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MULTIDISCIPLINARY OPTIMIZATION OF COMBUSTION CHAMBER FOR SMALL GAS TURBINE ENGINES

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Abstract

The comprehensive multidisciplinary optimization of combustion chamber for small gas turbine engines is summarized and presented. The in-house post processing methodology and commercial CFD solvers are proposed and verified for the NASA low-cost turbine engine of 650 lbf sea-level static thrust with the key parameters such as combustion efficiency, total-pressure loss, exit-temperature profile, reference velocity. The comparison of experiment data with CFD results (1/12, 1/2 and full scale of combustor chamber) is presented for the radial and circumferential exit temperature profile. The analysis results show very good agreements with the experimental data for the 1/2 and full scale combustion chamber at the sea-level test conditions from the NASA TM X-2857 report.

The proposed methodology is implemented for the 1st combustor prototype while satisfying the requirements. The fabrication and test results of 1st prototype are obtained and enhanced for the reliable aero-thermal analysis and the next design loop of combustor. The 2nd combustor configuration derivatives are quickly presented by implementing the verified aero-thermal analysis and customized DoE to the small changes of diffuser dimension and mini-tank in the small gas engine research and development phase. The proposed methodology shows the feasibility and effectivity of the cost saving and low turnaround time solution for the small gas engine development stage.

Keywords: Multidisciplinary Optimization, Combustion chamber, Aero-thermal Analysis, Small Gas-Turbine, CFD, combustion test data.

1. Introduction

The significant characteristics of combustion are unsteady and instability. Traditionally, the gas turbine combustor design and development have extensively used the empirical/analytical relations corrected by the exhausted scale and full scaled combustor with the sub-components and components tests data. This methodology has been successfully applied in many combustor developments for small and big engines [1]. However, it shows the limitations in the scaling combustors, big jumps in technology levels for combustor temperature rise, cycle pressure ratio, combustor performance and durability levels, and low-fidelity analysis model for the novel or revolutionary combustor concepts [2]. Recently, Computational Fluid Dynamics (CFD) is widely applied in the combustion analysis and combustor design in big engine development companies such as Rolls–Royce, GE, Pratt & Whitney, MTU and Hanwha Aero-engines with the help of advanced high performance computing system (HPC). Three main models for turbulent analysis are Reynolds-averaged Navier–Stokes simulations (RANS), large-eddy simulations (LES), and direct numerical simulations (DNS) [3]. The RANS has been mainly used in the gas turbine industry due to the fast turnaround time and estimated and the exit temperature profile for the design guidance reliably. In addition, Huram et al. proposed the Computational

Combustion Dynamics (CCD) codes for interpreting test results, confounding invariably, and guiding for combustor design process [4]. This CCD is used with the semi-analytical mechanistic (SAM) during the combustor design and development process for CFM56 dual annular combustor, CF6-80 low-emissions single annular combustor, and GE90 DACII in GE aircraft engines known as the most conventional "cut-and-try" design practice [5]. Multidisciplinary Design Optimization (MDO) is widely implemented to support many sophisticated system such as UCAV, UAV, and aircraft system [6-8]. The combustor system design and development phases are required to consider the multidisciplinary analysis models such as aero-thermal, structure, material, acoustics, and emission analysis in order to support the design guidance during the preliminary, detailed design stage and refining the engine models during the ground and flight testing phases for optimizing the final engine models.
The multidisciplinary optimization of combustion chamber for the small and low-cost gas turbine process is proposed by implementing the multidisciplinary optimization approach with the initial sizing combustor in-house code, verified aero-thermal, structural analysis to provide the fast and reliable combustor configuration. In addition, the proposed process also supports to enhance the combustor test data for improving the next version of combustor.

2. Multidisciplinary optimization of combustion chamber for small gas turbine process

Multidisciplinary optimization of combustion chamber for small gas turbine process is proposed and addressed in the Figure 1. The main steps are described as follows

