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Comparison of Effectiveness for Performance Tuning of Liquid Rocket Engine

Won Kook Cho*† and Chun Il Kim**
*Rocket Engine Team, Korea Aerospace Research Istitute
169-84, Gwahak-ro, Yuseong-Gu, Daejeon, 34133 KOREA
†E-mail:wkcho@kari.re.kr
**Department of Mechanical Engineering, University of Alberta
9211 116 St. NW, Edmonton, AB, Canada, T6G 1H9

Abstract

An analysis has been made on the performance variation due to pressure drop change at propellant supply pipes of liquid rocket engine. The objective is to compare the effectiveness of control variables to tune the liquid rocket engine performance. The mode analysis program has been used to estimate the engine performance for different modes which is realized by controlling the flow rate of propellant. The oxidizer of combustion chamber, the fuel of combustion chamber, the oxidizer of gas generator and the fuel of gas generator are the independent variables to control engine thrust, engine mixture ratio and temperature of gas generator product gas. The analysis program is validated by comparing with the powerpack test results. The error range of compared variables is order of 4%. After comparison of tuning effectiveness it is turned out that the pressure drop at oxidizer pipe of gas generator and pressure drop at combustion chamber fuel pipe and the pressure drop at the fuel pipe of gas generator can effectively tune the thrust of engine, mixture ratio of engine and temperature of product gas from gas generator respectively.

Key Words : Liquid rocket engine, Gas generator cycle, Lox/kerosene, Mode analysis, Engine Control, Thrust control, Mixture ratio control, Performance tuning

1. Introduction

A liquid rocket engine is one of the most important parts in a space launcher. Severe requirements are requested to a liquid rocket engine for satellite launching. The thrust and mixture ratio should be very precise or controllable to satisfy the orbital precision requirement and efficient propellant usage. Also simplest structured engine is frequently requested at the same time for maximum reliability of the launcher. The liquid rocket engine is usually composed of more than a thousand parts. A non-controlled engine with minimum parts can guarantee performance precision with performance tuning technique which can be realized by adjusting the propellant supply characteristics[1,2].

The control valves of the four major propellant supply pipes - combustion chamber oxidizer pipe, combustion chamber fuel pipe, gas generator oxidizer pipe and gas generator fuel pipe - can change the pressure drop of propellant. The combination of the above pressure drop of four pipes can change the engine thrust, mixture ratio and the gas generator temperature. The engine thrust is the most important parameter for satellite orbit precision. The engine mixture ratio determines the efficient propellant usage of the launcher. And the gas generator temperature must be accurate not to exceed the temperature limit of the turbine. In the

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† Corresponding Author
Tel: +82-42-860-2937, E-mail: wkcho@kari.re.kr

present paper, the effectiveness of the pressure drop of major four propellant pipes to liquid rocket engine performance is parametrically studied. The quantitative parametric analysis of liquid rocket engine with verified methodology against test data is hard to find though the rocket engine tuning is a well known concept. So the present paper can be applied to liquid rocket engine development project.

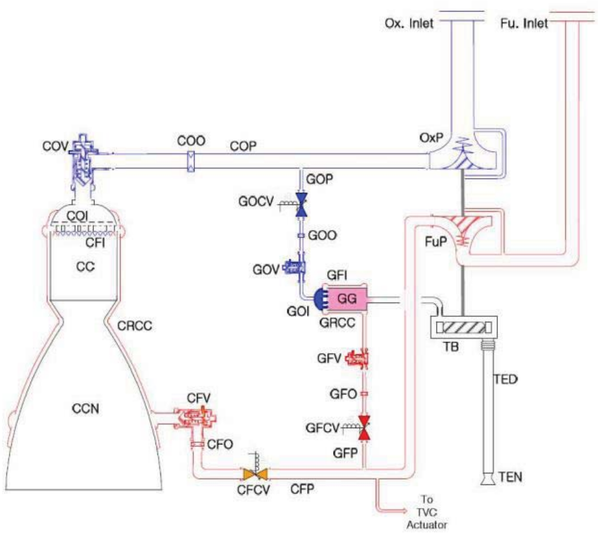

Fig. 1 Schematic of gas generator cycle liquid rocket engine[2]

2. Analysis method

The engine performance can be predicted with the combination of the performances of components. The mode analysis program is used. This program calculates the balanced pressure, mass and power condition to give engine performance for a given mode[2]. This program gives the followings;

(1) Engine performance for changed propellant supply condition.

