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Effect of Turbofan Cycle Variables on Aircraft Cruise Performance

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A graphic analysis method has been developed to predict the effect of engine cycle variables on aircraft cruise performance. The results of the analysis are presented for a subsonic turbofan-powered aircraft; however, the method is applicable to any vehicle whose performance may be approximated by the Breguet range equation. The trends shown by the analysis are explained in terms of the distribution of energy to thrust, nacelle drag, and various losses in the turbofan cycle. Parametric turbofan cycle data are presented for a range of cycle pressure ratios, turbine inlet temperatures, and bypass ratios representing current and near-future technology. The cycle data are combined with installed weight and drag data for pylon-mounted engine installations to establish a relationship between engine cycle variables, installed fuel consumption, and installed engine thrust-to-weight ratio. The aircraft cruise range is expressed also as a function of the installed specific fuel consumption and installed engine thrust-to-weight ratio by a modified version of the Breguet Range Equation. This permits a graphic solution of the effects of engine cycle variables on aircraft range by superimposing engine and aircraft performance data on a common plot. This plot permits immediate identification of optimum combinations of the cycle variables and the sensitivity of aircraft range to these variables.

Nomenclature

CPR = cycle pressure ratio = P_{t3}/P_a
F_n = net thrust, uninstalled engine
F_{nc} = net thrust, installed engine = thrust minus drag
g = gravity constant
J = mechanical equivalent of heat
L/D = aircraft lift-to-drag ratio excluding nacelle, pylon, and interference drag
LHV = fuel heating value (BTU/lb)
Ln = natural logarithm
Mn = Mach number
P_a = ambient pressure
P_{t3} = compressor discharge, total pressure
R = range (computed by Breguet equation)
R' = reference range = a constant
SFC = specific fuel consumption, uninstalled engines (lb/hr/lb)
SFC_c = specific fuel consumption, installed engine
T_{t4} = turbine inlet total temperature
V_0 = flight velocity
V_j = jet velocity of fan and gas-generator streams
W_0 = aircraft gross weight, start of cruise
W_1 = aircraft gross weight, end of cruise
W_a = total engine airflow rate
W_e = weight of installed engine
W_{ef} = weight of installed engine plus fuel
W_f = fuel flow rate
γ = ratio of specific heats
η_g = gas-generator efficiency
η_p = propulsive efficiency
η_t = Brayton cycle ideal efficiency

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Introduction

EFFICIENT engine cruise performance is of primary importance to long-range transport aircraft. As much as 90% of the fuel load is burned during the cruise segment of a typical turbofan transport mission, and this can represent as much as 40% of the aircraft takeoff gross weight. Therefore, the engine cruise efficiency can influence significantly aircraft size, payload/range relationship, and operating costs.

Significant technological advancements have been made in the propulsion industry since the design of the currently operational turbofan engines. As a result of this improved technology, higher compression ratios, turbine temperatures, and bypass ratios are feasible, and engine weight and drag reductions are possible. All of these advancements contribute to potential improvements in engine design. To obtain the maximum benefit from this progress in the design of a new transport turbofan engine, the proper combination of cycle variables must be selected for a given mission requirement.

This paper investigates the problem of turbofan cycle selection from the viewpoint of the aircraft designer. The final selection of an engine cycle is dependent obviously upon the relationship between engine and aircraft performance at all mission flight conditions. However, because of the impor-

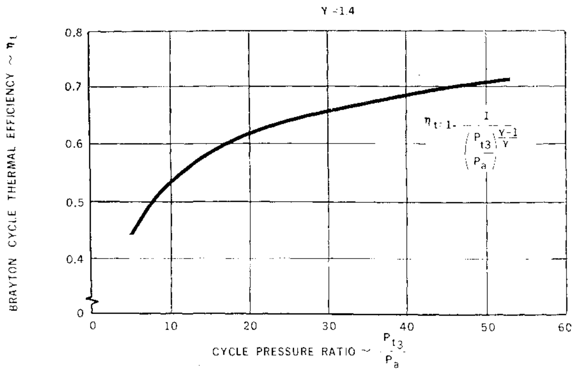


Fig. 1 Brayton cycle efficiency.

tance of cruise performance to the transport mission, evaluation of engine cycles on the basis of cruise efficiency provides significant insight into the appropriate cycle to be considered for a given transport mission problem.

