Practical Flight Test Method for Determining Reciprocating Engine Cooling Requirements

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A practical flight test procedure has been developed and verified for determining the cooling requirements for an installed air cooled reciprocating engine of any type The technique is based on a simple modification of the NACA cooling correlation method developed some 40 years ago for radial engines The modification involves replacing an empirically determined combustion gas temperature T_g with a measured exhaust gas temperature T_{egt} This latter parameter can be easily obtained utilizing cockpit instruments only The modification, therefore, reduces the need for extensive and expensive ground tests Data were collected in two separate flight test programs that verify the technique first with a rather completely instrumented airplane and later with minimal instrumentation and a minimal number of data points In both cases the data fits that comprise the correlation procedure were quite good Even the second tests with the sparse test matrix produced root mean square fit errors of less than 4% suggesting that the procedure provides consistent answers for practical flight test situations

Nomenclature

a a_1 a_4	= constants used in power law relationships, Eqs (1 5)
b b_1 b_4	= constants used in power law relationships, Eqs (1 5)
c	= cooling correlation constants used in Eqs (5 7)
H	= rate of heat transfer to and from the engine
k_1 k_3	= general constants for a linear equation
P_I	= indicated power developed by the engine
m n	= cooling correlation constants used in Eqs (6 7)
T_a	= ambient cooling air temperature
T_{egt}	= exhaust gas temperature
T_g	= effective combustion gas temperature
T_h	= engine cylinder head temperature
w	= mass flow rate of cooling air through the engine
W_c	= mass flow rate of fuel air mixture to cylinders
Δp	= cooling air baffle pressure drop across the engine
σ	= ratio of ambient density to standard sea level density
σ_{ex}	= ratio of heated cooling air density to standard sea level density

Introduction

EFFICIENT and effective cooling of air cooled reciprocating aircraft engines is a continuing problem for the general aviation industry A number of different models in production will overheat during some operating conditions The operators of these airplanes pay for cooling deficiencies with increased fuel consumption and reduced performance When overheating occurs, the pilot has three options: 1) decrease the combustion temperature by increasing the fuel flow to the engine through the mixture control, 2) increase the cooling airflow over the engine by opening the cowl flaps, or 3) alter the flight condition to one which results in acceptable engine operating temperatures. The first of these options obviously increases fuel consumption; the second increases drag, thus reducing performance while increasing fuel consumption; and the third is often unacceptable from a purely operational point of view and may even be dangerous, as in an emergency climb situation.

There are a number of factors that may contribute to overheating The use of supercharging to increase the obtainable power from an engine of a given size also increases the heat which must be removed through the engine's external surface area The increased heat rejection requires a corresponding increase in cooling air mass flow, which can be realized with either an increase in airspeed or a redesign of the cooling installation The problem is compounded by increased operating altitude associated with supercharged engines While the heat rejection for a given power setting remains constant up to the critical altitude, the ability to generate the required cooling air mass flow decreases with altitude The decrease in air temperature with altitude is not sufficient to compensate for the reduced mass flow rate The geometry of the horizontally opposed aircraft engine, though it helps reduce the frontal area of the airplane, hinders the cooling installation design The efficient passage of cooling air through the engine compartment requires large openings and large plenums, while low drag and styling dictate reduced frontal area, a tightly cowled engine and small plenums Finally, the engine cooling data supplied by the engine manufacturer are in a form useful for cooling installation design but the data are not compatible with subsequent ground and flight test procedures to resolve the cooling questions

The controlling variables for cooling and installation aerodynamics have been investigated previously and the results reported by Miley et al [1] As part of this original investigation, experimental methods were developed to determine cooling requirements of an instrumented prototype or production airplane One of these methods—a flight test procedure—has been refined and further verified with additional testing and is shown herein to be a straightforward means of determining cooling requirements with minimal instrumentation

