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A numerical prediction and flight test of the transient fuel temperatures in an aircraft

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Abstract

A numerical prediction of the transient fuel temperature in an aircraft was made and verified with a flight test. The analysis was studied with the finite difference method. Numerical calculation was performed by an explicit method of the modified Dufort-Frankel scheme. Convective heat transfer coefficients were used in calculating heat transfer between the aircraft surface and the ambient air. For an aircraft on the ground, an empirical equation represented as a function of free-stream air velocity was used. And, the heat transfer coefficient for flat plate turbulent flow suggested by Eckert was employed for in-flight phases. The governing equations used in this analysis are the mass and energy conservation equations on fuel and oils. The analysis was verified with the flight test data of a fuel system with additional fuel supplies and return concept. As a result of the verification, the difference of the fuel temperatures obtained by the analysis from those of the flight test data was relatively small with a tendency to increase in the later phases of the flight.

Keywords: Fuel temperature; Aircraft fuel system; Additional fuel supplies; Fuel Thermal Management System (FTMS); Flight test

1. Introduction

As aircraft performance has improved and the operational role of both commercial and military aircrafts has developed, requirements for the aircraft thermal management have arisen. With today's aircraft operating at high subsonic or supersonic speed, the emphasis has been moved toward the provision of cooling systems, although heating is still required [1].

The cooling problem brought about by the heat sources must be solved to successfully cool the aircraft systems in flight. Heat must be transferred from these sources to a heat sink and rejected from an aircraft. Around an aircraft, available heat sinks can include the ambient air and the internal fuel. The major systems and components generating heat in an aircraft are an aircraft hydraulic system, an electricity-generation system, an airframe mounted gearbox system, avionic equipment and instruments, etc.

Typically, for aircraft the thermal control air thermal management system (ATMS) and fuel thermal management system (FTMS) can be considered in association with the heat sink to be used. The air thermal management has a disadvantage in that it increases the aircraft drag because the resistance of the scoop, the pipework and the heat exchanger slows down the ram airflow. Also, the use of ram air as a cooling medium has its limitations since the ram air temperature increases with airspeed and soon exceeds the temperature required for cooling of aircraft systems or components [1].

However, the fuel is much better than the ambient air as a heat sink because it has a higher heat capacity and enables a higher heat transfer coefficient. Therefore, the heat generated in the heat sources of aircraft

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systems needs to be 'sunk' in the fuel. The heat dissipated from the heat sources is absorbed by hydraulic oil and lubricant oil circulating different heat sources. The fuel thermal management system (FTMS) uses the fuel to cool the oils absorbing heat generated in an aircraft during flight.

The primary purpose of an aircraft fuel system is to provide a reliable supply of fuel to the engine. In addition, the aircraft fuel system should be designed to provide thermal management for aircraft subsystems and engine which utilize fuel as a heat sink. Fuel temperature in an aircraft is changed with flight conditions and it affects directly the aircraft systems and components using fuel as a heat sink. Therefore, fuel thermal analysis should be considered with the conditions of the aircraft systems and components.

The fuel thermal management system has to be considered with the requirements of each aircraft system, component and engine that use fuel as a heat sink. A fuel system model with additional fuel supplies and return concept was employed in the present study, and a military aircraft was selected as the aircraft model. However, an analysis of the engine temperature was not performed in the present study because the engine is regarded as an independent unit from the aircraft cooling system.

In the present study the transient fuel temperatures are predicted and compared quantitatively with the measured temperatures in a flight test. In the analysis, fuel temperatures in different fuel tanks have been calculated in a fuel thermal management system where the fuel and the ambient air are used as the sink of heats generated in an aircraft hydraulic system, avionic equipment and instruments. For a fuel system model, including the forward and reward tank group, a numerical analysis on the fuel temperature has been performed for a given flight model.

2. Model

2.1 Fuel system model

The aircraft fuel system considered in the present study is shown schematically in Fig. 1. Fuel is stored internally in five fuselage tanks and two wing tanks. The five fuselage tanks are *I-1, I-2, I-3, I-4* and *I-5* tanks.

The forward (FWD) tank group is composed of the left wing tank (*W-1*), the forward tanks (*I-1, I-2*) and the forward reservoir tank (*I-3*). The rearward (RWD) tank group is composed of the right wing tank (*W-2*), the rearward tank (*I-5*) and the rearward reservoir tank (*I-4*). The order of the tanks listed above in each group means the sequence of fuel consumption in the tanks. All of the internal tanks in each group are interconnected by siphon tube and fuel channel through which the fuel is transferred by pumps.

Fuel in a tank is exhausted completely before the use of the fuel in the next tank. In actuality, after the wing tanks are fully emptied, the two forward tanks and the one rearward tank begin to empty, and when they are fully emptied, then the two reservoir tanks begin to empty finally.

The fuel is transferred by two independent methods to the reservoir tanks from which fuel is fed to the engine. One method, which is primary, is by siphoning action through tubes. The other method uses power to support fuel transfer and to scavenge the tanks. The powered fuel transfer in the internal tanks is provided by two electrical pumps: one for transferring fuel from the *I-1* tank to the *I-3* tank, and the other for transferring fuel from the *I-5* tank to the *I-4* tank. Average flow capacity of the transfer pump is 453.9 kg/h under all flight conditions. Each transfer pump dissipates 500 W of heat to the surrounding fuel in each tank.