Cycle Energy Distribution

To analyze properly the effects of the cycle variables on the over-all efficiency of turbofan performance, it is important to consider the distribution of energy within the process. The energy released by combustion of the fuel is accounted for in two primary categories: 1) residual heat, and 2) acceleration of the mass flow through the engine. The residual heat category is divided into losses attributable to the gas-generator inefficiency, to ideal cycle heat rejection, and to heat losses in the fan drive turbine, the fan, and the nozzles (both fan and gas-generator). A part of the energy apportioned to acceleration of the engine mass flow appears as a residual velocity loss; the remainder is the thrust energy. Each of these losses is analyzed below.

Gas-Generator Efficiency

For the Brayton cycle on which the gas-generator is based, the efficiency is defined

$$\eta_t = 1 - \frac{1}{(P_{t3}/P_a)^{[(\gamma-1)/\gamma]}} \tag{1}$$

The relation is shown in Fig. 1. This efficiency is independent of temperature but represents an ideal engine having 100% component efficiencies and zero pressure loss. In practice, where losses do exist, the gas-generator efficiency (η_g) is affected by turbine inlet temperature. The net effect, as illustrated in Fig. 2, is to cause the η_g to vary from zero, at the value of T_{t4} where the heat addition is just sufficient to overcome the losses, to η_t at an infinite value of heat addition. It can be shown that this trend is similar for all values of compressor pressure ratio and that, at practical values of turbine inlet temperature, η_g increases with increasing cycle pressure ratio.

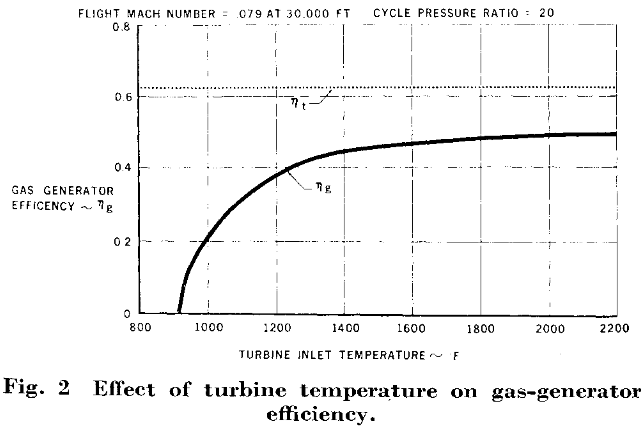


Fig. 2 Effect of turbine temperature on gas-generator efficiency.

Fan, Fan Turbine, and Nozzle Losses

The losses in the fan, turbine, and the fan and gas-generator nozzles are aerodynamic losses associated with the compression and expansion processes. For fixed component efficiencies, these losses are a function of the amount of energy associated with each process. Therefore, this loss fraction is constant for a given bypass ratio, and is independent of turbine inlet temperature.

Residual Velocity Loss

The losses discussed to this point show up as residual heat in the fan and gas-generator exhaust streams. The remaining fraction of the combustion heat release acts to accelerate the engine airflow; however, only a portion of this energy is available for the production of thrust. The unavailable remainder is dissipated as a loss that is the unavoidable consequence of producing thrust by an airbreathing engine. This is called residual velocity loss (RVL) and is associated with the relative motion of the jet and the undisturbed ambient air. This loss is purely mechanical as opposed to the thermodynamic origin of the gas-generator losses.

Consider a turbofan engine flying at a velocity of V_0 with a jet velocity of V_j for both the fan and gas-generator exhaust streams. V_j must be greater than V_0 for air to be accelerated and thrust produced. This relative velocity, opposite to the direction of flight, results in kinetic energy in the jet wake given by

$$KE_{\text{jet}} = \text{residual velocity loss} = [(V_j - V_0)^2/2g] \tag{2}$$

Since this energy is supplied by the engine but does not contribute to the production of thrust, it shows up as a loss in the cycle.

Thrust Energy

The remaining fraction of combustion heat release available for the production of thrust can be expressed in terms of jet velocities as follows:

$$\text{thrust energy fraction} = \frac{\text{thrust energy}}{\text{combustion heat release}} = \frac{(V_0/g)(V_j - V_0)}{(W_f/W_a)(LHV)} \tag{3}$$

The relation between the residual velocity loss and the thrust energy (TE) fraction is known commonly as the propulsive efficiency η_p, which is defined

$$\eta_p = \frac{TE}{TE + RVL} = \frac{(V_0/g)(V_j - V_0)}{(V_0/g)(V_j - V_0) + [(V_j - V_0)^2/2g]} = \frac{2}{1 + (V_j/V_0)} \tag{4}$$