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Background

The problem to be addressed is that there are no applicable cooling data for a reciprocating engine once the engine has been installed in a particular aircraft If a cooling deficiency surfaces after the engine is installed, there is no baseline from which to attack the problem Figure 1 illustrates a typical ground test setup used by the engine manufacturer to determine engine cooling requirements Representative cooling data are given in Fig 2 The left side of the graph gives the relationship between the cooling air mass flow through the engine's cooling fin passages and the corresponding pressure difference across the engine Due to the use of baffle plates on past and present engines to control the cooling air flow this pressure difference is commonly referred to as the baffle pressure drop. The relationship between the air mass flow rate and the pressure drop is that of an orifice

$$w=a(\sigma_{ex}\Delta P)^{b} \quad (1)$$

The parameters a and b are constant for a specific engine model and depend upon cooling fin spacing, passage length and total passage cross sectional area The exit density ratio is a measure of the density of the heated cooling air after it passes through the engine compartment Previous research on radial air cooled engines shows that this parameter accurately accounts for altitude and heating effects on basic orifice behavior The altitude curves in Fig 2 are obtained by extrapolating the ground based test cell data using Eq (1) The test arrangement in Fig 1 is ideal in terms of cooling effectiveness and the measurement of the important parameters The cooling airflow enters the fin passages uniformly The baffle pressure drop is easily measurable with a static ring in the sides of the duct above the engine

The right side of Fig 2 gives the cooling air mass flow necessary to maintain the indicated cylinder head temperatures as a function of engine power and ambient air temperature Using both sides of Fig 2, the functional relationship between the required cooling air mass flow and the baffle pressure drop at a given altitude can be obtained In theory the cooling requirements determined from Fig 2 can also be used after the engine has been installed in the aircraft After the engine is installed, the baffle pressure drop can still be measured and related to the cooling air mass flow rate This mass flow rate itself, however, cannot be measured in flight by any practical means Accordingly the data collected from ground tests and summarized in Fig 2 should serve as the basis for solving cooling drag problems during flight test development However the ground tests are not always practical and they do not provide all the answers needed

A typical engine installation for general aviation airplanes is sketched in Fig 3 The cooling air passes through the propeller and enters the upper plenum chamber through inlets on either side of the propeller shaft The flow must then make a right angle turn to pass through the engine and exit at the lower aft part of the cowl The flow path is decidedly different from the flow path through the ground test cell installation in Fig. 1 One cannot expect to measure the same orifice behavior as is found in the test cell installation The central question is: What pressure measurement should be used in the upper plenum of the aircraft installation to give equivalent values to the test cell measurements? The industry practice in measuring this upper plenum pressure is far from uniform An investigation of a variety of measurement techniques[1] showed that correlation was possible, but only after an impractical amount of work. For the flight test and ground test data to correlate, ground test cell configuration data had to be taken after the specific engine was installed on the airframe. Then after extensive instrumentation of the flight test article, compatible airborne measurements could be made An alternative approach, and one that is more practical, is offered in this paper It eliminates the need to correlate pressure measurements between the ground test cell and the airplane by generating the cooling data directly from flight tests It will be shown that the NACA cooling correlation method, suitably modified, serves as the mathematical model for the process Furthermore, experimental evidence will be presented, showing that the model is a practical tool that can be used on engines installed in airplanes with little instrumentation other than the cockpit instruments

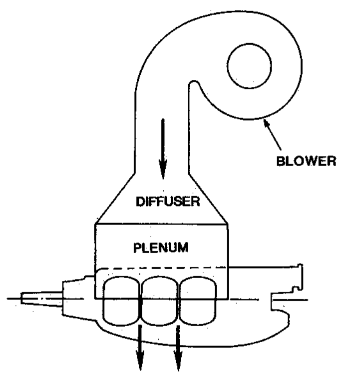

Fig 1 Ground test cell determination of cooling requirements.