The re-circulated hot fuel from the fuel/oil heat exchanger flows into a turbine pump located in each wing tank, which transfers the fuel at the rate of 1,452 kg/h to the reservoir tank. One turbine pump transfers the fuel from the *W-1* tank to the *I-3* tank, and the other from the *W-2* tank to the *I-4* tank. The turbine pump has not been considered as a heat source be

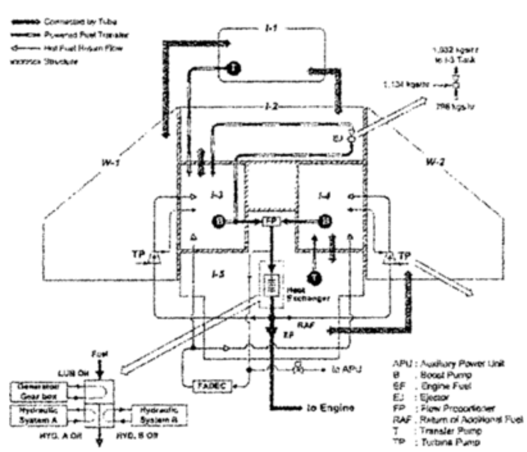


Fig. 1. Fuel system schematic.

cause heat dissipation of the turbine pump is negligiBle.

The fuel is fed from the reservoir tanks to the engine by an electrical boost pump located in each reservoir. Each boost pump has a maximum flow capacity of 14,515 kg/h and the fuel flow rate is changed depending on the flight condition. Heat dissipation rate of the boost pump in each reservoir is changed linearly with the required engine feed fuel flow ranging from the dry operation (1,114 W) to the maximum 14,515 kg/h (1,536 W). When the fuel is pumped from the reservoir, the fuel passes through the flow proportioner (which ensures the equal consumption of the fuel from the forward tank group and rearward tank group), and then passes through the fuel/oil heat exchanger. In the heat exchanger, heat transfer takes place between the fuel and the aircraft system oils, which have been heated up by the heat dissipated from the two hydraulic systems, the generator and the gear box as indicated in Fig. 1. Because of the heat exchange in the heat exchanger, the fuel is heated up and the aircraft system oils are cooled. After the heat exchange is finished, some part of the fuel is fed to the engine for engine combustion while the rest of the fuel (that is, the additional fuel for system cooling) is cooled and returned to the reservoir tanks.

Also, a Full Authority Digital Engine Control (FADEC) was considered as an additional heat source. Heat dissipation rate from this source has been considered as 230 W regardless of flight conditions. To cool the heat dissipated from the FADEC, the fuel flow is bifurcated at the heat exchanger inlet and then supplied to the FADEC. The flow of the fuel heated by the FADEC is divided and returned to each reservoir tank at equal flow rate. The rates of the additional fuel flow for the cooling of the systems (the hydraulic systems A and B, the generator and the gearbox) and the FADEC cooling fuel flow are

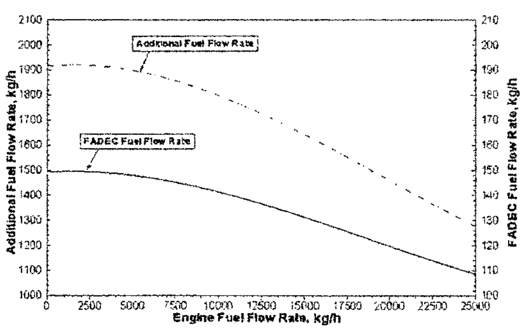


Fig. 2. Additional and FADEC fuel flow rates.

shown in Fig. 2 as a function of the engine fuel flow rate.

2.2 *Flight model*

An aircraft performs various flight missions in association with required purposes. A flight mission consists of a number of phases. Each flight phase is characterized by the flight time as well as by the typical flight conditions (altitude, Mach number) from the engine starting to the landing [2]. The flight model considered in the present study is shown in Table 1.

3. Analysis and flight tests

3.1 *Governing equations and numerical analysis*

The governing equations used in this analysis are the continuity and the energy conservation equations for fuel and oils.

Continuity:

$$\sum \frac{\partial m_i}{\partial t} = 0 \tag{1}$$

Table 1. Phases in the flight model used in the present study.