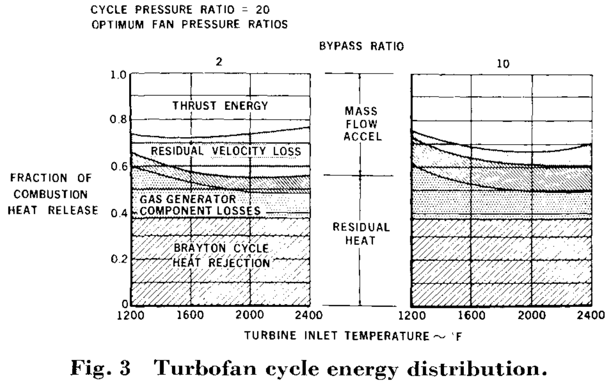


Fig. 3 Turbofan cycle energy distribution.

Effect of Cycle Losses

The distribution of energy described in the previous paragraphs is presented in Fig. 3. The relative magnitude and character of each increment of the combustion energy as a function of turbine inlet temperature and for various turbofan bypass ratios is shown. These data were obtained from computation of the cruise performance of a family of turbofan engines designed for optimum performance at a fixed cruise flight condition. For purposes of this paper, the flight Mach number was held constant at $Mn = 0.79$, and the cruise altitude was fixed at 30,000 ft. A complete analysis of airplane/engine matching could require comparison at several flight conditions. The computed performance covers a range of cycle parameters of expected interest for the next generation of subsonic turbofan engines, and assumes component efficiencies that are estimated to be consistent with the technological advances. Examination of the trends (Fig. 3) indicates that gas-generator efficiency improves with turbine temperature as discussed previously, and that remaining residual heat losses are essentially constant for a given bypass ratio. Similarly, it can be seen that as turbine temperature increases, a larger fraction of the heat release is available for acceleration of the engine airflow. This trend suggests a potential improvement in specific fuel consumption (SFC) with turbine inlet temperature; however, the residual velocity loss also increases with turbine inlet temperature and partially counteracts the improvement. Examination of the propulsive efficiency equation shows that this occurs because the thrust energy varies with $(V_j - V_0)$ whereas the residual velocity loss varies as $(V_j - V_0)^2$.

The incompatibility between high temperatures and low exhaust velocities is resolved by the selection of the proper bypass ratio. The effect of bypass ratio on energy distribution is given in Fig. 4, which is a crossplot of Fig. 3 at a fixed-turbine inlet temperature. As bypass ratio increases, the gas-generator energy output is used to accelerate a larger mass flow through a lower $(V_j - V_0)$. Thus, the residual velocity losses for any value of turbine temperature are seen to decrease with increasing bypass ratio. As a result, the gas-generator losses may be reduced at higher bypass ratios by increasing the turbine temperature.

Optimum Fan Pressure Ratio

All performance data in this paper are given at the optimum fan pressure ratio. The reason for the occurrence of an optimum fan pressure ratio can be explained in terms of the residual velocity losses or propulsive efficiency. For a given engine having fixed values of turbine temperature, bypass ratio, and cycle pressure ratio, the gas-generator useful energy output is fixed (Fig. 4). Thus, the jet velocities of the gas-generator and fan are controlled by the fan pressure ratio. At very low fan pressure ratios, the fan exit velocity is low, causing the thrust contribution of the fan to be negligible. The energy extracted from the primary stream to drive the fan at this condition is low, causing a high gas-generator jet velocity. Thus, the propulsive efficiency of the stream producing the major portion of the thrust is relatively low. As the fan pressure ratio increases, the gas-generator jet velocity approaches V_0. At this condition, there is no thrust from the gas-generator and the fan jet velocity is high, causing a low relative propulsive efficiency of the fan stream. Thus, a point must lie between these fan pressure ratio extremes where the combined thrust of both streams reaches a maximum. For ideal conditions, it can be shown that this point occurs when the fan and gas-generator jet velocities are equal. Figure 5 shows the effects on SFC and F_n/W_a caused by variations in fan pressure ratio and turbine inlet temperature for a fixed value of bypass ratio and compressor pressure ratio. The optimum fan pressure ratio is indicated clearly.