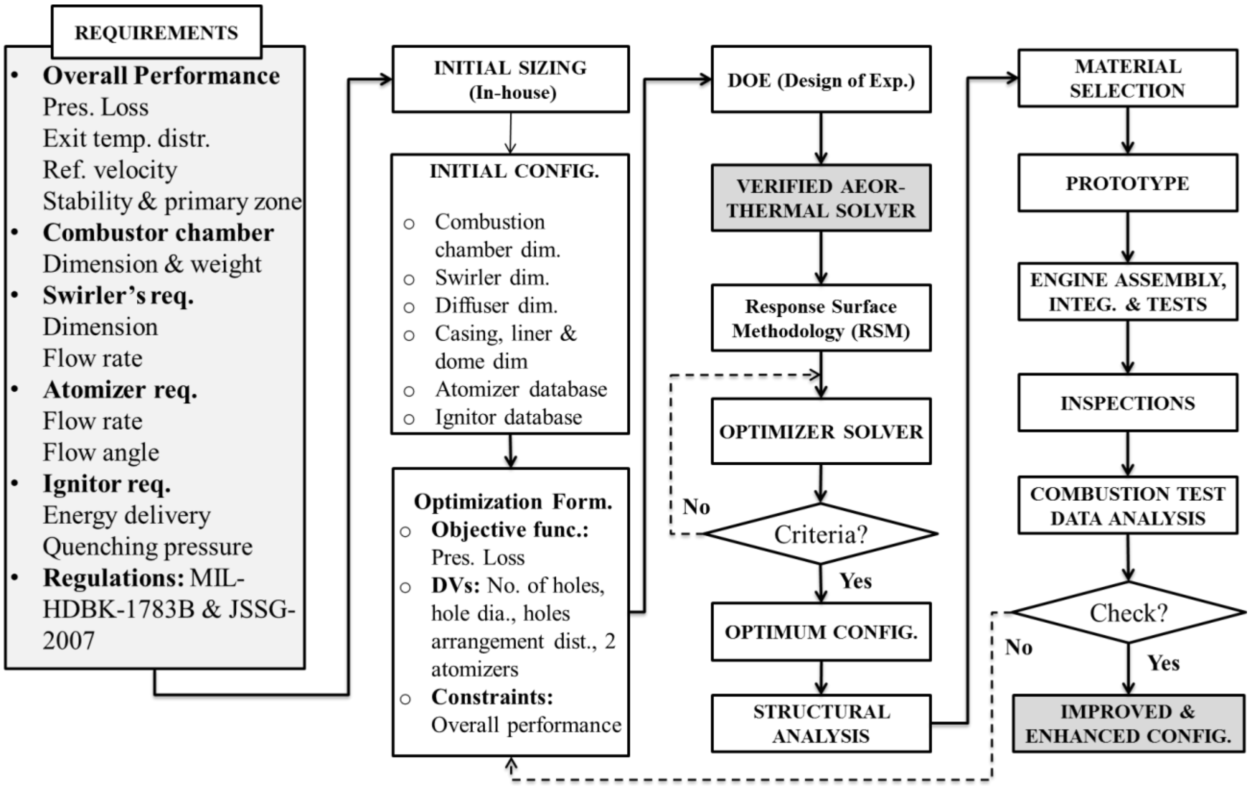


Figure 1 – Multidisciplinary optimization of combustion chamber for small gas turbine process

Step 1: Requirements
Combustor design requirements are needed to address for overall performance, dimension & weight limits, swirler, atomizer, ignitor and regulations (MIL-HDBK-1783B & JSSG-2007) as shown in Figure 1 for designed combustor system.

Step 2: Initial sizing
The in-house code based on the empirical relations to generate the initial configuration for the swirler, chamber, diffuser, casing, liner and dome dimension. Based on the requirements of atomizer and ignitor, the database of COTS atomizer and ignitor are built.

Step 3: Optimization formulation
The objective function, design variables and constraints of combustor system are required to define for the multidisciplinary optimization

Step 4: DoE
The customized design of experiment points are used to run for the combustor configuration combination cases based on the design variables of number of holes, hole diameter, hole arrangement, and 2 COTS atomizers.

Step 5: Verified aero-thermal solver
Aero-thermal analysis is used to predict the aerodynamics and thermal analysis in the combustor with the different atomizers, and different combustor configuration.
The RANS K-eps model is used for the fast turnaround time. The calibration of aero-thermal model is performed on the existing NASA combustor. The verification of solver will be presented in the next section. The solver in the ANSYS commercial S/W is used [9].

Step 6: Response Surface Methodology (RSM) and Optimizer
RSMs for objective function and constraints of combustor performance are generated for the second order and the adjective squared is check to ensure the reliable of models.
The optimum configuration is obtained with the full considerations of requirement's satisfaction

Step 7: Structural analysis and material selection
The structural analysis is performed on the optimum combustor configuration to ensure the thermal loads with the guidelines in MIL-HDBK-1783B for the thermal loads limits [10,11]
The material selection is proposed in the engine material database
The prototype of combustor is fabricated

Step 8: Engine assemble, integration and tests
The proposed combustor is assembled, integrated and tested. The data acquisition related to combustor is obtained including pressure ratio Inlet/outlet, temperature distribution at outlet according to the different RPM.
The combustion inspection is performed to check the visualization of primary zone, secondary and dilution zone of designed combustor.

Step 9: Improved and enhanced combustor
After combustor test data analysis and visualization check, if the requirements are not satisfied or marginal, the changes in optimization formulation are made to start over for the process.