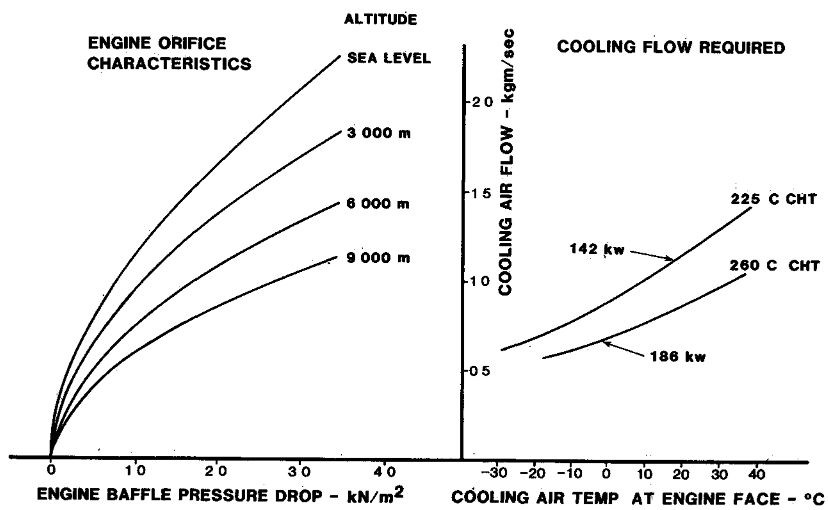

Fig 2 Representative engine cooling requirements data

Theory

The development history of the NACA cooling correlation method is well documented[2-9] Similar cooling correlation schemes are given in Refs 10 and 11. The objective of the NACA method is to provide a reliable means for extrapolating engine cooling behavior from ground measurements to operational conditions The underlying physical principle is that the heat generated by combustion and the heat given up to the cooling air must balance The rate of heat transferred to the cylinder from the combustion gases is

$$H = a_1 \dot{W}_c^{b_1} (T_g - T_h) \tag{2}$$

where $\dot{W}_c$ is the charge flow rate (air plus fuel) to the cylinder, T_g the effective combustion gas temperature, T_h the cylinder head temperature, and a_1 and b_1 are constants

The charge flow rate is directly related to the power developed by the cylinder and Eq (2) can be rewritten as

$$H = a_2 P_I^{b_2} (T_g - T_h) \tag{3}$$

where P_I is the indicated engine power The heat given up by the cyinder is

$$H = a_3 w^{b_3} (T_h - T_a) \tag{4}$$

where w is the cooling airflow rate, and T_a the temperature of the cooling air.

Since it is impractical to measure the cooling airflow in flight, the orifice relationship given by Eq. (1) can be used, giving

$$H = a_4 (\sigma_{ex} \Delta p)^{b_4} (T_h - T_a) \tag{5}$$

For equilibrium conditions (constant cylinder head temperature), the heat transferred to the cylinder by the combustion gases is equal to the heat transferred from the cylinder head to the cooling air Equating Eqs (3) and (5) and rearranging,

$$(T_h - T_a)/(T_g - T_h) = cP_I^m/(\sigma_{ex} \Delta p)^n \tag{6}$$

Eq (6) is the relationship referred to earlier as the NACA cooling correlation

This correlation algorithm was used extensively for air cooled engine ground and flight test research and development from the late 1930's through the end of World War II References 12 through 25 describe the scope and diversity of this usage. When the emphasis changed to jet propulsion after World War II, references to the application of the NACA cooling correlation method, along with other radial engine technologies, disappeared from published literature It has apparently been used little by the general aviation aircraft industry, and this fact may be one of the reasons for the number of cooling problems reported by users of general aviation airplanes today

The theoretical discussion to this point has considered only one engine cylinder but Eq (6) is also valid for the complete engine The correlation constants c m and n are determined for a particular engine model through the ground tests alluded to earlier. The parameters T_h T_a P_I, σ_{ex}, and Δp are all measurable in flight The effective combustion temperature T_g is a correlation variable that typically requires extensive ground testing It is a measure of the combustion heat transferred to the cylinder walls T_g is a function of a number of different engine variables including power, fuel/air ratio, induction air temperature, ignition timing, and exhaust back pressure

