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SIMULATION OF THERMO-ACOUSTIC INSTABILITIES INCLUDING MEAN FLOW EFFECTS IN THE TIME DOMAIN

Jutta Pieringer*[1], Thomas Sattelmayer[1]

[1]Lehrstuhl für Thermodynamik, Technische Universität München
Boltzmannstraße 15, D-85748 Garching, Germany
pieringer@td.mw.tum.de

Abstract

Thermo-acoustic instabilities may arise in various technical combustion systems, such as industrial and household burners, gas turbines, afterburners or rocket engines. These self-excited oscillations originate from the interaction of combustor acoustics and flame dynamics. To predict their occurrence it is necessary to describe the interaction between the propagation of acoustic perturbations, flame behaviour and acoustic losses correctly. Most methods for calculating thermo-acoustic instabilities suffer from a high degree of simplification. Mean flow effects are usually completely neglected.

A new approach for the simulation of combustion instabilities in the time domain including mean flow effects is presented. The basic idea of the method is to solve the unsteady linearised governing equations for a three-dimensional combustor geometry. A rocket engine where mean flow effects are particularly strong is used as example. It is shown that this method delivers promising results that do not require the use of nozzle admittances. The differences in oscillation amplitudes with and without consideration of the mean flow are highlighted.

INTRODUCTION

Efforts to predict combustion instabilities started when this phenomenon became a problem in the development of rocket engines. Many scientists have done considerable research work on that subject since. In the early pioneering work published by Crocco [1][2], Zinn [3] and Culick [4] strong simplifications of models and equations were required due to the limited computer performance at that time. The inclusion of the rocket nozzle was realized by the use of a “nozzle admittance”, which reduces the complex three-dimensional reflections of acoustic waves in the convergent part of the nozzle to a simplified, one-dimensional behaviour.

Taking advantage of the improved performance of modern computers new field methods have been developed in the last years. These solve either the wave-equation or the Navier-Stokes-equations numerically. This paper presents a field method solving linearised governing equations including terms describing mean flow effects. This approach allows incorporating areas with strong mean-flow gradients into the computational domain. The feasibility will be demonstrated for the case of a rocket engine where the convergent part of the nozzle is integrated into the computational domain.

MODELLING STRATEGY

Basic Approach

Combustion instabilities are caused by the interaction of chamber acoustics and the combustion process. It has been shown that field methods solving an inhomogeneous wave-equation [6] [5] lead to promising results for an annular combustor and a liquid propellant rocket engine. In both cases mean flow effects were omitted. In the case of the annular combustor, Mach-numbers are very low. Therefore the assumption of negligible mean flow effects is justified. In rocket engines, however, the mean flow is accelerated up to very high Mach-numbers, reaching sonic conditions in the nozzle throat. The classical approach to include these effects is to represent the nozzle by a nozzle admittance boundary condition. This method was also applied in [5].

The use of nozzle admittances has several drawbacks: first of all they are difficult to apply in time dependant approaches, because they depend on the nozzle geometry, oscillation frequency and oscillation modes [3]. Secondly models that propose constant values of nozzle admittances as e.g. the model derived by Marble & Candel [7] are only valid for compact nozzles. Additionally nozzle admittances reduce the complex three-dimensional reflection behaviour of nozzles to a single value. However, the correct inclusion of acoustic losses caused by the nozzle is essential for the prediction of combustion instabilities. Particularly for rocket thrust chambers without acoustic absorbers, damping induced by the nozzle is the only essential damping effect aside the droplet drag in case of liquid propellants.

These drawbacks can be avoided by including the convergent part of the nozzle into the computational domain. Solving the linearised Euler equations or acoustic perturbation equations for a given mean flow profile, refraction and reflection effects induced by the important mean flow gradients in this zone can be taken into account. It is shown that oscillation amplitudes obtained by this new approach differ considerably from those obtained by the classical method.

The subject of this paper is restricted to the inclusion of the nozzle. The method presented assumes linear acoustics and considers the flame via a linear flame transfer function. However, to asses the stability of a real rocket engine, a more elaborate model incorporating damping and non-linear effects is necessary.

Governing Equations

Combustion instabilities are characterised by fluctuations of the pressure p, the velocity **u** and the density ρ. The employed modelling approach is based on the direct solution of the linearised governing equations for the fluctuation quantities p', **u**' and ρ' in the time domain for a three-dimensional combustor geometry. Important advantages of the method are that it does not depend on the assumption of oscillation mode shapes and that both, the modes and frequencies are direct results of the simulation.