Step 10: Improved and enhanced configuration
The final combustor configuration is obtained

3. Verification of CFD solvers for combustion chamber

The in-house analysis using CFD and post processing methodology is proposed and verified for the NASA low-cost turbine engine of 650 lbf sea-level static thrust with the key parameters such as combustion efficiency, total-pressure loss, exit-temperature profile, reference velocity, heat-release rate, emission index and smoke number as shown in the Figure 2 for both engine model and flow chart. The NASA combustor model including case, liners, diffuser, atomizer and swirler is generated in the

Figure 3. The comparison of experiment data with CFD results (1/12, 1/2 and full scale of combustor chamber) is presented in Figure 4 for the radial and circumferential exit temperature profile. The analysis results show very good agreements with the experimental data for the 1/2 and full scale combustion chamber at the sea-level test conditions from the NASA TM X-2857 report. The calibration of methodology is made during the calculation of 1/12, ½ and full scale of combustor compared with the experiment data. The thermal visualization of NASA combustor is shown in the Figure 5. Hence, the verification of aero-thermal solver is performed and calibrated to use reliably for the multidisciplinary combustor of the small gas turbine engine with the low turnaround time.

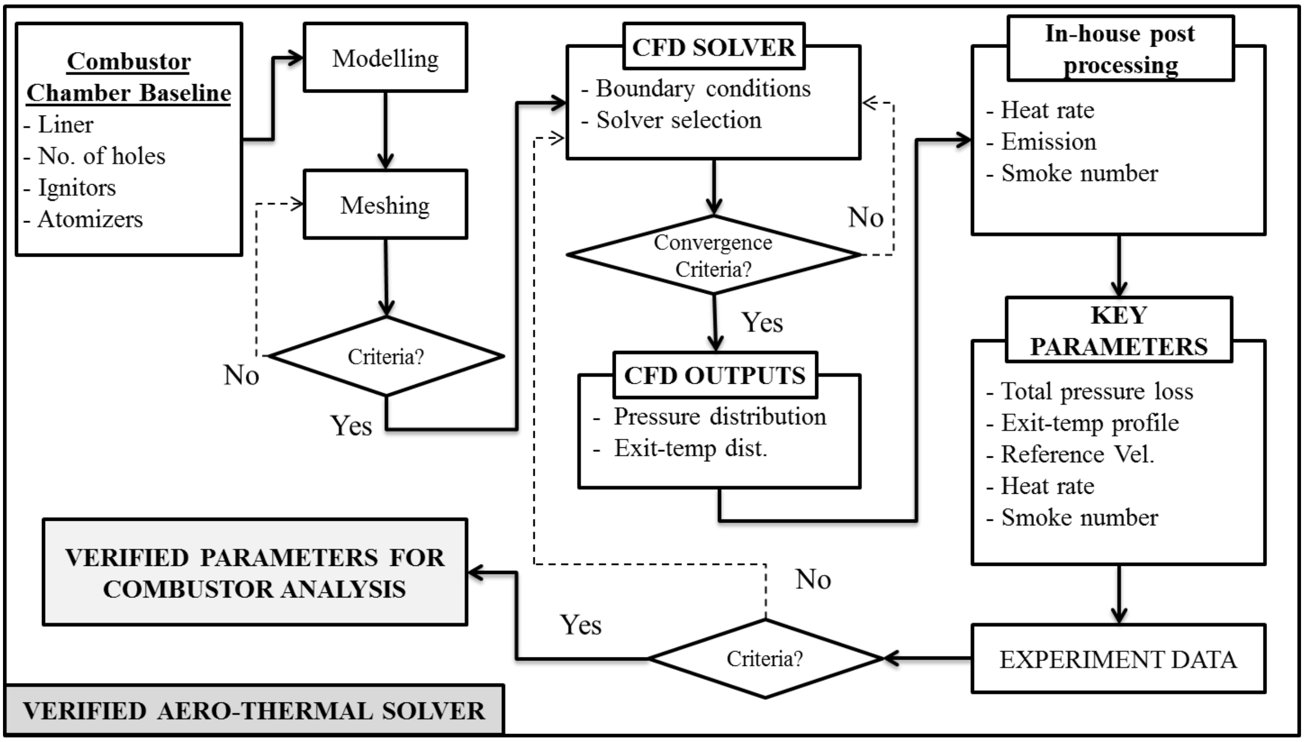


Figure 2– Verification of CFD solvers for combustion chamber process

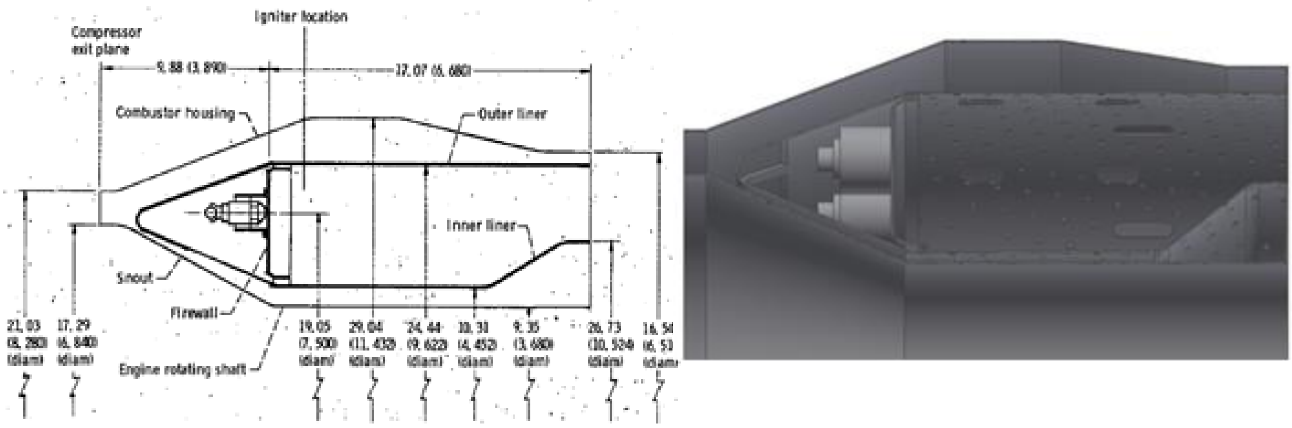


Figure 3 – The combustor chamber of NASA low-cost turbine engine modelling

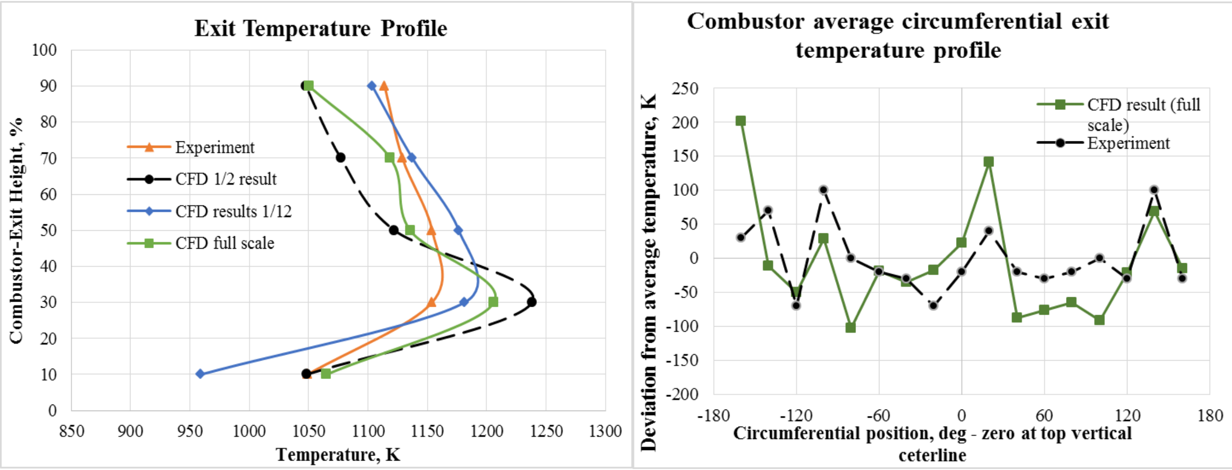


Figure 4 – Validation results with TM X-2857 report

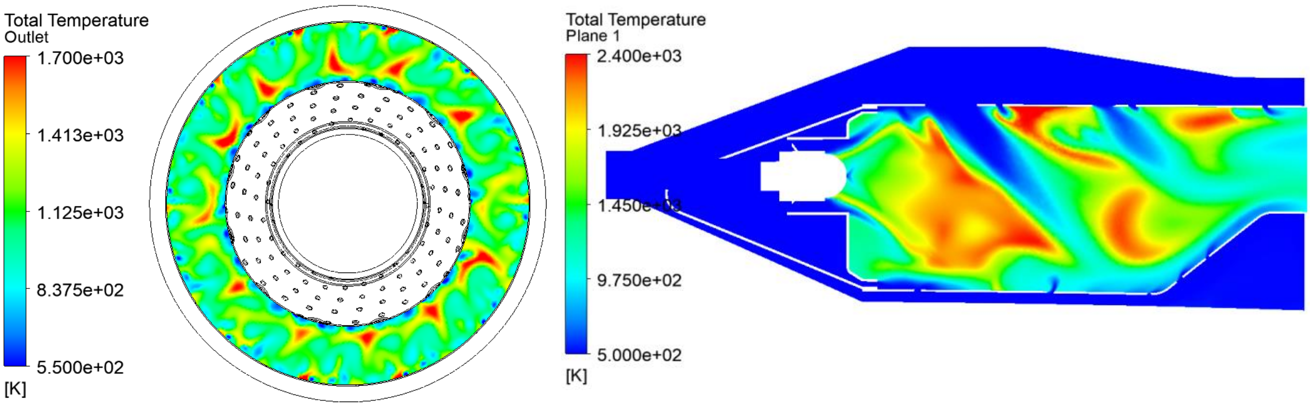


Figure 5 – Combustion chamber visualization of NASA low-cost turbine engine

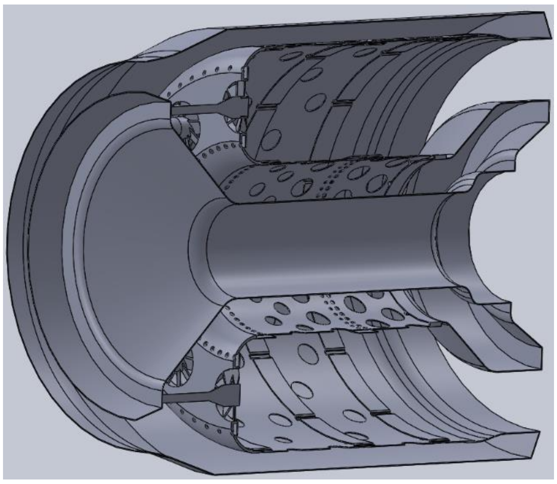
4. Case study: Improvement and enhancement of combustor for small turbine engine