Application of the NACA cooling correlation method in its present form to general aviation cooling problems is not practical The correlation constants c and n depend upon the orifice characteristics of the particular engine, which in turn depend upon the cooling air flow pattern about the engine and the method of baffle pressure drop measurement Because of radial engine geometry, the flow pattern and pressure measurement method are the same for such an engine whether it is installed in a test cell on the ground or in an aircraft nacelle in flight Consequently, the ground test values of c and n apply to flight test also However, the geometry of the horizontally opposed engine gives a different flow pattern and complicates measurement of pressures between ground tests and flight tests Accordingly, the ground test values of c and n cannot be applied to flight test data A second restriction on the NACA cooling correlation is that determination of T_g requires an extensive ground test program in a facility where the cooling air temperature can be adjusted over a wide range This task was a relatively minor effort for the major radial engine companies during World War II, but is beyond the resources of current general aviation engine companies

Proposed Modification of the NACA Cooling Correlation

Since T_g, the effective combuston gas temperature, is merely a measure of the heat transferred to the cylinders by the combustion process, it was postulated that T_{egt}, the temperature of the exhaust gases, might serve equally well as a measure of the heat transferred Replacing T_g with T_{egt},

$$(T_h - T_a)/(T_{egt} - T_h) = cP_I^m/(\sigma_{ex} \Delta p)^n \tag{7}$$

The effective combustion gas temperature is an empirically determined parameter related to the heat transferred to the cylinder head. Since the concern is strickly with a functional relationship, the exhaust gas temperature should also functionally correspond to the heat transferred It is not necessary—indeed, it is highly unlikely—that T_{egt} will be equal to the combustion temperature in the cylinder; it is only required that T_{egt} vary with T_g monotonically If the constants in Eq (7) can be determined readily for any flight condition, the equation will also provide an accurate and useful model of the cooling airflow requirements at any other flight condition specified by P_I T_a, T_h and T_{egt}

For the sake of practicality, another objective was to use cockpit instrumentation for the measurements if possible Most of the functional variables in Eq (7) can be measured with sensors already in the typical general aviation instrument panel Experience suggests that a gage resolution of ± 2 deg is needed for the cylinder head temperature T_h The exhaust gas temperature should be measured at a point in the exhaust manifold where an average of the individual cylinder gas temperature exists For turbocharged engines having a cockpit gage for the turbine inlet temperature, that measurement would likely suffice The engine operating condition can be adequately modeled with manifold pressure and tachometer readings, the common instruments used by the pilot to set power Cooling air exit density ratio σ_{ex} requires an additional temperature measurement at the cowl exit However, again only because functional relationships are sought, it may be possible to substitute the ambient density ratio σ and eliminate the need for this temperature sensor An early version of the NACA cooling correlation utilized σ However compressbility effects on the flow through the cylinder cooling fin passages correlated better with σ_{ex} Experimental data indicate that σ can be used below 6000 m through σ_{ex} should be

