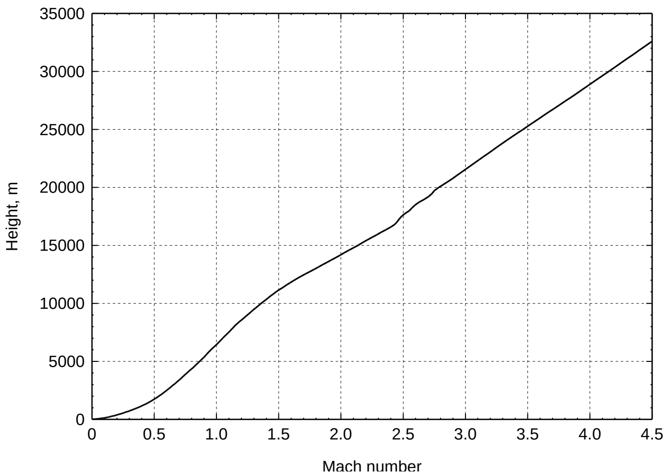

Fig. 5 Trajectory of the launch vehicle

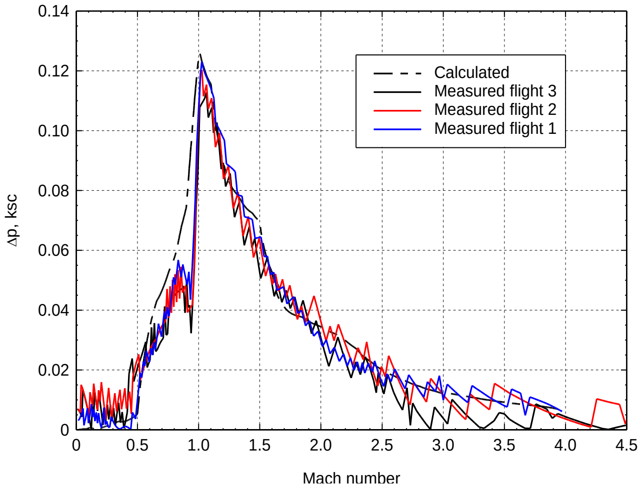

Fig. 7 Reconstructed differential pressure using predicated discharge coefficients

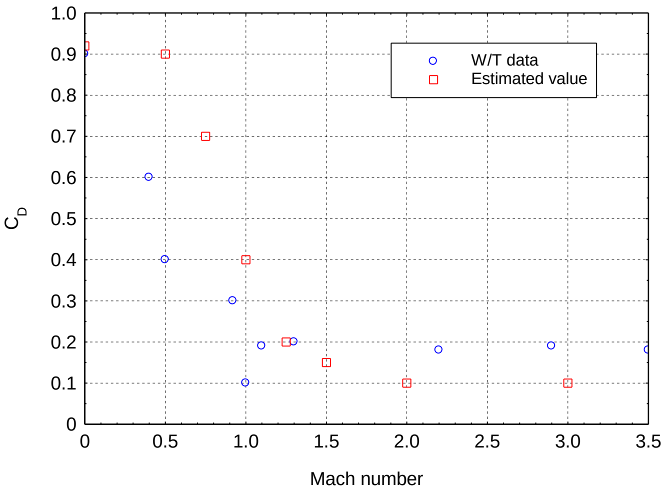

Fig. 8 Comparison between estimated and experimental discharge coefficients

The details of controlled random search algorithm are described in the estimation of convective heat transfer coefficient by Mehta and Tiwari (2007). The CRS algorithm predicts the discharge coefficient as a function of stepwise Mach number. The trajectory of the vehicle is depicted in Fig. 5. The ambient pressure is computed using the atmospheric data (Anathasayanam *et al.* 1987) in conjunction with the vehicle trajectory. Fig. 6 displays the measured internal pressure inside the payload fairing for different flights (Sachdeva and Mehta 2008). The measured (Sachdeva and Mehta 2008) and reconstructed differential pressures with respect to the freestream Mach number are shown in Fig. 7. The differential pressure for the payload compartment should not exceed 0.15×10^5 Pa. The maximum differential pressure is seen in the vicinity of sonic Mach number, which is attributing the choked condition in Fig. 7. Reconstructed differential pressures show good agreement with the numerical results. It is important to mention here that the maximum differential pressure and the rate of decrease as the differential pressure are within the permissible limits as mentioned (Isakowitz *et al.* 2004). Table 1 shows the estimated value of discharge coefficient in the Mach number range. The predicted value of the discharge coefficient is compared with measured value (John *et al.* 1974) and shown in Fig. 8.

The comparisons between the estimated values obtained using the controlled random search method and experimental values depict good agreement between them.

5. Conclusions

Venting analysis of a typical launch vehicle payload compartment is carried out by solving the first order nonlinear differential equation using a fourth order Runge-Kutta method. The discharge coefficients have been estimated employing the controlled random search method using compartment pressure measurements from the first flight. The estimated discharge coefficients show good agreement for the subsequent flights. The measured compartment pressures are found to be consistent with the reconstructed differential pressure in conjunction with estimated discharge coefficient as a function of the flight Mach number.

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