(2) The effect of orifice change or control valve opening ratio

(3) Predicted results for malfunction of components - usually leakage or blockage

The important components of gas generator cycle engine are combustion chamber, gas generator, turbopump, valves and pipes. Fig. 1 presents the schematic of the liquid rocket engine considered in the present paper. The performances of components are obtained from experiments or analyses. Eq(1) to Eq.(15) are solved to determine the engine mode. Each equation defines the residual of the performance variable meaning the numerical difference between the obtained variables and the mathematical correlation. The equations are (1) pressure of oxidizer pump, (2) pressure of fuel pump, (3) combustion chamber oxidizer supply pressure, (4) combustion chamber fuel supply pressure, (5) mass flow rate of combustion chamber, (6) gas generator oxidizer supply pressure, (7) gas generator fuel supply pressure, (8) mass flow rate of gas generator, (9) adiabatic spouting velocity of turbine, (10) correlation between adiabatic spouting velocity and pressure ratio, (11) turbine power balance, (12) efficiency of turbine, (13) total pressure of turbine exit, (14) temperature rise of oxidizer at pump exit, (15) temperature rise of fuel at pump exit

$$R_1 = p_{out,oxp} - \left(p_{in,oxp} - A_{H,ox}\rho_{o,0}n_{tp}^2 + B_{H,ox}n_{tp}\dot{m}_{ox} + C_{H,ox}\dot{m}_{ox}^2/\rho_{o,0}\right) \quad (1)$$

$$R_2 = p_{out,fup} - \left(p_{in,fup} - A_{H,fu}\rho_{f,0}n_{tp}^2 + B_{H,fu}n_{tp}\dot{m}_{fu} + C_{H,fu}\dot{m}_{fu}^2/\rho_{f,0}\right) \quad (2)$$

$$R_3 = p_{cc} - \left(p_{out,oxp} - \sum_i \frac{\xi_{co,i}}{A_i^2}\frac{\dot{m}_{co}^2}{2\rho_o}\right) \quad (3)$$

$$R_4 = p_{cc} - \left(p_{out,fup} - \sum_i \frac{\xi_{cf,i}}{A_i^2}\frac{\dot{m}_{cf}^2}{2\rho_f}\right) \quad (4)$$

$$R_5 = p_{cc} - \frac{\eta_{c^*,cc}(\dot{m}_{co}+\dot{m}_{cf})}{C_{d,cc}A_{n,cc}\sqrt{\frac{\gamma_{cc}}{R_{cc}T_{cc}}\left(\frac{2}{\gamma_{cc}+1}\right)^{\frac{\gamma_{cc}+1}{\gamma_{cc}-1}}}} \quad (5)$$

$$R_6 = p_{gg} - \left(p_{out,oxp} - \sum_i \xi_{go,i}\frac{\dot{m}_{go}^2}{2\rho_{o,1}}\right) \quad (6)$$

$$R_7 = p_{gg} - \left(p_{out,fup} - \sum_i \xi_{gf,i}\frac{\dot{m}_{gf}^2}{2\rho_{f,1}}\right) \quad (7)$$

$$R_8 = p_{gg} - \frac{\dot{m}_{go}+\dot{m}_{gf}}{C_{d,gg}A_{n,gg}\sqrt{\frac{\gamma_{gg}}{R_{gggg}}\left(\frac{2}{\gamma_{gg}+1}\right)^{\frac{\gamma_{gg}+1}{\gamma_{gg}-1}}}} \quad (8)$$

$$R_9 = C_{ad,tb} - \sqrt{\frac{2\gamma_{gg}}{\gamma_{gg}-1}R_{gg}T_{gg}\left\{1-\left(\frac{p_{tbe}}{p_{gg}}\right)^{\frac{\gamma_{gg}}{\gamma_{gg}-1}}\right\}} \quad (9)$$