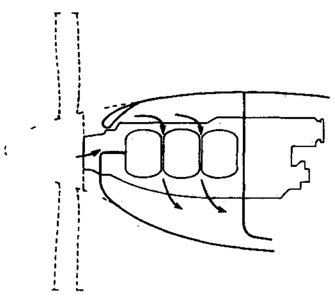

Fig 3 Typical aircraft installation for cooling

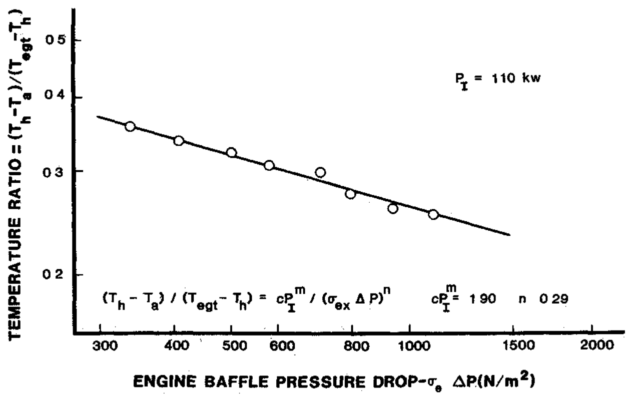

Fig. 4 Solution for n with $P_I = 110$ kW

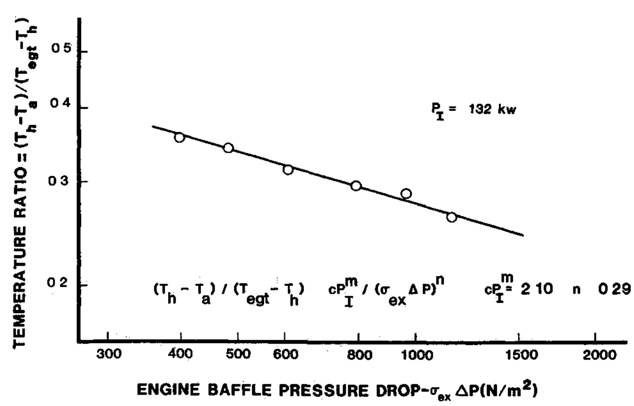

Fig 5 Solution for n with $P_I = 132$ kW

used above this altitude The baffle pressure drop can be measured in one of several straightforward ways [1]

This power law reduces to a linear relationship when it is put in logarithmic form When the logarithm of $(T_h - T_a)/(T_{egt} - T_h)$ is plotted vs the logarithm of $\sigma_{ex}\Delta p$ or $\sigma\Delta p$ (either of which represents the baffle pressure drop), a family of parallel straight lines is produced The slope of these lines is the correlation constant n The individual intercepts for each curve in the family are functions of the indicated engine power Since these intercepts take on specific values of $cP_I{}^m$ for each value of P_I, the constants c and m can be determined rather simply In fact, a graphical technique is possible if the logarithm of the product $cP_I{}^m$ is plotted vs the logarithm of the indicated engine power

The test matrix consists of selecting four or more indicated power settings over the range of interest For each power setting, the baffle pressure drop is varied by controlling airspeed and cowl flap setting. For multiengine airplanes, asymmetric power operation can be used For single engine aircraft, landing gear, flaps, and/or turns can be used to provide the necessary variation of airspeed at constant power settings

