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Multidisciplinary simulation of a regeneratively cooled thrust chamber of liquid rocket engine: Turbulent combustion and nozzle flow

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ABSTRACT

The present study proposed a unified framework to simulate multi-physical processes which are crucial for trade-off design of liquid rocket thrust chambers among propulsive performance, regenerative cooling, and pressure budget. As the first part, a turbulent combustion model based on the flamelet approach was developed to effectively incorporate detailed chemistry of high hydrocarbon fuel, turbulent mixing, enthalpy loss, and pressure variations within the nonadiabatic nozzle flow. In order to correctly capture the convective heat transfer and viscous friction in the turbulent boundary layer at the chamber wall, an advanced low-Reynolds number turbulence model is adopted in an axisymmetric compressible RANS solver, which is interactively coupled with a cooling analysis module for the conjugate heat transfer and hydraulics through the regenerative cooling channels. The present method has been applied to an actual regeneratively cooled thrust chamber and compared with measurement of hot-firing tests in terms of specific impulse, characteristic velocity, and thrust coefficient. Based on the numerical results, the effects of additional fuel cooling injection and wall friction on the propulsive parameters are discussed.

1. Introduction

The role of thrust chamber in a liquid propellant rocket engine (LPRE) is to burn the propellants provided by the feed system in the combustion chamber, to subsequently accelerate the combustion gas to supersonic velocities through the nozzle, and to eventually provide a propulsive force to the engine and the vehicle. The design of thrust chambers has to meet simultaneously many conflicting requirements such as high performance, reliable cooling, limited pressure budget, and minimal weight as well as structural safety [1]. Among them, the cooling design is very important for structural integrity and lifetime. At the same time, it has been a major challenging task because the thermal environment is extremely severe due to high velocity flows with adiabatic flame temperature exceeding 3500 K within the combustion chamber. Furthermore, one of the technical trends in LPRE development is to increase chamber pressure for the advantages of higher specific impulse, more compactness, and relatively higher nozzle expansion ratio [2]. Elevating the chamber pressure makes the cooling of thrust chamber more difficult because the heat transfer is approximately linearly increased with the chamber pressure [2].

Regenerative cooling has long been a standard method [1], which circulates one of the propellants through cooling passages inside the chamber wall before it is fed into the manifold of the mixing head. The coolant absorbs the heat from the combustion gas by the forced convective heat transfer and decreases the wall temperature to an acceptable level. The increase in the coolant velocity enhances the cooling capability, and at the same time increases pressure loss through the cooling passage significantly, which has negative influence on the development of turbo-pump unit of the engine system.

In order to supplement the regenerative cooling, many of modern pump-fed LPREs have adopted additional cooling method in which one of the propellants (usually fuel) is injected near the chamber wall. Then, the off-stoichiometric mixture forms a relatively low temperature gas layer between the chamber wall and high temperature core flow, and effectively reduces the thermal load on the wall. However, it leads to relatively large loss in specific impulse due to incomplete mixing and combustion [3].

There exists no universal rule for the best cooling design of a given thrust chamber because it depends heavily on many considerations such as propellant combination, chamber pressure, thrust chamber configuration, and the engine system design [1]. In this context, development of reliable tool for the trade-off analysis is very desirable to reduce trial-and-error during new LPRE

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Nomenclature

A_t	nozzle throat area (m^2)
C_F	thrust coefficient
c^*	characteristic exhaust velocity (m/s)
c_p	constant pressure specific heat (J/kg K)
F	thrust force (N)
g	acceleration of gravity (=9.8066 m/s^2)
h	specific enthalpy (J/kg)
I_{sp}	specific impulse (s)
k	turbulent kinetic energy (m^2/s^2)
$\dot{m}$	total mass flow rate of propellants (kg/s)
P	probability density function
P_{cns}	total pressure at the nozzle entrance (Pa)
p	pressure (Pa)
T	temperature (K)
t	time (s)
Y	mass fraction of chemical species
y_p^+	normalized wall distance of adjacent grid
Z	mixture fraction

Greek symbols

Γ	normalized parameter of the lookup table
ϵ	dissipation rate of turbulent kinetic energy (m^2/s^3)
ζ	normalized enthalpy variable
ρ	density (kg/m^3)
Φ_ϕ	multidimensional lookup table for property ϕ
ϕ	thermochemical properties of combustion gas
χ	scalar dissipation rate (s^{-1})
$\dot{\omega}$	chemical mass production rate (kg/m^3 s)