The linearised Euler equations can be derived from the conservation equations of mass, momentum and energy by linearization. Their solution consists of vortical, entropy and acoustic components which propagate and interact in non-uniform mean flows [9]. Acoustic perturbation equations (APE) are obtained from the linearised Euler equations by "source filtering" [8] and describe wave propagation in non-uniform mean flows. In contrast to the linearised Euler equations they do not describe convection of vorticity and entropy modes [8]. As the focus of this paper lies on the acoustical behaviour, an APE system for compressible mean flows was chosen. Simulating self-excited combustion instabilities, only the source term involving heat release fluctuations $\dot{q}'$ [W/m^3] has to be considered. The resulting set of equations reads:

$$\frac{\partial p'}{\partial t} + \bar{c}^2 \nabla \cdot \left(\bar{\rho}\mathbf{u}' + \bar{\mathbf{u}} \frac{p'}{\bar{c}^2} \right) = (\gamma - 1)\dot{q}' \tag{1}$$

$$\frac{\partial \mathbf{u}'}{\partial t} + \nabla(\bar{\mathbf{u}} \cdot \mathbf{u}') + \nabla\left(\frac{p'}{\bar{\rho}}\right) = 0 \tag{2}$$

The density fluctuation ρ' is linked to the pressure fluctuation via the linearised isentropic relation $p' = \bar{c}^2 \rho'$.

Heat Release Model

To get a closed form of equation (1) a model for the volumetric heat release fluctuation must be provided. In [5] the classical model introduced by Crocco [1], known as the "n-τ-model", had delivered suitable results in a time dependent approach based on the wave equation. Therefore, like in [5], the following model for the heat release fluctuation is used:

$$\dot{q}'(t,x) = \Delta H_R \cdot n \cdot \frac{\bar{\dot{m}}_V}{\bar{p}_c} \cdot \left(p'(t,x) - p'(t-\tau,x)\right) \tag{3}$$

This model is based on the form proposed by Culick [4]. It involves the mean flow of propellant converted from liquid to gas $\bar{\dot{m}}_V$, the mean chamber pressure $\bar{p}_c$ and the released reaction enthalpy of both propellants ΔH_R. n is the so-called interaction index, which can be interpreted as a measure of the sensitivity of the vaporisation rate of the propellants to the pressure fluctuations p'. The time lag τ is called sensitive time delay and considers the intrinsic interaction between pressure oscillation and hot gas production variation inside the combustion zone. If n_{sim} is introduced for the product $\Delta H_R \cdot n \cdot \dot{m}_v / p_c$ the heat release model (3) reduces to

$$\dot{q}'(t, x) = n_{sim}\left(p'(t, x) - p'(t - \tau, x)\right). \tag{4}$$

Numerics

To solve the APE system for the three-dimensional combustor geometry, the computational aeroacoustics code PIANO (Perturbation Investigation of Aeroacoustic Noise) is used. PIANO is developed by the Deutsche Zentrum für Luft- und Raumfahrt (DLR) in Braunschweig (Germany). For spatial discretisation a fourth-order dispersion relation preserving scheme (DRP-Scheme) [10] for curvilinear block-structured grids is employed. Time integration is based on a low dissipation and low dispersion Runge-Kutta scheme (LDDRK-Scheme). Slip walls are realised via the ghost point concept of Tam & Dong [11], whereas acoustically non-reflecting boundary-conditions are implemented as “radiation boundary condition” according to Tam & Webb [10]. To suppress non-physical short wave components, the solution is filtered using a Padé-filter [12].

TESTCASE

Geometry

The approach was tested for two different configurations. The first one was chosen in analogy to the configuration presented in [5]: The test geometry has cylindrical shape, and boundaries are defined as slip walls. As the nozzle is not included in the computational domain, a proper boundary condition must be selected at the end of the cylindrical domain. Applying the nozzle admittance condition as proposed by Marble & Candel is similar to setting a slip wall [5]. Hence no losses of acoustic energy over the boundaries can occur.

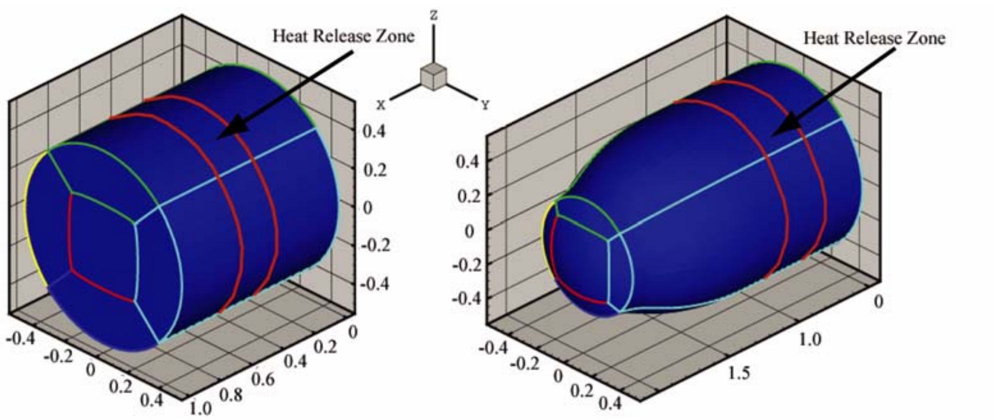

Figure 1: Geometry of testcases

The second case corresponds to the real geometry of a rocket combustor. The geometry includes the convergent part of the nozzle, which is cut off near the sonic line. As acoustic perturbation can not propagate upstream against sonic conditions, an acoustically non-reflecting boundary condition was implemented at the nozzle throat.