$$\mathrm{R}_{10} = \eta_{tb} - \left\{A_{e,tb}\left(\frac{p_{gg}}{p_{tbe}}\right)^2 + B_{e,tb}\left(\frac{p_{gg}}{p_{tbe}}\right) + C_{e,tb}\right\} \times \left(\frac{D_{m,tb}n_{tp}}{2C_{ad,tb}}\right)^2 - \left\{D_{e,tb}\left(\frac{p_{gg}}{p_{tbe}}\right)^2 + E_{e,tb}\left(\frac{p_{gg}}{p_{tbe}}\right) + F_{e,tb}\right\} \times \left\{\left(\frac{D_{m,tb}n_{tp}}{2C_{ad,tb}}\right)^2 + G_{e,tb}\right\} \quad (10)$$

$$\mathrm{R}_{11} = \frac{1}{2}\eta_{tb}C_{ad,tb}^2\left(\dot{m}_{go}+\dot{m}_{gf}-\dot{m}_{roll}\right)\eta_{m,tp} - \left\{\frac{\dot{m}_{ox}(p_{out,oxp}-p_{in,oxp})}{\rho_{o,1}\eta_{oxp}} + \frac{(\dot{m}_{fu}+\dot{m}_{tvc})(p_{out,fup}-p_{in,fup})}{\rho_{f,1}\eta_{fup}}\right\} \quad (11)$$

$$\mathrm{R}_{12} = \eta_{tb} - \frac{T_{gg}-T_{0,tbe}}{T_{gg}\left\{1-\left(\frac{p_{tbe}}{p_{gg}}\right)^{\frac{\gamma_{gg}-1}{\gamma_{gg}}}\right\}} \quad (12)$$

$$\mathrm{R}_{13} = p_{tbe} - \frac{\dot{m}_{go}+\dot{m}_{gf}-\dot{m}_{roll}}{A_{tbe}\sqrt{\frac{\gamma_{gg}}{R_{gg}T_{0,tbe}}\left(\frac{2}{\gamma_{gg}+1}\right)^{\frac{\gamma_{gg}+1}{\gamma_{gg}-1}}}} \quad (13)$$

$$\mathrm{R}_{14} = \Delta\mathrm{T}_{\mathrm{ox}} - \frac{1}{C_{p,ox}}\left(\frac{p_{out,oxp}}{\rho_{o,1}} - \frac{p_{in,oxp}}{\rho_{o,0}}\right)\left(\frac{1}{\eta_{oxp}}-1\right) \quad (14)$$

$$\mathrm{R}_{15} = \Delta\mathrm{T}_{\mathrm{fu}} - \frac{1}{C_{p,fu}}\left(\frac{p_{out,fup}}{\rho_{f,1}} - \frac{p_{in,fup}}{\rho_{f,0}}\right)\left(\frac{1}{\eta_{fup}}-1\right) \quad (15)$$

The variables used in the above equations are defined as followings

$$\rho_{o,0} = f(T_{in,oxp}, p_{in,oxp}) \tag{16}$$
$$\rho_{o,1} = f(T_{out,oxp}, p_{out,oxp}) \tag{17}$$
$$\rho_{f,0} = f(T_{in,fup}, p_{in,fup}) \tag{18}$$
$$\rho_{f,1} = f(T_{out,fup}, p_{out,fup}) \tag{19}$$
$$\Delta \mathrm{T_{ox}} = \mathrm{T_{out,oxp}} - \mathrm{T_{in,oxp}} \tag{20}$$
$$\Delta \mathrm{T_{fu}} = \mathrm{T_{out,fup}} - \mathrm{T_{in,fup}} \tag{21}$$
$$\dot{m}_{ox} = \dot{m}_{co} + \dot{m}_{go} \tag{22}$$
$$\dot{m}_{fu} = \dot{m}_{cf} + \dot{m}_{gf} \tag{23}$$
$$\varphi_{\mathrm{oxp}} = \frac{m_{ox}}{\frac{(\rho_{o,0}+\rho_{o,1})}{2}\frac{1}{4}\pi(D_{t1,oxp}^2 - D_{h1,oxp}^2)\frac{D_{t1,oxp}}{2}n_{tp}} \tag{24}$$
$$\varphi_{\mathrm{fup}} = \frac{m_{fu}}{\frac{(\rho_{f,0}+\rho_{f,1})}{2}\frac{1}{4}\pi(D_{t1,fup}^2 - D_{h1,fup}^2)\frac{D_{t1,fup}}{2}n_{tp}} \tag{25}$$
$$\eta_{\mathrm{oxp}} = \mathrm{A_{e,oxp}}\varphi_{oxp}\left(\varphi_{oxp} + B_{e,oxp}\right) + \mathrm{C_{e,oxp}} \tag{26}$$
$$\eta_{\mathrm{fup}} = \mathrm{A_{e,fup}}\varphi_{fup}\left(\varphi_{fup} + B_{e,fup}\right) + \mathrm{C_{e,fup}} \tag{27}$$
$$\gamma_{\mathrm{cc}} = f_{\gamma}^{cc}(p_{cc}, \frac{\dot{m}_{co}}{\dot{m}_{cf}}) \tag{28}$$
$$\mathrm{R_{cc}} = f_{R}^{cc}(p_{cc}, \frac{\dot{m}_{co}}{\dot{m}_{cf}}) \tag{29}$$
$$\mathrm{T_{cc}} = f_{T}^{cc}(p_{cc}, \frac{\dot{m}_{co}}{\dot{m}_{cf}}) \tag{30}$$
$$\gamma_{\mathrm{gg}} = f_{\gamma}^{gg}(p_{gg}, \frac{\dot{m}_{go}}{\dot{m}_{gf}}) \tag{31}$$
$$\mathrm{R_{gg}} = f_{R}^{gg}(p_{gg}, \frac{\dot{m}_{go}}{\dot{m}_{gf}}) \tag{32}$$
$$\mathrm{T_{gg}} = f_{T}^{gg}(p_{gg}, \frac{\dot{m}_{go}}{\dot{m}_{gf}}) \tag{33}$$