Subscripts

ad	adiabatic condition
ch	combustion chamber
fu	fuel (kerosene)
k	k_{th} chemical species
max	maximum limit of enthalpy
min	minimum limit of enthalpy
ox	oxidizer (oxygen)
S.L.	sea level
st	stoichiometry

Superscripts

l	lower limit of propellant temperature
sfl	steady flamelet solution
u	upper limit of propellant temperature

development which otherwise has to rely mainly on empirical design rules and experimental verification of high-cost hot firing tests. For this goal, it is necessary to model the combustion and cooling processes relevant to regeneratively cooled thrust chamber in a coupled and efficient manner.

Remarkable progresses [4–7] have recently been made for simulation of transcritical or supercritical turbulent flames formed in rocket injectors with focus on thermodynamic non-idealities and transport anomalies [8], which are critical at high pressure conditions relevant to liquid rocket thrust chambers. Especially, the numerical results of LES (Large Eddy Simulation) allow deeper insight underlying complex multi-physical phenomena and their interactions. However, three-dimensional simulation involving details of the individual injector seems not to be feasible for analysis of actual thrust chambers because they have usually hundreds of injectors. Moreover, most of the current models are devoted to simulation of precise flame structures of gaseous fuel (hydrogen or methane) and liquid oxygen near single coaxial injector under adiabatic assumption without any heat loss. Therefore, the turbulent combustion model has to be further developed, in particular, to incorporate detailed chemistry of high hydrocarbon fuel (i.e., kerosene) [9] and nonadiabatic flame due to wall heat losses [10].

From a practical point of view, noticeable efforts have continuously been made by researchers of Astrium Space Transportation. They have developed a combustion and heat transfer prediction tool, so called ROCFLAM, and applied successfully to resolve engineering problems such as performance prediction, nozzle contouring, and cooling design evaluation encountered during the European LPRE development programs [11]. The Astrium CFD tool utilizes an axisymmetric spray-combustion Navier–Stokes code with various physical models, which have long been established and validated based on their own experimental database. However, it has been developed originally for storable bipropellants (MMH/NTO) [12] and cryogenic propellants (LOx/H_2) [13], although extension to a uni-element subscale combustor with kerosene and gaseous oxygen is recently reported [14].

This study is aimed to develop a numerical methodology to systematically evaluate effects of design parameters on propulsive performance, cooling, and hydraulic characteristics of a regeneratively cooled thrust chamber with the propellant combination of kerosene and liquid oxygen. For this goal, the solution algorithm adopted in this study consists of two separate parts which are interactively coupled with each other, as schematically illustrated in Fig. 1. Firstly, an axisymmetric compressible flow solver is employed as a numerical framework to simulate all Mach number reacting turbulent flows within liquid rocket thrust chambers. The second one is related to the conjugate heat transfer from the combustion gas to the coolant flow and hydraulics within the regenerative cooling passages. The modeling issues associated with the former are dealt with in this paper while the latter will be reported in more detail elsewhere. In order to assess the predictive capability, the present method has been applied to an actual regeneratively cooled thrust chamber and validated against measured data obtained through hot firing tests.

In the present study, the turbulent combustion modeling is devised based on a recently developed flamelet model [9] to effectively account for nonequilibrium chemistry of the kerosene fuel. Special effort is devoted to extension of the flamelet model to nonadiabatic nozzle flows where the sensible enthalpy and pressure drastically decrease. The turbulent combustion model is incorporated in a compressible flow solver which adopts an advanced low-Reynolds turbulence model to correctly capture the convective heat transfer and friction occurred in turbulent boundary layer on the chamber wall. Based on the numerical results, the effects of cooling injection and friction loss on the propulsive performances are discussed in terms of specific impulse, characteristic velocity, and nozzle thrust coefficient.

2. Turbulent combustion modeling

2.1. Flamelet-based combustion model

The chemical equilibrium assumption has often been used to predict the propulsive performance of liquid rocket thrust chambers and justified by the high pressure and high temperature combustion environment. Since the kerosene fuel under consideration