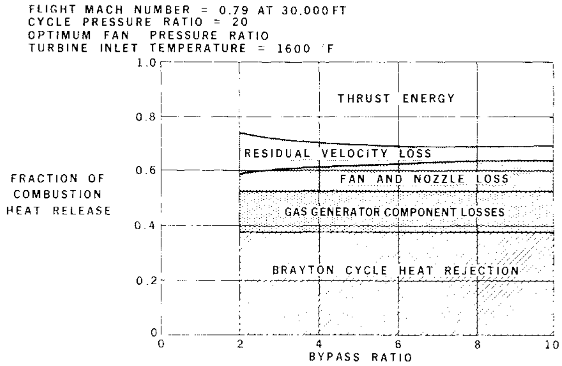


Fig. 4 Effect of bypass ratio on energy distribution.

Engine Performance Analysis

The preceding discussion covered the characteristics of the turbofan engine in terms of efficiencies obtainable by variations of the cycle parameters which affect the distribution of energy within the cycle. To relate the engine efficiency to aircraft performance, it is convenient to express the engine efficiency in terms of specific fuel consumption. As shown previously, the over-all efficiency is proportional to the fraction of combustion energy available for thrust, and Eq. (3) can be reduced to the form:

$$\text{thrust energy fraction} = [V_0/(SFC)(LHV)(J)] \qquad (5)$$

Therefore, it is apparent that maximization of the V_0/SFC term corresponds to maximization of the thrust energy fraction (or over-all engine efficiency). It will be shown later how variation of this term is related to maximum aircraft range. As noted previously, an increase in V_0 beyond approximately 0.8 Mn is not beneficial because of drag rise onset. Thus, this study considers only variations of SFC.

The performance of the family of potential turbofan engines representing the range of parameters investigated is illustrated in Fig. 6. In these curves, engine SFC is shown as a function of F_n/W_a for various bypass ratios. Each bypass ratio line is the locus of a series of design-point engines having varying turbine inlet temperatures, and having the optimum fan pressure ratio. The effect of compressor pressure ratio is obtained by comparison of the three curves, which indicates decreasing fuel consumption with increasing cycle pressure ratio. Several important trends are apparent from each of these curves: 1) for any given bypass ratio, there is a turbine temperature that produces the minimum SFC; 2) as bypass ratio is increased, this optimum turbine temperature increases; and 3) an envelope exists that represents the minimum SFC obtainable at any value of specific thrust.

Analysis of Aircraft Range

On the basis of the previous curves, the apparent trend to follow for maximum cruise performance is toward higher

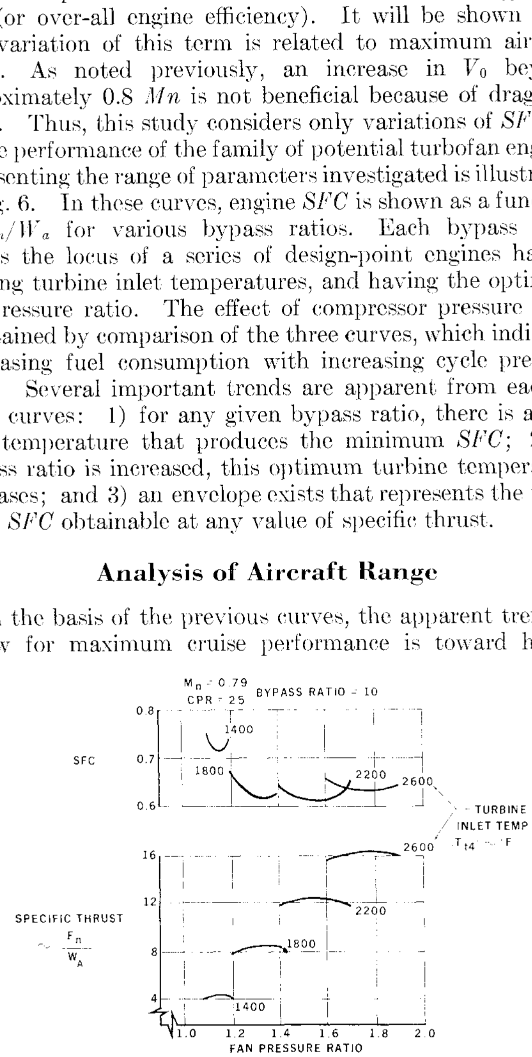
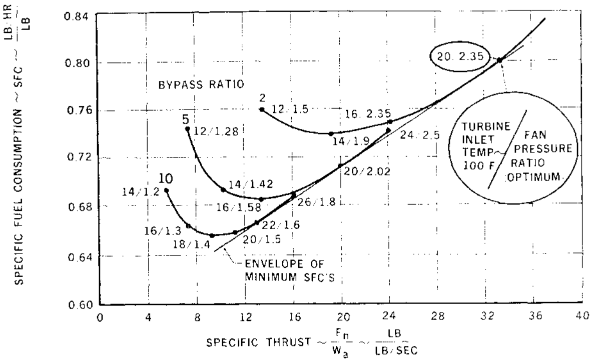
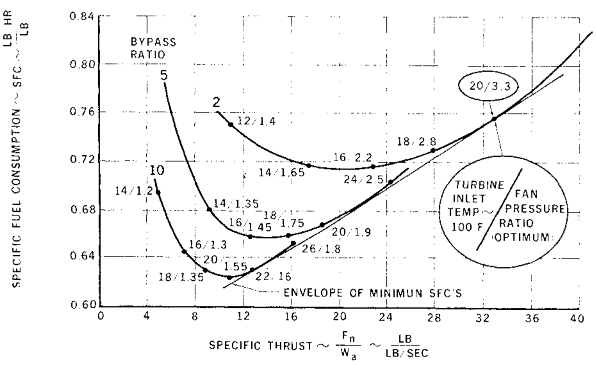