The residual vector $\mathrm{R} = (\mathrm{R_1}, \mathrm{R_2}, \ldots \mathrm{R_{15}})^{\mathrm{T}}$ is a function of the performance variables defined as Eq.(34).

$$\mathrm{X} = (\mathrm{n_{tb}}, \dot{m}_{co}, \dot{m}_{cf}, p_{out,oxp}, p_{out,fup}, p_{cc}, \dot{m}_{go}, \dot{m}_{gf}, p_{gg}, C_{ad,tb}, \eta_{tb}, T_{0,tb}, p_{tbe}, \Delta\mathrm{T_{ox}}, \Delta\mathrm{T_{fx}})^T \tag{34}$$

The performance of liquid rocket engine is given as the following equation.

$$\mathrm{R(X)} = 0 \tag{35}$$

Equation (35) can be solved by Newton Raphson method. Fig. 2[3] presents the solution algorithm.

The material properties of combustion gas in combustion chamber is calculated by using CEA[4]. The material properties of gas generator combustion product is estimated as a function of mixture ratio[5]. As the mixture ratio increases the temperature, specific heat ratio, gas constant and characteristic velocity of combustion gas increases. This means that higher mixture ratio gives better working fluid for the turbine[6].

The present analysis program has been verified by comparing with the experiment results of powerpack for 7 ton class liquid rocket engine[7]. The powerpack is a test article of liquid rocket engine without combustion chamber. A safer experiment can be conducted with powerpack because the combustion chamber produces most of the engine thrust. The powerpack gives similar mass flow rate with the engine and enables versatile experiment with reduced handling risk. Table 1 summarizes relative errors. The relative error ranges from 1% to 4% for turbine pressure ratio, turbine exit temperature, turbine exit pressure. The reason of error is inaccurate model for the material properties of the gas generator. The relative errors for other parameters are less than 1%.