Phase	Flight time (min)	Altitude (m)	Mach number
Gnd. Oper.	0.0~14.55	0.0	0.00
Taxing	14.55~25.80	0.0	0.00
Last check	25.80~26.00	0.0	0.00
Takeoff/Accel.	26.00~26.91	951.7	0.47
Climb	26.91~29.25	1589.5	0.53
Aircraft check	29.25~29.73	1600.9	0.55
Climb	29.73~31.10	2987.8	0.55
Pitch/Roll maneuver	31.10~64.15	2938.3	0.46
Climb	64.15~67.87	6089.9	0.58
Pitch/Roll maneuver	67.87~77.33	5993.1	0.89
Loiter	77.33~78.58	5965.7	0.83
Descent	78.58~83.13	772.7	0.46
Approach	83.13~85.55	410.0	0.25
Touch & Go	85.55~86.60	0.0	0.23
Climb	86.60~87.00	609.6	0.38
Approach	87.00~88.20	432.8	0.25
Touch & Go	88.20~89.50	0.0	0.24
Climb	89.50~90.42	560.1	0.41
Approach	90.42~91.03	528.1	0.26
Landing	91.03~94.00	0.0	0.00
Gnd. Oper.	94.00~100.0	0.0	0.00

Energy conservation:

$$\sum \frac{\partial Q_i}{\partial t} = 0 \tag{2}$$

As shown in Fig. 3, major heat transfer factors related with the temperature variation in a fuel tank are as follows:

- External heat transfer from the atmosphere - $\dot{Q}_{external}$
- Internal heat transfer through the tank partition area commonly wetted by the fuel in adjacent fuel tanks - $\dot{Q}_{internal}$
- Thermal energy flow associated with the fuel flow through tanks - $\dot{Q}_{flow}$
- Heat generation caused by the heat sources in tank - $\dot{Q}_{source}$

With the heat transfer mechanism in Fig. 3, the basic energy conservation equation is represented as:

$$\begin{aligned} \left(mc_p \frac{\partial T}{\partial t}\right)_i &= \dot{Q}_{external} + \dot{Q}_{internal} + \dot{Q}_{flow} + \dot{Q}_{source} \\ &= U_{af} A_{af} (T_{air} - T_{\text{tank } i}) + h_f A_w (T_{tankj} - T_{tanki}) \\ &\quad + \sum (\dot{m} c_p T)_j - \sum (\dot{m} c_p T)_i + \dot{Q}_{source} \end{aligned} \tag{3}$$

where U_{af} is the overall heat transfer coefficient for heat transfer from the atmosphere to a tank. A_{af} is the heat transfer area of a tank exposed to the ambience. T_{air} is the ambient air temperature, and T_{tank} is the fuel temperature within a tank. h_f is the overall heat transfer coefficient between two tanks and A_w is the tank partition area commonly wetted by the fuel in adjacent fuel tanks. The subscript *tank i* represents the fuel tank whose temperature is to be solved, and *tank j* represents the fuel tanks adjacent to the *tank i*. $\dot{m}$ is the fuel flow rate and c_p is the specific heat of the fuel. The source term represents the heat rejected by the electrical fuel pumps located in the *I-1* and *I-5* fuel tanks and in the forward and rearward reservoirs as shown in Fig. 1.

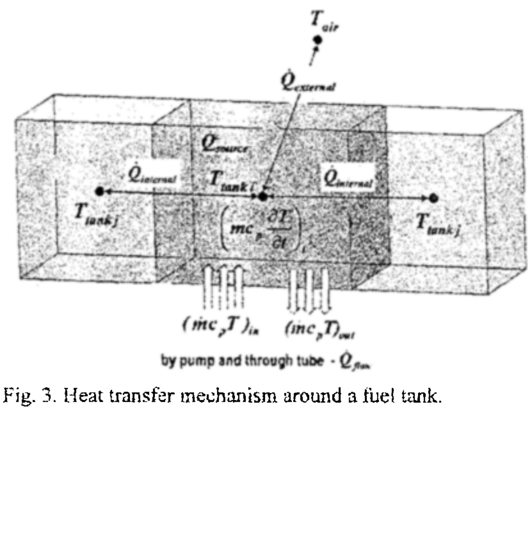


Fig. 3. Heat transfer mechanism around a fuel tank.

The aircraft fuel and oil systems in Fig. 1 are modeled to be a conductance-capacitance network for heat transfer with nodes as shown in Fig. 4, where a thermal network model including the fuel tanks and the fuel flow path with the heat exchanger is shown. Here, the diffusion nodes and arithmetic nodes are considered. The former means the node where heat capacitance is considered. A fuel tank is regarded as a diffusion node. The latter indicates the node where heat capacitance is not considered. The temperature nodes for the ambient air and for the junction of fuel pipe are the arithmetic nodes. Fuel and oil temperatures in an aircraft have been calculated to get transient solutions at each node. Finite difference numerical calculation has been performed by the SINDA/G code employing an explicit method called modified Dufort-Frankel scheme [3].

These nodes are thermally-interconnected with each other by conductor to reflect the heat and mass transfer caused by the fuel and oil flow in the fuel/oil system as well as environmentally induced heating and cooling effects. The conductor has conductance value, which is represented by the mass flowrate times the specific heat.