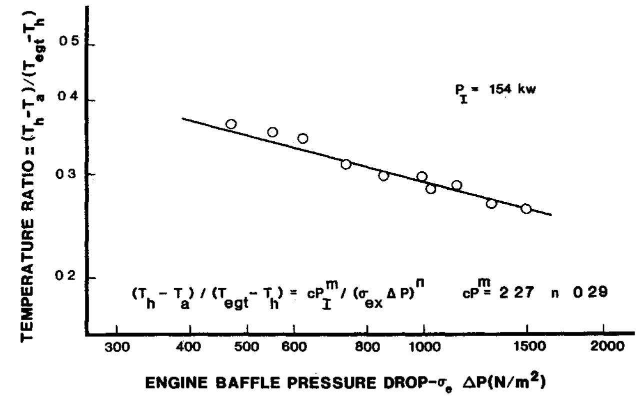

Fig 6 Solution for n with $P_I = 154$ kW

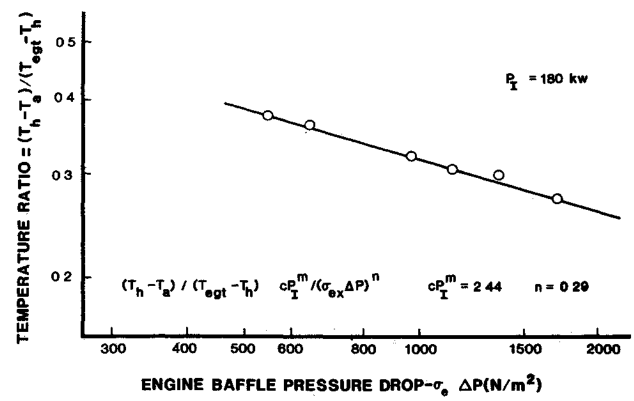

Fig 7 Solution for n with $P_I = 180$ kW

Results for a Highly Instrumented Light Twin

The first series of tests were flown in a Piper PA 41P Aztec at Mississippi State University. The powerplants were Lycoming TIO 540 L1AD turbocharged engines rated at 201 kW (275 hp) The instrumentation for this series of tests was extensive The cylinder head temperatures were measured by bayonet-type thermocouples connected to a digital ther mometer indicator The exhaust gas temperature was measured at the supercharger turbine inlet by a conventional EGT probe and a dial indicator Ambient air temperature was measured by a shielded probe thermistor sensor The cooling air exit temperature was measured by a thermocouple probe shielded from exhaust pipe and engine radiation The engine baffle pressure drop was measured by Kiel probes located at the rear engine baffle in the upper (high pressure) plenum, and baffle shield pressure probes in the lower (low pressure) plenum This system[1] gave the same cooling air mass flow vs baffle pressure drop as the ground test cell data This in strumentation was more complex than is desirable for practical flight tests, but in the initial stages of the project, it provided a means of determining reasons for possible failure of the model

Four engine power settings were used, ranging from roughly 55% power to 90% power on the test engine in stallation For each of these power settings, altitudes from 1000 up to 7300 m were flown The mixture setting and cowl flap position were also varied at each power level Two forms of cylinder head temperature data were used in the analysis First, the average of all six cylinders was used Also, the temperature of the hottest cylinder, which was consistently cylinder No 6 for this installation, was used Both measurements gave similar results The data presented in Figs 4 7 were those obtained by averaging the cylinder head temperatures of all six cylinders.

It must be emphasized that Figs 4 7 each include the full range of altitudes, mixture settings, and cowl flap positions

The correlation parameters obtained from each curve are also listed at the bottom of the figure The slopes of the curves on this log log grid determine the value of n Figure 8 illustrates how the data can be cross plotted to give the correlation constants c and m The complete correlation is summarized in Fig 9

Results for a Light Twin with Minimal Instrumentation

The second series of tests was performed at Texas A&M University approximately two years after the first series The subject airplane for these tests was a Gulfstream Aerospace Corporation Commander 700 powered by two Lycoming TIO 540 R2AD turbocharged engines rated at 254 kW (340 hp) The objective of this second set of tests was to demonstrate that the modified NACA cooling correlation would produce credible results with little more than cockpit instrumentation and just enough data points to establish the governing curves

This minimal instrumentation package though it was primarily cockpit instruments, did include sensors not on a production airplane The existing cylinder head temperature gages, for example, lacked the sensitivity needed for these tests A conventional bayonet resistance probe was installed in cylinder No 6 and a portable digital voltmeter was used as the manual readout for the sensor The circuit was excited by a 6 V lantern battery and a 1000 Ω resistor was the only "signal conditioning" electronics Baffle pressure drop was measured from piccolo probes[1] mounted above and below the engine in the nacelle An altimeter was used as a differential pressure gage with a valve to switch between the upper and lower piccolo probes An airspeed indicator was also connected across the two pressures to provide a backup pressure measurement and to give a real time measure of the pressure difference