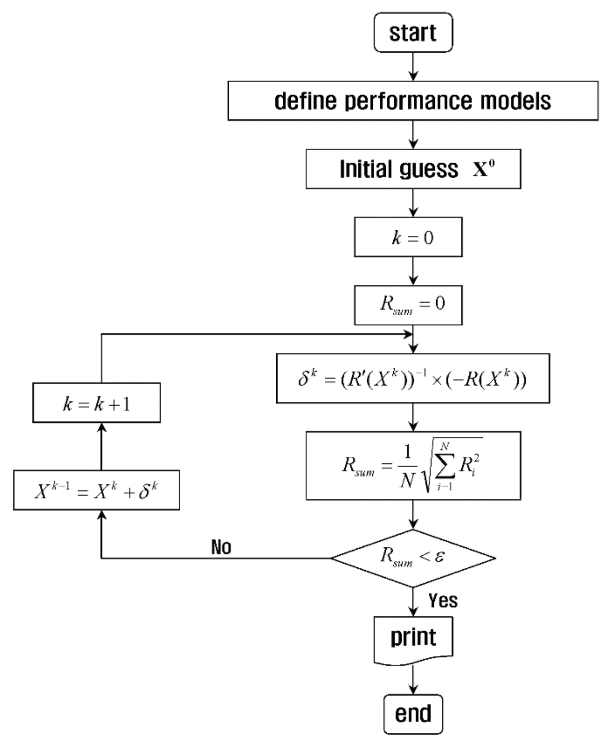

Fig. 2 Algorithm of mode analysis[3]

Table 1 Verification of simulation method against measured data of 7 ton class rocket engine powerpack[7].

parameter	analysis	relative difference
engine mixture ratio	2.2	0.03%
mass flow rate of oxidizer pump (kg/s)	14.8	-0.01%
mass flow rate of fuel pump (kg/s)	6.8	-0.04%
outlet pressure of oxidizer pump (MPa)	8.7	-0.08%

outlet pressure of fuel pump (MPa)	11.9	0.05%
oxidizer pump head (MPa)	8.4	-0.09%
fuel pump head (MPa)	11.8	0.04%
efficiency of oxidizer pump	0.66	-0.03%
efficiency of fuel pump	0.61	-0.03%
power of oxidizer pump (kW)	168	-0.44%
power of fuel pump (kW)	167	-0.60%
turbine mass flow rate (kg/s)	1.0	-0.35%
turbine efficiency	0.54	-0.42%
turbine pressure ratio	18.2	1.37%
turbine power (kW)	336	-0.52%
gas generator combustion pressure (MPa)	6.3	-0.72%
gas generator oxidizer mass flow rate (kg/s)	0.25	-0.34%
gas generator fuel mass flow rate (kg/s)	0.79	-0.34%
turbine inlet temperature (K)	906	0.44%
turbine exit temperature (K)	747	-4.15%
turbine exit pressure (MPa)	0.34	-2.06%

3. Results and Discussions

Figure 3 depicts the RPM change with respect to the pressure drop through COO, CFCV, GOCV and GFCV. In the x-axis "0" is not physically minus pressure drop but the reference pressure drop. So minus pressure drop in the figure means increasing the opening ratio of the valve from the predefined reference value. In the given rocket engine COO pressure drop is realized by replacing the orifice and the other pressure drops are realized by control valves. However all the above pressure drops have same meaning in analysis. The engine given in Fig. 1 is the 3rd stage engine for KSLV-II[8,9]. The nominal combustion pressure is 7.0 MPa and the mixture ratio is 2.2. The engine is turbopump-fed type so the performance is governed by turbopump RPM. CFCV pressure drop is not efficient to change the RPM because the slope is gentle. The pressure drop at COO, GFCV, GOCV can be effectively used to control RPM. The pressure drop at GOCV is the most efficient as the RPM slope is rapid. This means that only a little pressure change at GOCV can change RPM much. The control valve is operated as half open condition to make it possible to reduce pressure drop by open more and to increase pressure drop by closing valve. So it is impossible to reduce pressure drop after fully open the valve. In this connection the sensitivity of performance with respect to pressure drop has important meaning in liquid rocket engine. The ultimate reason to change RPM is to control combustion chamber pressure which finally changes engine thrust. The change of gas generator pressure has same tendency as RPM as shown in Fig.4. Quantitatively a little reduced influence is observed for the parameters COO and GFCV because turbopump RPM is influenced by both gas generator pressure (or mass flow rate) and gas generator temperature. The impact to gas generator temperature is described in Fig. 5. The gas generator temperature increases sharply as GFCV pressure drop increases. COO pressure drop mainly controls the combustion chamber fuel flow so it has only indirect effect to gas generator.