4.1. 1st prototype combustor for small gas turbine engine

The proposed process is performed for the practical combustor design of the small turbine engine applications. The combustor performance requirements are shown in the Table 1.

Table 1: Requirements of combustor subsystem

No.	Parameters	Required Value	Unit	Conditions	Note
1	Total Pressure Loss	≤ 7	%	Design Point	
2	Combustion Efficiency	≥ 95	%	Design Point	
3	Radial Pattern Factor	≤ 25	%	Design Point	
4	Exit average temperature	1150±20	K	Design Point	
5	Peak temperature	≤ 1400	K	Design point	



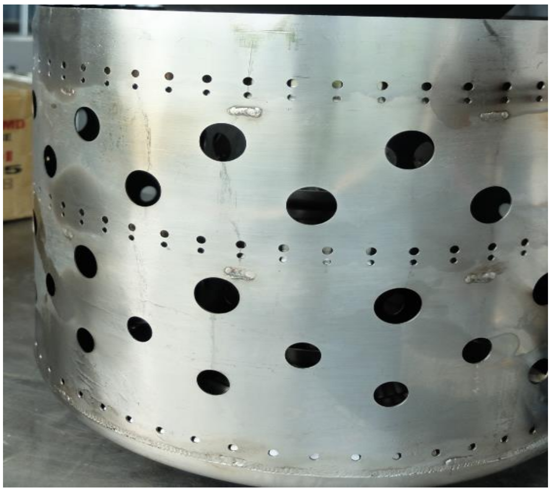


Figure 6 – 3D optimum configuration

Figure 7 – 1st prototype fabrication

The initial sizing in-house code and multidisciplinary optimization are performed with the given requirements to provide the 3D optimum configuration in the Figure 6. The 1st prototype fabrication is shown in the Figure 7. The results of combustor performance are shown in the Table 2 while satisfying the requirements.