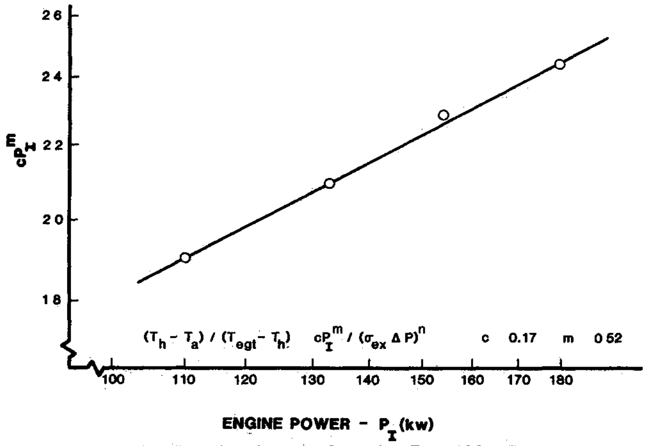

Fig 8 Solution for c and m with $P_I = 180$ kW

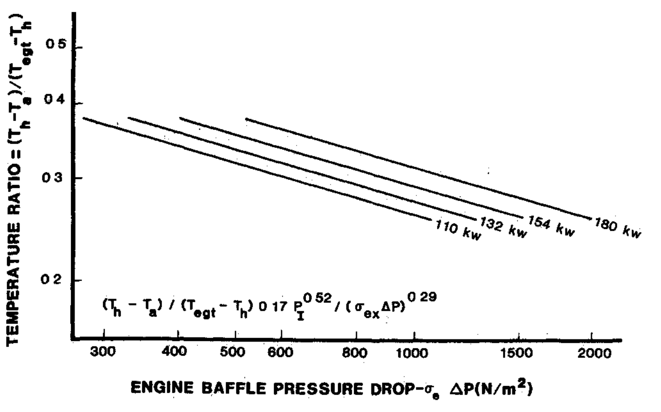

Fig 9 Summary of cooling correlation test results

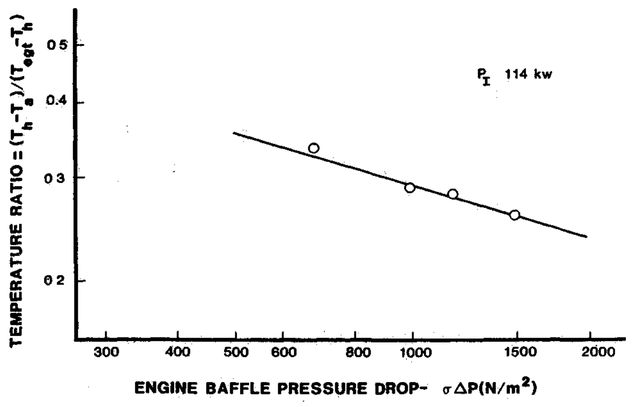

Fig 10 Cooling correlation results for $P_I = 114$ kW

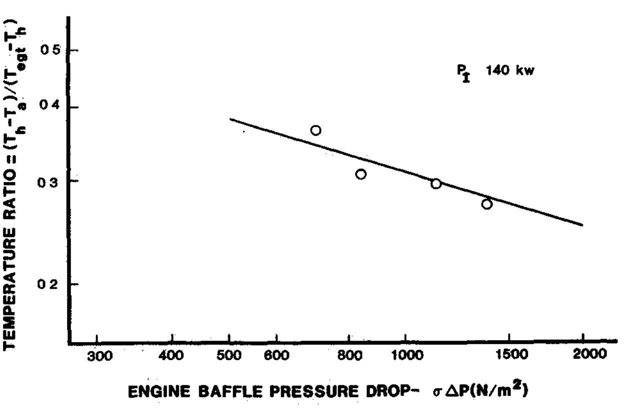

Fig. 11 Cooling correlation results for $P_I = 140$ kW

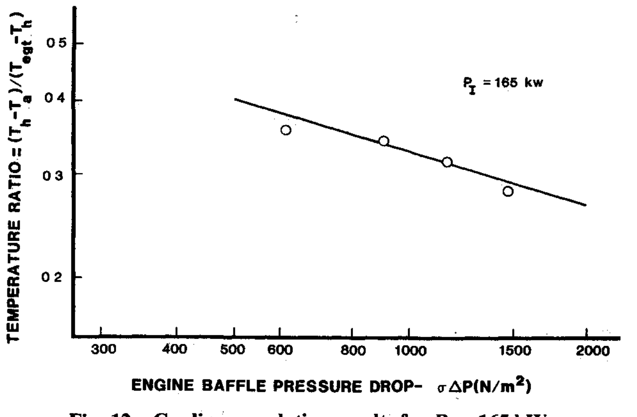

Fig 12 Cooling correlation results for $P_I = 165$ kW

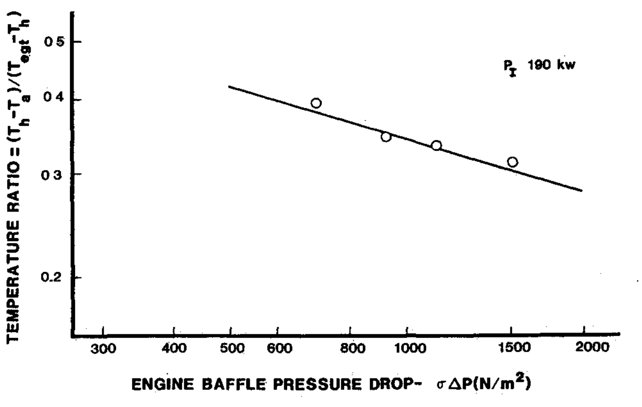

Fig 13 Cooling correlation results for $P_I = 190$ kW

The remainder of the sensors were the same gages used by the pilot to fly the airplane The cooling air and exhaust gas temperatures T_a and T_{egt}, respectively, were read from a standard outside air temperature gage and the gages used to set the fuel/air mixture for best power and/or maximum economy for cruise The usual manifold pressure and tachometer gages were used to set engine power

Four indicated engine power settings were used: $P_I = 114$ kW (45%), $P_I = 140$ kW (55%), $P_I = 165$ kW (65%), and $P_I = 190$ kW (75%) Data were collected at four different airspeeds for each power setting, giving a test matrix of only 16 total data points Airspeed was varied at each power setting by using asymmetric power; that is, the test engine was maintained at the selected power setting while the power on the opposite engine was controlled as necessary to obtain the four airspeeds The nominal test altitude for the entire series was 2400 m

Reduced data from these tests are plotted in Figs 10 13 The ambient density ratio was used to reduce the data since the test altitude was considerably below 6000 ms As has been earlier stated, using σ instead of σ_{ex} eliminates the need for a temperature measurement at the cooling flow exit and has little effect on the validity of the correlation

Initially, the data were analyzed as described in the preceding paragraph for the heavily instrumented airplane A least squares fit to the four data points taken at a constant power setting was used to determine the values of n and the respective cP_I^m values The random variations in n and cP_I^m were large enough with this very limited data set that it was impossible to complete the determination of c and m Following the suggestion of Corson,[8] the analysis was altered to use all 16 of the data points in the least squares algorithm instead of using just 4 at a time Taking the logarithm of Eq (7),