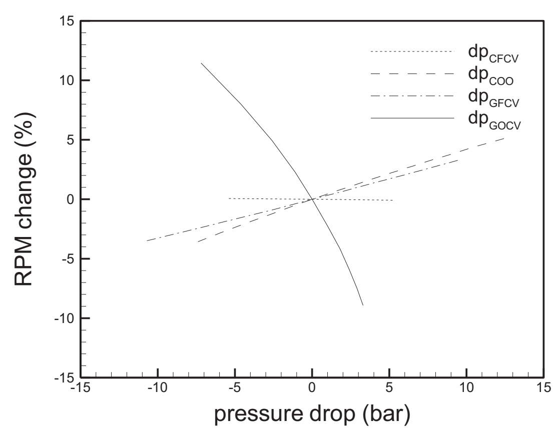

Fig. 3 RPM change vs pressure drop

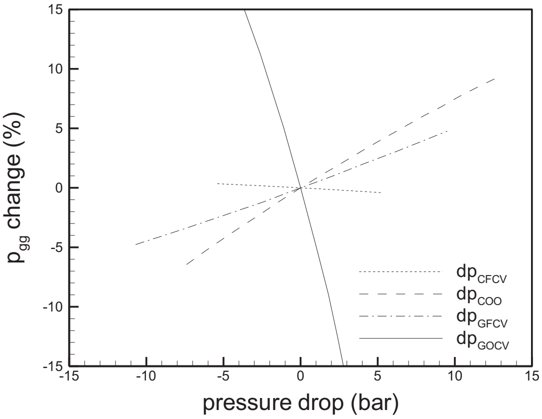

Fig. 4 p_{gg} change vs pressure drop

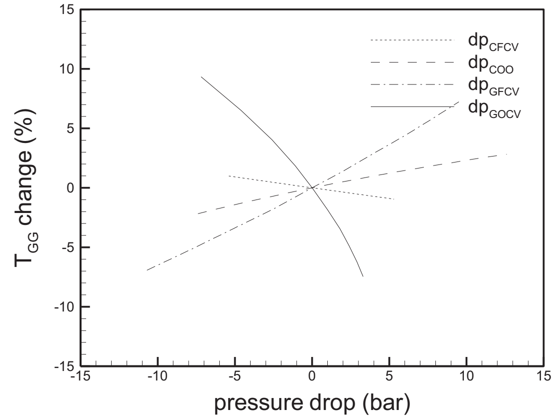

Fig. 5 T_{gg} change vs pressure drop

The main combustion chamber pressure which is directly proportional to engine thrust is depicted in Fig. 6. The combustion chamber pressure is proportional to both turbopump RPM and mixture ratio. Combustion chamber pressure has the same tendency as turbopump RPM given in Fig. 4. However the COO pressure drop has limited effect to combustion chamber pressure as COO pressure drop decreases mixture ratio which offsets the effect of increased turbopump RPM. So the COO pressure drop is not effective to control the engine thrust. Conclusively, pressure drop at GFCV or GOCV is efficient for the tuning of engine thrust. If thrust is tuned it also changes gas generator temperature which must be nominal to keep the turbine material under operational temperature range. The gas generator temperature is shown in Fig. 5. The pressure drop at CFCV is not adequate because the gas generator temperature slope is too slow which means that excessive pressure drop should be reserved for enough gas generator temperature change. COO pressure drop is also applicable however it is more efficient to use GFCV or GOCV. In conclusion once GOCV is used for thrust tuning, GFCV is the optimal for the tuning of gas generator temperature tuning.

Figure 7 describes the combustion chamber mixture ratio. The combustion chamber mixture ratio is effectively tuned by using COO or CFCV. The control part GFCV or GOCV has indirect effect to combustion chamber mixture ratio.