To perform numerical calculation for the diffusion nodes, the energy equation of Eq. (3) is presented as the following finite differential equation, Eq. (4), which involves three time levels and their corresponding temperatures:

$$\begin{aligned} &\left[1 + \frac{\Delta t}{2C_i} \sum_{j=1}^{l} G_{ij}\right] T_i^{n+1} \\ &= T_i^n + \frac{\Delta t}{C_i} \sum_{j=1}^{l} G_{ij} T_j^n \\ &\quad - \frac{\Delta t}{2C_i} \sum_{j=1}^{l} G_{ij} T_i^{n-1} + \frac{\Delta t}{C_i} \dot{Q}_i \end{aligned} \tag{4}$$

where T^n is the present temperature, T^{n-1} is the past temperature, and T^{n+1} is the unknown temperature. l is the total number of diffusion nodes and $\dot{Q}_i$ means

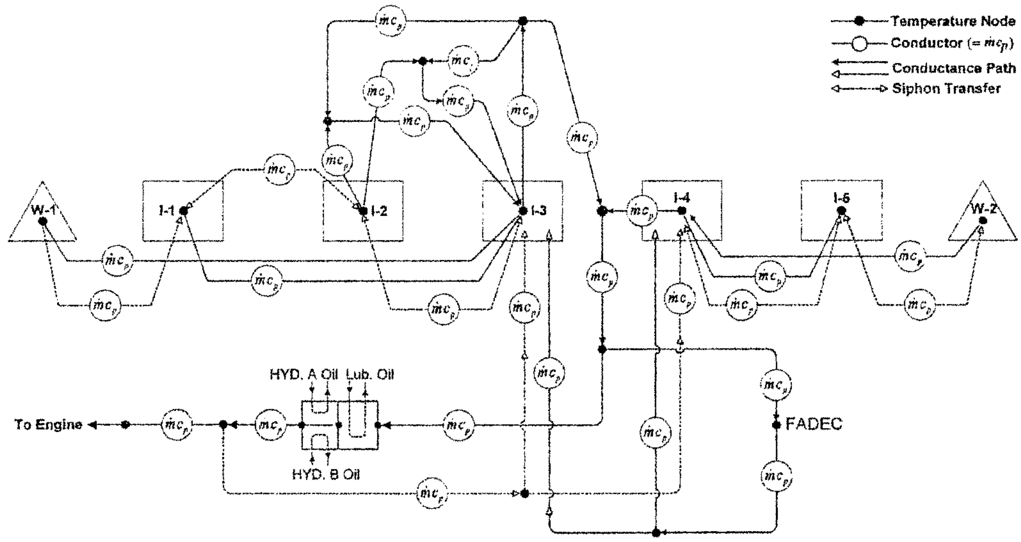


Fig. 4. Thermal network model including fuel tanks and flow path.

the source term. C is the thermal capacitance of a node, G_{ij} is the thermal conductance between nodes, Δt means the computational time step size. After the calculation of the diffusion nodes is completed, the temperatures of the arithmetic nodes are computed at the nodes where temperature calculation is needed. The temperature of arithmetic nodes is determined by Laplace or Poisson equation [4] and solved iteratively by the Gauss-Seidel method.

The arithmetic nodes are iterated until the convergence criterion, Eq. (5) is achieved.

$$|T_i^{old} - T_i^{new}| \leq 0.001 \tag{5}$$

The stability of each diffusion node is discriminated by Eq. (6). The computational time step size used in the present analysis starts with 0.05 minute. This time step size was used as the first time step size to avoid the startup error caused by the use of a large computational time step size. After the completion of computation with a chosen time step size, the current time step size of 0.05 min and a temporarily estimated time step size for the next computation obtained by Eq. (6) were compared and the smaller value out of the two time step sizes was chosen to be a time step size for the next calculation.

$$\Delta t \leq \left(\frac{C_i}{\Sigma G_{ij}} \right)_{\min} \tag{6}$$

3.2 Analysis conditions

3.2.1 Initial conditions and assumptions

The JP-8 aviation fuel, MIL-PRF-83282 fluid and MIL-PRF-23699 fluid have been considered as fuel, hydraulic oil and lubricant oil, respectively [5,6,7]. Initial fuel quantities are 322.9, 266.0, 236.7, 240.2, 531.6, 263.3, 261.1 kg for *I-1, I-2, I-3, I-4, I-5, W-1, W-2* tank, respectively Also, the following assumptions have been used on items which are too difficult to simulate as it is.

- As the tank fuel is decreasing, heat transfer area is changing. And the aircraft is always assumed to maintain horizontal flight, which allows the calculation of the heat transfer area wetted by the fuel in fuel tanks.
- Free-stream air flows from the forward to rearward and side-wind flow with respect to the aircraft is not considered.
- Thermal resistance of the conduction through the tank partition and tube walls is negligible compared with the thermal resistance of convective heat transfer.
- The air exists in the fuel tank with fuel decreasing and the heat transfer occurs between the ambient air and the tank fuel. But, the variation of the air temperature in a fuel tank has been ignored because the thermal capacity of the remaining fuel in a tank is much larger than that of the air.
- Heat losses from the surface of components such as the gearbox and generator to the ambience are not

considered. Heat transfer between the ambient air and the air in each component bay is negligible because it is closed by the aircraft surface.

3.2.2 Characteristic air temperatures

Adiabatic wall temperature (T_{aw}), total temperature (T_{tot}), ventilation air temperature (T_{vent}) and engine ventilation air temperature ($T_{eng.\ vent}$) were used as characteristic air temperatures to calculate the heat transfer rate between the ambience and fuel tanks. The adiabatic wall temperature and total temperature were used as the characteristic temperature of the aircraft surface exposed to external aerodynamic heating and exposed to engine inlet air, respectively. In addition, the total temperature was used for the ventilation air temperature between the *I-1* and *I-2* tanks as shown in Fig. 1. Also, the engine ventilation air temperature between the *I-5* tank and engine nacelle was assumed to be higher by 16.7 °C than the total temperature.