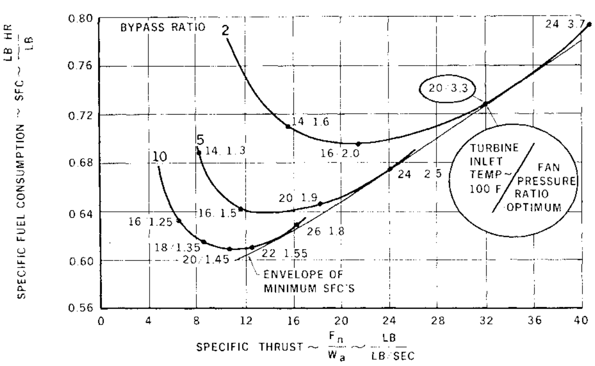
Fig. 5 Fan pressure ratio optimization.



a) cycle pressure ratio = 15



b) cycle pressure ratio = 20



c) cycle pressure ratio = 25

Fig. 6 Uninstalled turbofan cruise performance (M_n = 0.79)

bypass ratios and the corresponding lower specific thrust values (higher propulsive efficiency). This is a valid trend for the engine alone, but its installation into the aircraft introduces other considerations (external drag and engine weight) which influence these trends. Therefore, to determine the appropriate cycle for maximum aircraft range, the engine performance must be modified to account for the effect of the installation, and this installed performance then must be related to the variables that influence aircraft range. It will be shown how installed performance and relative range can be compared graphically on a common presentation.

Installed Specific Fuel Consumption

The installed specific fuel consumption was computed by correcting the SFC values shown in Fig. 6 for the effects of aerodynamic drag of the pylon, nacelle, and nacelle-pylon-wing-interference. The drag correction was obtained from Douglas studies of long-range transport aircraft designs and illustrated as a function of specific thrust in Fig. 7. The dotted line on this figure represents the drag characteristic if drag is assumed to be proportional to nacelle frontal area. The divergence of these two curves shows that a significant error would be introduced if this simplifying assumption were used.

Using the curve developed from the study designs, the installed specific fuel consumption (SFC_c) for any value of F_n/W_a is computed by the equation

$$SFC_c = \frac{SFC}{1 - (D/F_n)} \tag{6}$$

Installed Engine Thrust-to-Weight Ratio

Installed engine thrust-to-weight ratio influences aircraft range; therefore, a relationship was developed to permit engine specific thrust to be expressed in terms of installed engine thrust-to-weight ratio. This relationship was determined also from Douglas studies of long-range transport aircraft designs, and the results are shown in Fig. 8. Installed weight, as used in this relationship, includes the weight of the complete installation below the wing, including pylon, nacelle, engine, thrust reverser, and all engine-mounted accessories and systems. The installed thrust is the bare engine thrust minus the drag, discussed previously. The empirical relationship thus developed is based on engines of a fixed-cruise thrust and having a fixed-cycle pressure ratio. Engine weight data were not adjusted for variations in cycle pressure ratio; however, this is not considered as having a significant effect on the results of the study.

Relative Aircraft Range

The procedures described previously permitted installed engine fuel consumption to be plotted as a function of installed engine thrust-to-weight ratio. The reason for this manipulation is that aircraft range as computed by the Breguet range equation can be expressed also by these variables, thus providing a graphical means of assessing the effect of the engine cycle variables on aircraft range.