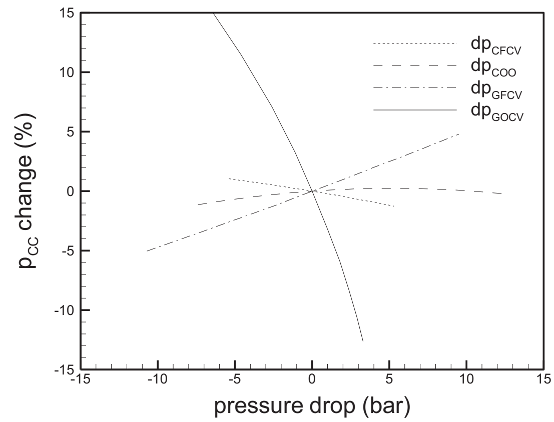

Fig. 6 p_{cc} change vs pressure drop

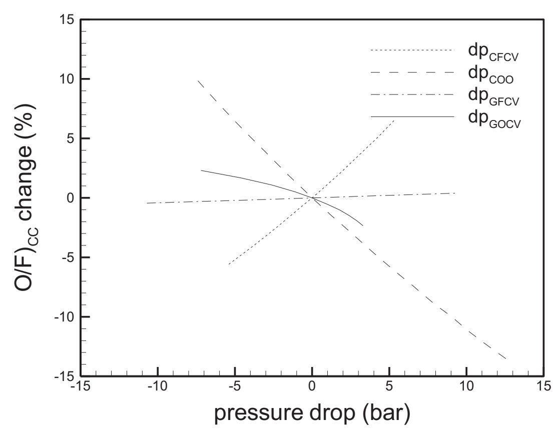

Fig. 7 $O/F)_{cc}$ change vs pressure drop

4. Conclusions

Performance tuning is the major technology to minimize the performance deviation from nominal value for a non-controlled liquid rocket engine. In the present study, the influence of pressure difference in COO, CFCV, GFCV and GOCV to thrust, combustion chamber mixture ratio and gas generator mixture ratio are estimated. Pressure difference in GOCV is more effective than twice as the other parts to change the thrust. Both COO and CFCV are effective by two fold as the other parts for combustion chamber mixture ratio tuning. The pressure differences at GOCV, GFCV and CFCV can be one of the combinations for tuning the thrust, gas generator mixture ratio and combustion chamber mixture ratio.

Nomenclature

A	empirical coefficient
B	empirical coefficient
C	empirical coefficient, combustion chamber
CC	combustion chamber
C_{ad}	spouting velocity
D	empirical coefficient
E	empirical coefficient
F	empirical coefficient
GG	gas generator
H	pump head
Q	volume flow rate
R	gas constant
RPM	turbine/pump revolutions per minute
T	temperature
TIT	turbine inlet temperature
TP	turbopump
X	design variable vector
c^*	combustion characteristic velocity
c_P	specific heat
g	gravitational acceleration
$\dot{m}$	mass flow rate
n	rotational velocity
p	pressure
u	velocity
x	variable
Δp	pressure difference
Ω	rotational velocity
α	in/out angle of turbine absolute velocity
β	in/out angle of turbine relative velocity
$\emptyset$	flow coefficient, velocity coefficient of turbine stator
γ	specific heat ratio
η	efficiency
ρ	density
φ	head coefficient, velocity coefficient of turbine rotor
ξ	effective loss coefficient

Superscript/subscript

0	stagnation
1	inlet
2	exit
H	head
TB	turbine
c^*	combustion characteristic velocity
cc	combustion chamber
cf	combustion chamber fuel
ccf	combustion chamber fuel
co	combustion chamber oxidizer
cco	combustion chamber oxidizer
f	fuel
fup	fuel pump
gf	gas generator fuel
gg	gas generator
go	gas generator oxidizer
in	inlet
out	outlet
o	oxidizer
oxp	oxidizer pump
rt	turbine rotor
st	turbine stator
t	tangential direction
tbe	turbine exhaust nozzle
tp	turbopump

Acronym

CC	Combustion Chamber
CCN	CC Nozzle
CFCV	CC Fuel Control Valve
CFI	CC Fuel Injector
CFO	CC Fuel Orifice
CFP	CC Fuel Pipeline
CFV	CC Fuel shutoff Valve
COI	CC Oxidizer Injector
COO	CC Oxidizer Orifice
COP	CC Oxidizer Pipeline
COV	CC Oxidizer shutoff Valve
CRCC	CC Regenerative Cooling Channel
FuP	Fuel Pump
GFI	GG Fuel Injector
GFCV	GG Fuel Control Valve
GFO	GG Fuel Orifice
GFP	GG Fuel Pipeline
GFV	GG Fuel shutoff Valve

GG	Gas Generator
GOCV	GG Oxidizer Control Valve
GOI	GG Oxidizer Injector
GOO	GG Fuel Orifice
GOP	GG Oxidizer Pipeline
GOV	GG Oxidizer shutoff Valve
GRCC	GG Regenerative Cooling Channel
OxP	Oxidizer Pump
TB	Turbine
TED	Turbine Exhaust gas Duct
TEN	Turbine Exhaust Nozzle

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