3.2.3 Heat transfer coefficients

Convective heat transfer coefficients were considered in order to calculate heat transfer between the above characteristic temperatures and the aircraft surface. Equation (7) is an empirical equation represented as a function of free-stream air velocity (V_o, m/sec) when an aircraft is located on the ground [8,9,10]. For in-flight phases Eq. (8) is employed for the heat transfer coefficient of a flat plate turbulent flow suggested by Eckert [11].

$$h_s = 5.677 \cdot (2.0 + 0.0957 \cdot V_o) \tag{7}$$

$$h_c = 0.037 \left(\frac{k}{x_c} \right) \mathrm{Pr}^{1/3} \left(\frac{\rho\, V\, x_c}{\mu} \right)^{4/5} \tag{8}$$

where k means the thermal conductivity, ρ the density, V the free stream velocity, μ the dynamic viscosity and x_c represents the characteristic length of a flat plate for average heat transfer coefficient, which will be discussed in the following section. An average fuel side heat transfer coefficient of 226.7 W/m^2-°C has been used for all fuel tanks, which includes the effect of tank surface geometry and orientation as well as internal fuel velocities.

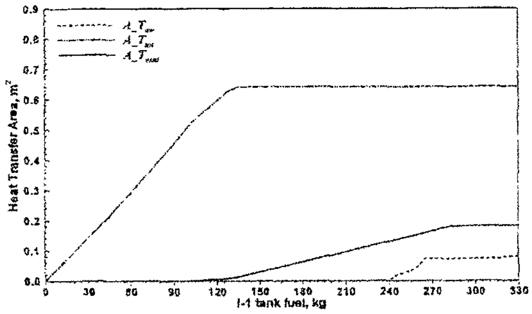
3.2.4 Characteristic lengths and heat transfer areas

The characteristic length used for the coefficient of heat transfer between the adiabatic wall temperature and tank surface is the distance from the aircraft nose to the mid-position of each tank. The characteristic length used for the coefficients of heat transfer between the engine inlet air and tank surface is the distance from the engine inlet duct to the mid-position of each tank. The characteristic length for the wing tank is the mid-length of the tank at the mid-span. The characteristic length for the *I-1* and *I-2* tanks affected by ventilation air is the half length of the tank exposed under ventilation air flow.

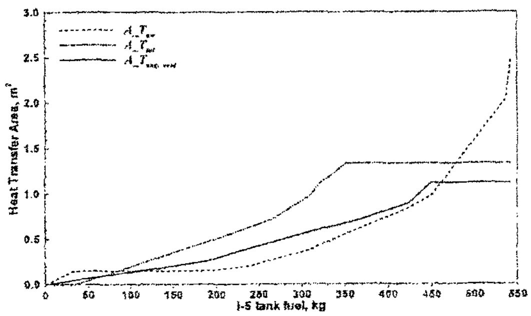
The characteristic length for the *I-5* tank affected by the engine ventilation air is the overlapped length of the *I-5* tank and engine. Shown in Table 2 are major characteristic lengths used to calculate the coefficient of heat transfer between the ambient air and the

Table 2. Characteristic lengths for tanks (m).

Tank	for h_{aw}	for h_{tot}	for h_{vent}	for $h_{eng.vent}$
I-1	5.96	0.98	0.54	
I-2	6.97	2.00	0.95	
I-3 & I-4	7.64	2.66		
I-5	8.57	3.60		0.63
W-1 & W-2	1.13			



(a) I-1 fuel tank



(b) I-5 fuel tank

Fig. 5. Variation of heat transfer areas.

rind of fuel tanks.

The amount of fuel in tanks is decreasing during flight. The heat transfer areas in fuel tanks exposed to the adiabatic wall temperature, the total temperature, the ventilation air temperature and the engine ventilation air temperature are also decreasing with fuel consumption. The heat transfer areas of the wing are only exposed to the adiabatic wall temperature. In addition, the variation of tank partition area commonly wetted by the fuel in adjacent fuel tanks has been considered. The heat transfer across the commonly wetted partition only occurs between (1) the *I-2* and reservoir fuel tanks, (2) the *I-5* and reservoir tanks, (3) the *W-1* and *I-2, I-3, I-5* fuel tanks, and (4) the *W-2* and *I-2, I-4, I-5* fuel tanks.

The wetted surface areas of the *I-1* and *I-5* fuel tanks are shown in Fig. 5 as a function of tank fuel quantity, respectively, and are reported in terms of the surface areas exposed to the adiabatic wall temperature (A_T_{aw}), to the total temperature (A_T_{tot}), to the ventilation air temperature (A_T_{vent}) and to the engine ventilation air temperature ($A\_T_{eng.vent}$). As shown in Fig. 5, the related heat transfer areas are decreased with fuel consumption proceeding from the right to the left of the abscissa.