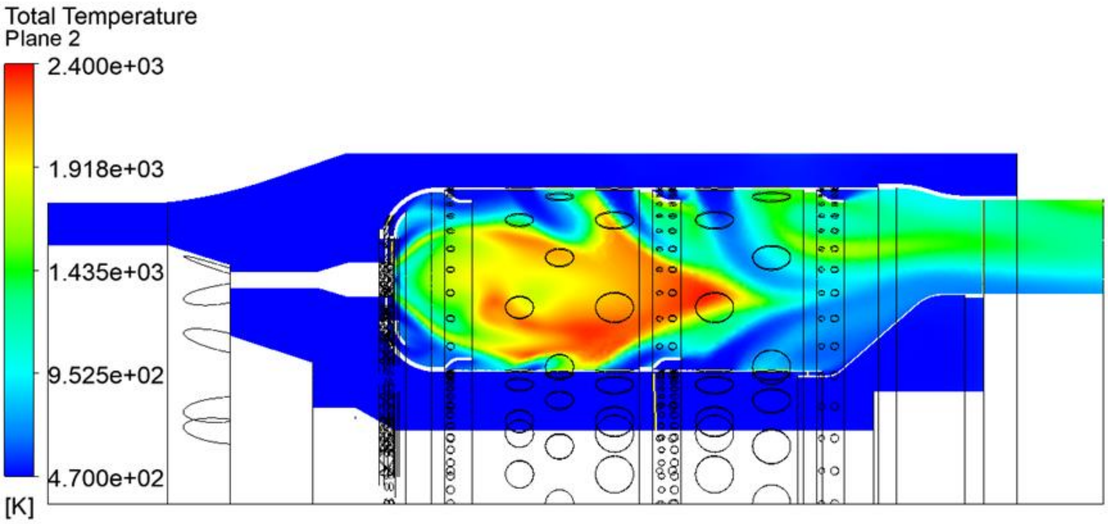


Figure 8 – Optimum combustion chamber thermal distribution

Table 2: Results of 1st prototype combustor

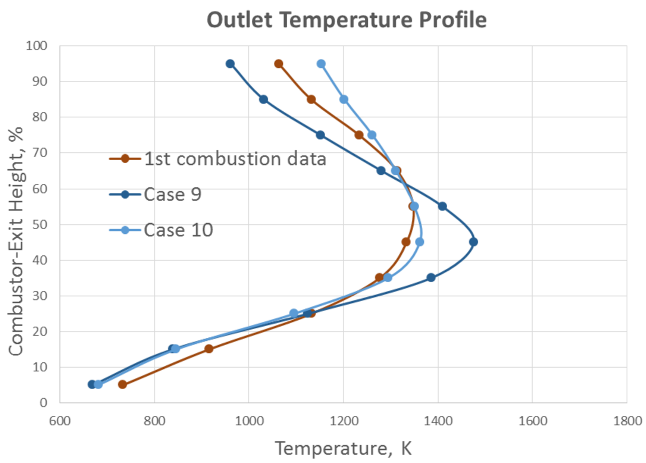
No.	Parameters	Value	Required Value	Evaluate
1	Total Pressure Loss	5.3 %	≤ 7%	Satisfied
2	Combustion Performance	95 %	≥ 95%	Satisfied
3	Radial Pattern Factor	24 %	≤ 25%	Satisfied
4	Exit average temperature	1150 K	1150±20 K	Satisfied

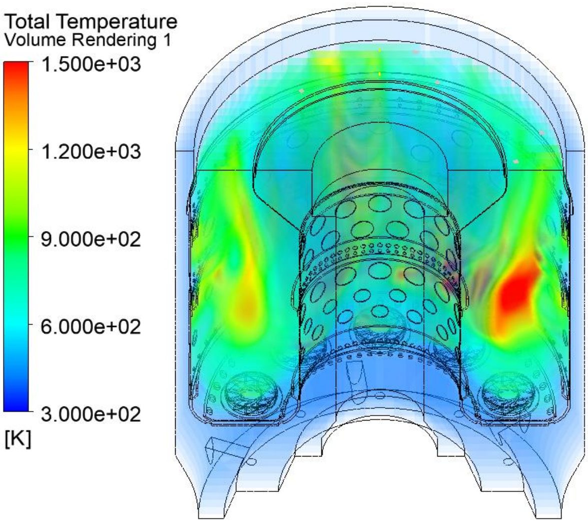
Figure 9 – Combustor manufacture and tests results

The optimum combustor is assembled and integrated to engine for testing as shown in the Figure 8. The combustor test data and visualization of 1st combustor are shown in the Figure 9. The proposed methodology shows the feasibility and effectivity of the cost saving and low turnaround time while developing the small and low-cost turbojet engine.

The combustor exit temperature profile test data are obtained with the two temperature sensors at the two exit locations of combustor as shown in the Figure 10. The 1st and 2nd temperature sensors show good agreements with the experiment data which is approximately the 22K and 30K temperature difference respectively with the calibrated CFD analysis solvers used for the 1st combustor design.

The CFD visualization of 1st combustor shows similar images captured in the primary zone, secondary zone and dilution zone while testing with the 1st combustor configuration.





(a) 1st configuration outlet temperature comparison (b) 1st combustor configuration CFD visualization

Figure 10 - 1st combustor configuration comparisons with the experiment data

4.2 2nd combustor optimum configuration

The new requirements are provided for the 2nd combustion configuration to satisfy the changes of diffuser dimension and mini-tank, therefore the 2nd combustor configuration design is processed to propose while satisfying the new requirements.

The 2nd combustor requirement is set as the base case as shown in the Table 3. The sensitivity combustor analysis is performed according to the changes as shown in the Table 3 with 8 cases. The CFD analysis results are performed without the thermal analysis for 8 cases in the Table 3 to save the computational loads. The good cases are obtained and indicated for the requirement in Table 3 which is case 1, case 4, case 6 and case 7.