Both geometries are given in dimensionless form. The contours are shown in Figure 1. In both cases the heat release zone is localised between x = 0.2 and x = 0.4 and uniformly distributed over the cross section of the chamber.

Mean Flow Properties

Codes of McBride and Gordon [13] were used to estimate the mean thermodynamic quantities in the rocket engine. In the first case the fluid properties were assumed to be of homogeneous composition and temperature. The mean flow velocity was set to zero.

To calculate the mean flow properties for the second geometry, a compressible, two-dimensional, non-viscid CFD-simulation was carried out, where the fluid properties were chosen according to McBride and Gordon [13]. Figure 2 shows the Mach-number distribution in the convergent part of the nozzle. The typical curved shape of the sonic line is clearly visible.

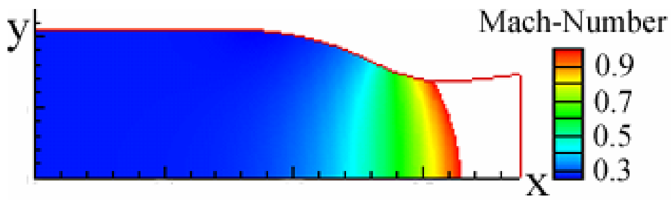

Figure 2: Mach-number distribution in the nozzle (CFD-Result)

For the acoustic simulations non-dimensional quantities were applied. All results in this paper are given in a dimensionless form, which will be indicated by *.

Initial Conditions

As initial condition an acoustic pressure pulse based on the Gaussian function

$$p'(x_i,0) = p_{max} \exp\left[-\ln 2 \cdot \frac{\left(x_i - x_i^c\right)^2}{b^2} \right] \qquad (5)$$

is used. x_i designates the coordinates of the actual position in the flow field, x_i^c the coordinates of the centre of the pressure pulse, p_{max} the magnitude of the pulse, and b the half-value radius of the Gaussian function. Figure 3 shows the shape of two different pressure pulses as a function of the coordinate x*.

As the approach is based on linear equations only frequencies can be excited, that are already present in the initial condition. The frequency content of the initial condition can be checked by fourier analysis. The smaller the half value radius of the pulse is chosen, the more frequencies are contained in the pulse, but the corresponding amplitudes get smaller. The expected oscillation frequencies lie below f* =2 and both pulses shown should be able to excite the corresponding modes in the simulation.

The simulations were carried out with a pressure pulse localised at the point $x_1 = x^* = 0.5$, $x_2 = y^* = 0.3$, $x_3 = z^* = 0.2$, with a half-value radius of b= 0.1 (see blue curve in Figure 3) and a magnitude of $p_{max} = 0.01$.

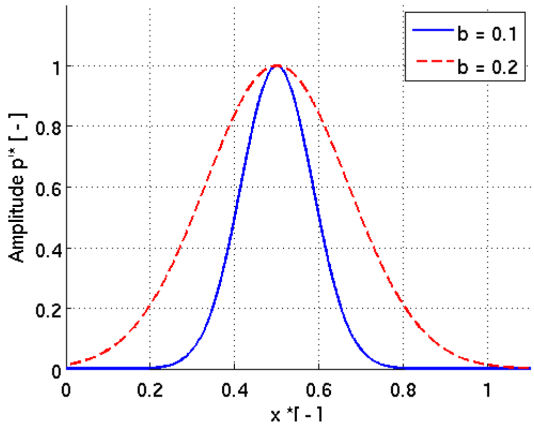

Figure 3: Shape of pressure pulse (1-D view)

RESULTS

The time lag τ* was varied between τ* = 0.5 and τ* = 1.1. Analysis of the Rayleigh-Integral shows that for theses values especially the first tangential mode and some higher order modes like the first tangential-longitudinal mode should be excited [5].

As result of the calculation the onset of self-excited oscillations is obtained. As already observed in [5] the first tangential mode is dominant in the case of the purely cylindrical first test case. Its shape is shown in Figure 4. In the heat release zone the mode shape seems to be disturbed. This can be explained by the high values of the source term for this kind of configuration. As no acoustic losses are present in this configuration, the cylindrical combustion chamber is always unstable and very high oscillation amplitudes can be reached.