3.2.5 Heat dissipation of aircraft systems

The heat dissipation of the aircraft hydraulic system comes from the hydraulic pump. The hydraulic oil is used as the coolant as well as the working fluid for transmitting hydraulic power from the pump to the actuator. The hydraulic oil is heated up when circulating in the hydraulic system and is flowing into the fuel/oil heat exchanger as shown in Fig. 1. The oil in the heat exchanger is cooled by the fuel and returned to each hydraulic system.

The heat dissipation of the main generator is related with the electrical load required by aircraft systems and equipments. The lubricant oil is also used as a coolant. The heat dissipated by the main generator is cooled in the fuel/oil heat exchanger. The heat load of the generator is increased with the increase in engine speed. In the domain of engine speed from 60% to 100% the heat dissipation ranges from 4.0 kW to 5.6 kW for 10 kVA load of the main generator, and from 5.3 kW to 7.1 kW for 30 kVA load, respectively.

The heat dissipation of the gearbox system is caused by the mechanical friction associated with gear rotation and power extraction. Here, the lubricant oil is also used as a coolant. The heat load of the gearbox system is related to the engine speed and to the horse power extracted by many different devices mounted on the gearbox [12]. The heat load of the gearbox system is increased with the increase in the horse power of the gearbox system. In the domain of total horse power (extracted by many different devices mounted on it) from 40 hp to 160 hp, the heat dissipation ranges from 2285 W to 2725 W for 70% of aircraft speed, and from 4310 W to 4485 W for 97% of aircraft speed, respectively. With the increase of engine speed from 60% to 100%, the lubricant oil flow rate in the gearbox system increases linearly from 31.6 l/min to 37.0 l/min.

3.3 Flight tests

The flight mission for flight tests was performed as given in Table 1 with high fidelity and the mass flow rate of the fuel, the temperatures and the amount of the fuel, the aircraft speed and altitude, and the ambient and total temperature were measured.

The mass flow rate of the fuel at the engine inlet was monitored and recorded in the FADEC. Fuel temperatures were measured in each tank and at the engine inlet. The temperature sensor is of P_t 100 ohm type(RTD). The uncertainty in the temperature measurement is +/- 1.0℃. As for the dimension of the temperature sensor, the lead length is 30.5 cm long with a diameter of 3.05 cm. The amount of fuel in each tank was measured by FQMS(Fuel Quantity measurement System).

The speed and altitude of an aircraft were measured with pitot tubes. The measured data can be used to check if the flight mission is carried out faithfully by the aircraft. The ambient temperature and the total temperature were gauged. Except for the data of the mass flow rate of the fuel at the engine inlet, the measured data were recorded in a data logger in the aircraft and transmitted to the ground station for data recording, simultaneously.

The time intervals for the measurement of different quantities are all different and are smaller than one second. However, the measured data are rearranged to express the variation of the quantities at every second.

4. Results and review

For a given flight condition shown in Table 1, transient fuel temperatures were predicted in the design stage of the fuel system, and also measured in the flight test. As indicated, the current flight condition includes many different phases with many different altitudes and flight speeds. After the initial ground operation, the taking-off follows starting at 26 minutes. The climbing to a higher altitude begins around at 30 minutes. Then the descending starts at around 79 minutes.

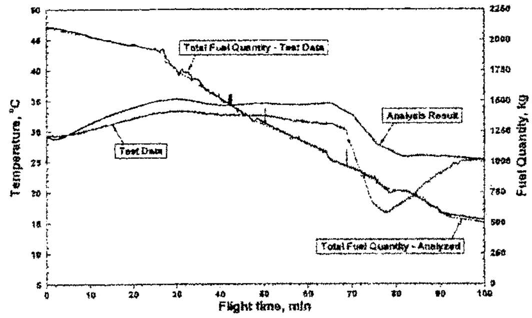
As mentioned earlier, the fuel in a tank is exhausted completely before the use of the fuel in the next tank. The current analysis and the measurement show a good agreement till the time when each tank is practically emptied. The *W-1* and *W-2* tank are emptied at 37 minutes, respectively, *I-1* tank at 70 minutes, *I-5* tank at 93 minutes. After the completion of the given flight, the amount of the remaining fuel in *I-2, I-3* and *I-4* tanks is 489.0kg, 228.7kg and 214.4kg, respectively.

The fuel temperature variation of the forward tank group is shown in Fig. 6, and that of the rearward tank group is shown in Fig. 7. As shown in these figures, the predicted fuel temperatures in the fuselage tanks are similar to those measured by the flight test, and the differences between the analysis results and flight test data are generally increased as the fuel consumption is increased, especially, in the later phases of the flight.