The additional case 9 and case 10 are proposed with the combination of these cases to perform the aero-thermal analysis as follows:

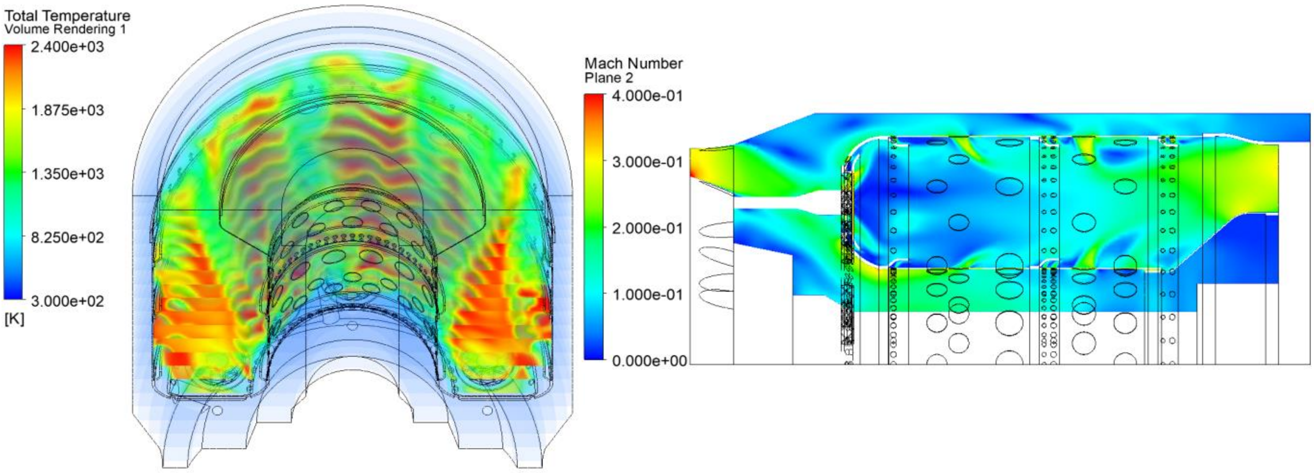
Case 9 is composed of case 1, case 6 and case 7

Case 10 is composed of case 1, case 4, case 6 and case 7.

Table 3. Number of analysis cases for the 2nd combustor sensitivity analysis

<table>
<tr><th>No.</th><th>Case</th><th colspan="2">Changes</th><th>Requirement</th></tr>
<tr><td>0</td><td>BASE CASE</td><td colspan="2">Smooth between diffuser and stator</td><td>Reference values</td></tr>
<tr><td>1</td><td>Primary air hole moves forward</td><td colspan="2">Foward: 9 mm</td><td>To reduce the burining vortex area</td></tr>
<tr><td>2</td><td rowspan="2">Reducing the ratio: Aft/Aref</td><td rowspan="2">Aft/Aref = 0.67</td><td>Diffuser R1 =118.5</td><td rowspan="2">Melconian and Modak recommendation, 1985 [12]</td></tr>
<tr><td>3</td><td>Diffuser R1 = 117</td></tr>
<tr><td>4</td><td rowspan="2">Swirler blades incidence</td><td colspan="2">θ = 60°, %m = 12%</td><td rowspan="2">To increase the vortex at the primary zone</td></tr>
<tr><td>5</td><td colspan="2">θ = 65°, %m = 12%</td></tr>
<tr><td>6</td><td>Change the radius under the Dome</td><td colspan="2">Increase the radius under the Dome
from R = 28.5 mm to R = 40 mm</td><td>Smoothing the airflow into the inner liner</td></tr>
<tr><td>7</td><td>To reduce the hole at the Dilution</td><td colspan="2">To reduce from D = 19 mm to D = 16 mm and to maintain the No. of hole</td><td>To reduce the airflow into the dilution area. To increase the airflow into the primary and secondary regions.</td></tr>
<tr><td>8</td><td>To move Swirlers after Atomizers</td><td colspan="2">Swirler moves backward after the Atomizer (Resemble to NASA combustion [1])</td><td>Evaluation the airflow</td></tr>
</table>

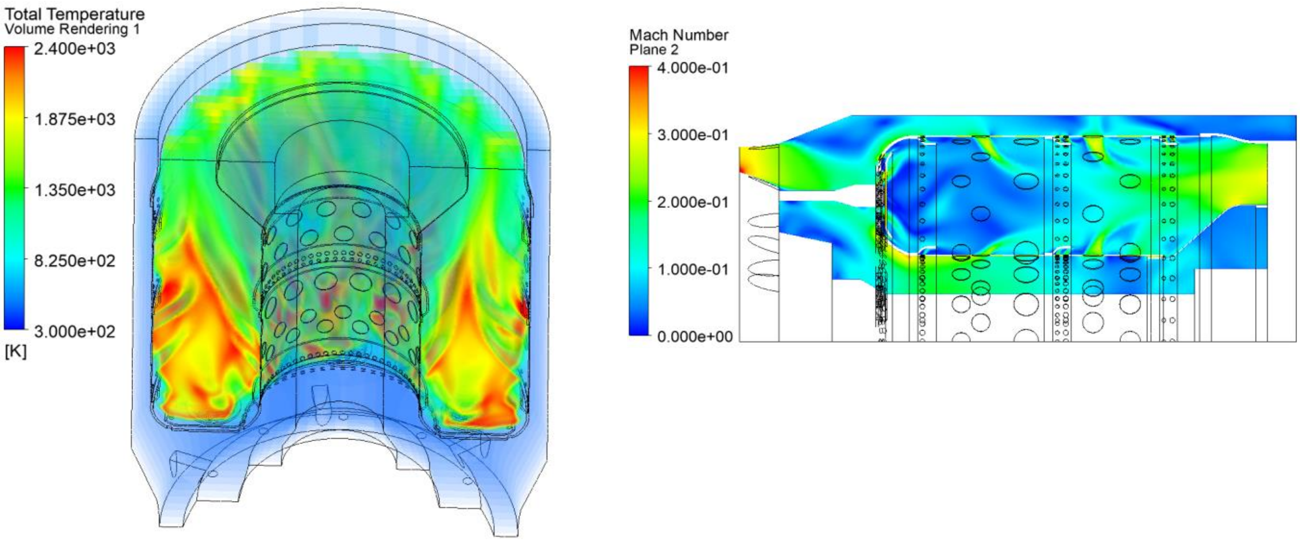
The simulation results of case 9 and case 10 are shown in the Figure 11 and 12 respectively. The case 9 shows the long flame along axial direction of combustor as shown in the Figure 11 (a) with the existing swirler of 55 degree. Hence, the peak temperature profile of case 9 indicates to 1480K as shown in the Figure 13 in which violates the requirements in the Table 1. Therefore, the case 9 is not selected.