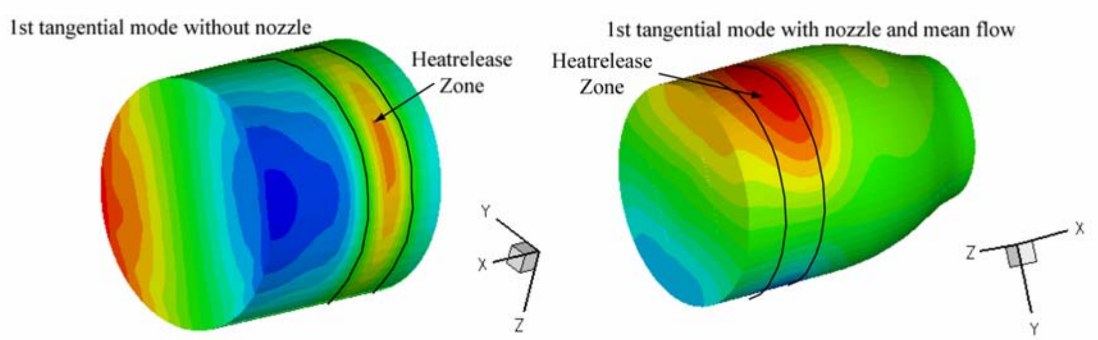

Figure 4: shape of the first tangential mode for both configurations

In the second case with mean flow and nozzle geometry, the first tangential mode prevails again. As shown in Figure 4, the shape of the oscillation mode is clearly three-dimensional. The mode is mainly localised in the cylindrical part of the geometry, whereas oscillation amplitudes decline quickly in the convergent part of the nozzle.

In contrast to the cylindrical geometry, acoustic losses through the nozzle throat occur. Therefore, when using the same values for the interaction index n_{sim} as in the case of the cylindrical geometry, oscillation amplitudes decay with time. To obtain an unstable behaviour, very high values of the interaction index have to be used. Growing oscillation amplitudes require that the acoustic energy added to the acoustic field by the flame exceeds the acoustic losses through the nozzle.

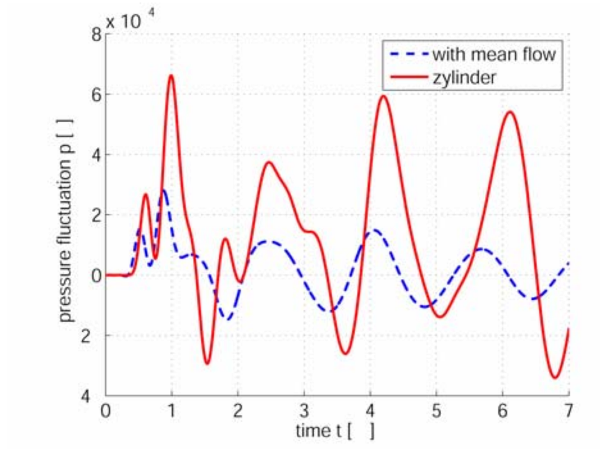

Figure 5: Temporal evolution of the oscillation in one point

Comparing the temporal evolution of the oscillation in the same point ($x^* = 0.7$, $y^* = 0.2$, $z^* = 0.2$) for both configurations ($n_{sim} = 2.8$, $\tau^* = 0.9$) shows that the results are substantially different. For the cylindrical geometry an increasing amplitude is obtained, whereas the amplitude in the case of the nozzle geometry decays. For both cases the same oscillation mode is observed, yielding similar frequencies, but the phase of both temporal signals is different. This can be explained by the fact that the mean flow modifies the propagation velocity of the acoustic disturbances. This results in a modified propagation time for the disturbances from their initial position to the heat release zone, where they are amplified.

CONCLUSIONS

A time dependent simulation of the onset of combustion instabilities for two different setups, both based on the same rocket engine, was carried out. The first test case corresponds to the classical configuration with zero mean flow and the representation of the nozzle by a constant nozzle admittance boundary-condition. The second test case incorporates the convergent part of the nozzle into the computational domain and includes mean flow effects.

For both cases, the first tangential mode is obtained as prevalent oscillation mode. However, the difference of the oscillation amplitudes is important: configurations that are unstable for the classical setup yield decaying oscillation amplitudes for the case with nozzle and mean flow. This indicates that the use of the constant nozzle admittance overpredicts oscillation amplitudes and can only be considered as a worst case for the stability layout of rocket chambers. For more accurate results as well as for the determination of reliable stability safety margins, the exact geometry and the mean flow behaviour should be taken into account.

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