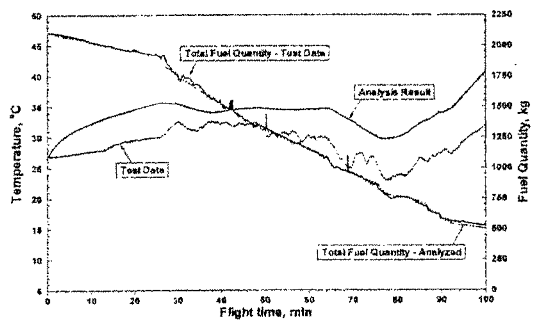
In the analysis, the fuel temperatures of all the tanks show a similar pattern with the exception of the fuel temperature in the *I-1* tank in the later phases. The fuel temperatures of the tanks increase during the period of the ground operation, taxing and last check. Then the fuel temperatures decrease during the period of 26 – 37 minutes as the altitude is increased with decreasing ambient air temperature. Now, the fuel temperatures rise again during the period of 37 - 64 minutes in the phase of pitch/roll maneuver at low speed. Again, the fuel temperatures decrease as the airplane undertakes the phases of climb and pitch/roll maneuver with high speed at a higher altitude, enabling high heat transfer rate from the fuel to the ambient air. Then the fuel temperatures increase again in the rest of phases (that is, during the phases from the loiter with light duty to the final ground operation).

The measured temperature in the *I-1* tank indicated in Fig.6 (a) begins to drop at around 66 minutes. Note that the fuel in the *I-1* tank is practically emptied at 70 minutes. In the phases of the climb, pitch/roll maneuver and loiter at a higher altitude with the ambient air temperature quite low, the fuel temperature in the *I-1* tank decreases rapidly with a negligible amount of fuel in the tank since the heat flow from the fuel to the ambient air through the fuel tank rind is considerable. However, when the altitude of the aircraft is much decreased, the above heat flow is not notable, which yields a temperature rise at around 78 minutes (in the measurement). Here, the *I-1* tank fuel temperature does not seem to affect much the temperatures of the other fuel tanks after 70 minutes because the negligible amount of the fuel in the *I-1* tank is not transferred to the adjacent tanks after 70 minutes.

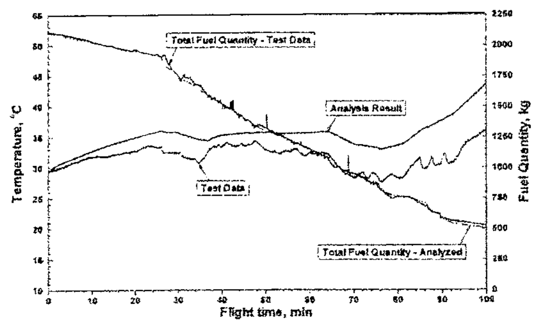
The difference in the temperature between the analysis and the test data in the *I-2* tank shown in Fig.



(a) I-1 tank fuel temperature



(b) I-2 tank fuel temperature

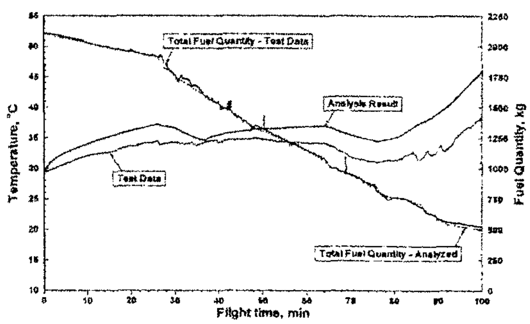


(c) I-3 tank fuel temperature

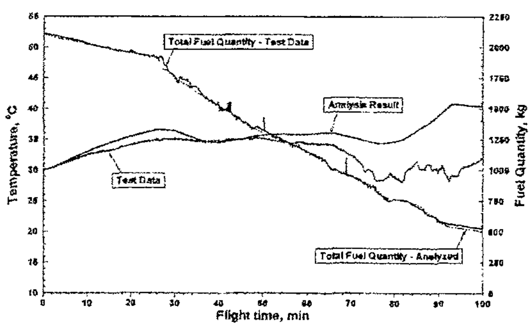
Fig. 6. Temperatures in forward tank group.

6 (b) is somewhat more informative during the initial ground operation phase (see Table 1). The averaged rate of the temperature increase in the analysis is somewhat greater than that obtained from measurement. There are several reasons for this discrepancy.

Firstly, the temperature sensor is located on the tank bottom in the opposite side of the siphon tube interconnected to the *I-3* tank. The hot fuel is returned from the heat exchanger to the *I-3* tank and then the *I-3* tank fuel overflows to the *I-2* tank. Therefore, the temperature sensor located far from the inlet of the hot fuel in the *I-2* tank could underestimate the average temperature of the *I-2* tank. Secondly, the physical tank configuration of the *I-2* tank has a horse saddle shape. Therefore, the overflow fuel from the *I-3* tank could not be mixed well with the rest of the fuel in the *I-2* tank, which may cause the temperature sensor to underestimate the representative temperature of the *I-2* tank. Thirdly, the hydraulic heat load may be overestimated in the analysis of the *I-2* tank. The hydraulic heat load directly affects the prediction of the fuel temperature of the *I-2* tank because the heat dissipation from the hydraulic lines passing through the *I-2* tank to the fuel in the tank has been considered



(a) I-4 tank fuel temperature



(b) I-5 tank fuel temperature

Fig. 7. Temperatures in rearward tank group

in the present study. It is thought that these factors have yielded the difference between the predicted temperature and the measured temperature in the *I-2* fuel tank during the phase of the initial Ground operation. Also, the comparison of the predicted quantity of the total fuel and the measured total fuel amount is shown with a reasonable degree of agreement, where the measured total fuel amount shows fluctuation indicating some errors in the measurement.