(a) Flame simulation results (b) Mach number distribution

Figure 11- Case 9 simulation results

The simulation results of case 10 is shown in the Figure 12 with the new swirler design of 60 degree and lower mass flow of 12% compared to 18% of existing swirler. The flame simulation results of case 10 show the shorter distance along the axial direction of combustor. It complies with the outlet temperature distribution as shown in the Figure 13 in which the main combustor requirements are satisfied. Therefore, the proposed 2nd optimum combustor configuration is addressed as the case 10.



(a) Flame simulation results (b) Mach number distribution

Figure 12- Case 10 simulation results

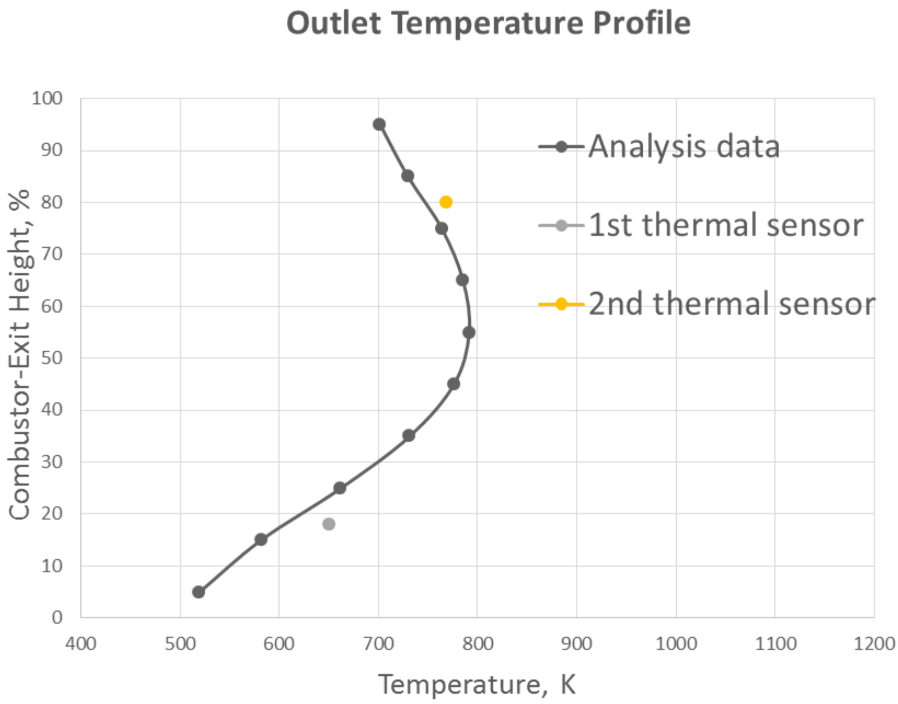


Figure 13-Outlet temperature profile comparisons of 1st combustor, case 9 and case 10

5. Conclusion and future works

The proposed process is successfully applied for the designing the 1st optimum combustor configuration and tested with the small gas engines with the low turnaround time and reliable multidisciplinary analysis.

The optimum combustor configuration, atomizer and ignitor are determined for minimizing the total pressure loss while maintaining the other performance requirements including exit temperature, combustion efficiency, radial pattern factor and the requirements from MIL-HDBK-1783B for the combustor material.

The 2nd optimum combustor configuration is presented by using the combination of customized design of experiment (DoE) points, aerodynamics analysis and aero-thermal analysis to provide the quick combustor design derivatives with the low turnaround time and reliable results enhanced by the 1st combustor configuration test data.

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