$$\ln[(T_h - T_a)/(T_{egt} - T_h)] = m\ln P_I - n\ln(\sigma\Delta p) + \ln c \quad (8)$$

This expression is of the general form

$$y = k_1 x + k_2 z + k_3$$

which is amenable to the least squares procedure but utilizes all the data points, not just a sparse subset For the 16 data points collected, the resulting correlation constants are: $c = 0\ 473$ $m = 0\ 33$ and $n = 0\ 30$ which gives a cooling correlation power law of

$$(T_h - T_a)/(T_{egt} - T_h) = (0\ 473 P_I)^{0\ 33}/(\sigma\Delta p)^{0\ 30} \quad (9)$$

The fit errors of even this extremely sparse data matrix were relatively low, with the root-mean-square (rms) error only 3 4% and the maximum error 7 3% The lines in Figs 10 and 13 show graphically how the fitted curves calculated from Eq (9) match the data

Equation (9) can be easily solved for the required baffle pressure drop to maintain a value of cylinder head tem perature T_h for any altitude specified by σ indicated engine power setting P_I, ambient air temperature T_a, and exhaust gas temperature T_{egt} Naturally, the baffle pressure drop is for the specific pressure probes used in the measurement installation As fixes for cooling deficiencies are designed, the changes can be evaluated using the same instrumentation scheme Use of the modified NACA cooling correlation, however, fixes the desired Δp. Either ground tests or additional flight tests can be used to develop and verify the fixes quickly and efficiently after the final configuration is chosen

Conclusion

A flight test procedure based on a modified NACA cooling correlation method, has been developed and verified. It gives cooling requirements in terms of the required pressure drop quickly and efficiently by relating parameters that are easy to measure with simple instrumentation Use of this approach in developmental flight tests allows the designer to evaluate proposed solutions for cooling tightly cowled reciprocating engines

Acknowledgments

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