As shown in Fig. 6 (c), Fig. 7 (a) and (b), the fuel temperature of the *I-3, I-4* and *I-5* tanks is generally similar to that of the *I-2* tank. As for the *I-5* tank, except for later phases of flight, the predicted fuel temperature agrees well with the measured data. Among the fuel temperatures of all the fuselage tanks the fuel temperature in the *I-5* tank is considered to be least affected by the heat load of the hydraulic system because the hydraulic oils from the hydraulic pump are cooled first when passing through the *I-3* and *I-4* tanks and then cooled again when passing through the *I-5* tank. And, the fuel quantity of the *I-5* tank is the greatest among all fuel tanks.

The fluctuating fuel temperatures in the fuselage tanks except the *I-1* fuel tank during the period of 84-92 minutes of the flight are closely related with the flight pattern in the corresponding phases including the two times of the Touch & Go indicated in the later part of the flight model(Table 1). During the above period the characteristics of the heat transfer between the tank bottom and the ambient air are rapidly changed in association with the opening and shutting of the main landing gear door. Especially, this temperature fluctuation is shown to be remarkable in the *I-3* and *I-4* tanks because the heat transfer area of the *I-3* and *I-4* tanks exposed to the main landing gear bay is larger than that of the *I-2* tank. The maximum difference between the analyzed prediction and the

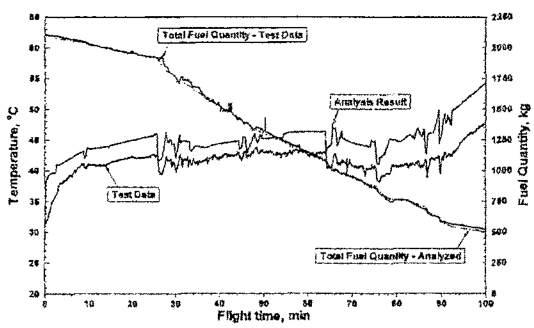


Fig. 8. Fuel temperatures at the engine inlet.

Table 3. Comparison of the average temperatures in various positions.

Fuel Temperature	Test Data (°C)	Analysis (°C)	Difference (%)
I-1 tank	28.71	31.56	9.93
I-2 tank	28.84	33.61	16.54
I-3 tank	31.70	35.16	10.92
I-4 tank	33.32	36.28	8.88
I-5 tank	32.74	35.54	8.55
Engine inlet	41.56	45.07	8.45

test data in the *I-5* tank is shown to occur at 93 minutes.

The fuel temperatures at the engine inlet are shown in Fig. 8, where the general behavior of the predicted temperatures and the test data are similar to those ofthe most fuel tanks, respectively, and here also the difference of the two temperatures is increased as the fuel consumption is increased, especially, in the later phases of the flight.

At t = 50 minutes, the fuel temperature at the engine inlet is higher than the fuel temperature of *I-3, I-4* tank around by 9°C both in the analysis and measurement. In actuality, this much of a temperature difference is observed between the fuel temperature of *I-3, I-4* tanks and the fuel temperature at the engine inlet during most of the flight.

In Fig. 8, several sudden temperature drops of the fuel temperature are seen, which can be explained by temporal high convective heat transfer on the surface of the airplane enabling higher heat dissipation from many different heat sources to the environment, yielding less heat load for the lubrication oil passing through the generator, gearbox and hydraulic systems, for example, around at t = 26 minutes when taking-off/acceleration, and around at t = 64 minutes when climbing to a high altitude. Also in the figure, observed is the temperature fluctuation in association with the opening and shutting of the main landing gear door again during 84-92 minutes.

The time-averaged predicted temperature and measured temperature in various positions for the whole duration of the flight model are listed in Table 3, where the relative difference in percentages is also given with the predicted temperature used as a reference temperature. Here, for comparison, the degree of the warmth of fuel above 0°C is considered for simplicity. With the current criterion, the relative difference of the two temperatures falls within 10% in most of the positions.

5. Conclusions

Fuel is used as a coolant in an aircraft and the fuel temperature in a military aircraft is influenced by many different heat sources such as the aircraft hydraulic system, electricity generation system, airframe mounted gearbox system, and FADEC. Therefore, the thermal management of the aircraft fuel system should be designed to satisfy the temperature requirements to ensure the normal operation of the heat generating components and engine during a flight.

In this study, the model of an aircraft fuel system with additional fuel supplies and return concept_was considered for the prediction and the measurement of fuel temperature in flight. A transient analysis on the fuel temperatures was studied by using finite difference method. Numerical calculation was performed by an explicit method and the measurement of the fuel temperatures during the flight was carried out. The results showed that the difference between the predicted fuel temperatures and those measured in the flight test is relatively small to be within 10% in most of the fuel tanks based on a simplified comparison criterion with a tendency to increase in the later phases of the flight after more than the two-thirds of the total fuel has been consumed. The current analysis has been shown to be an effective method to predict the fuel temperatures in an aircraft. Based on the result of the present study, the analysis method introduced in the present study can be applied to the design of the optimal thermal management system for commercial and military aircraft.

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