The basic Breguet range equation is written as follows:

$$R = [V_0/SFC_c][L/D]\ln[W_0/W_1] \tag{7}$$

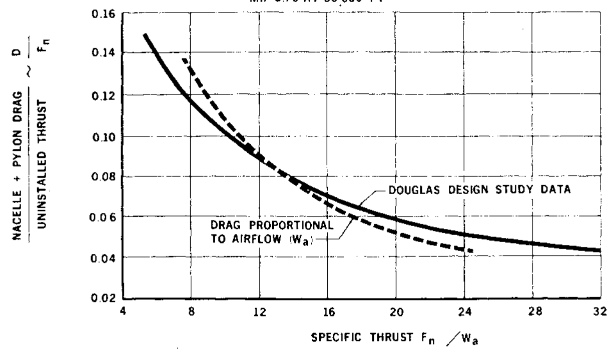


Fig. 7 Effect of specific thrust on installation drag.

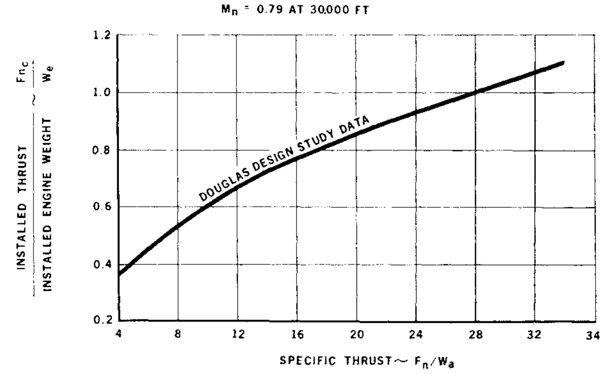


Fig. 8 Thrust/weight vs specific thrust.

Noting that installed thrust equals drag and lift equals weight, and if assumptions are made that gross weight at start of cruise remains constant, and weight of installed engines plus fuel remains constant, the Breguet equation may be written in the following form:

$$\frac{R}{R'} = \frac{V_0/SFC_c}{R'}\,\frac{L}{D}\,\ln\left\{\frac{1}{1-(W_{ef}/W_0)+[1/(F_{nc}/W_e)(L/D)]}\right\} \tag{8}$$

If appropriate assumptions are made for the values of V_0, L/D, W_{ef}/W_0, and R', Eq. (7) reduces to:

$$SFC_c = \frac{K_1}{(R/R')}\,\ln\left\{\frac{1}{K_2+[K_3/(F_{nc}/W_e)]}\right\} \tag{9}$$

where K_1, K_2, and K_3 are constants.

From this equation, lines of constant relative range R/R' can be plotted as a function of SFC_c and F_{nc}/W_e. Figure 9 shows both installed engine performance and relative range superimposed on common plots for various cycle pressure ratios. The effects of turbine temperature, bypass ratio, and cycle pressure ratio on range can be seen readily from these curves. Maximum relative range for a given turbine temperature occurs at the point of tangency of the temperature and range curves. The bypass ratio line passing through that point of tangency is the bypass ratio consistent with that range.

Conclusions

The most important conclusion to be drawn from this work is that, when the installation effects of an engine are considered, any selected turbine inlet temperature prescribes an optimum bypass ratio for maximum range. This is unlike the trend noted for an uninstalled engine, which suggests that an infinite bypass ratio is required for maximum range. Analysis of this optimization characteristic reveals the following trends:

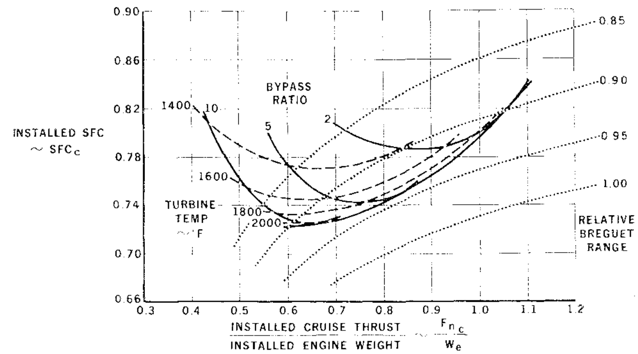
1) As turbine inlet temperature increases, the corresponding bypass ratio for maximum range must increase also. This is required to prevent excessive residual velocity losses.

2) An increasing turbine temperature (along with the corresponding bypass ratio) increases range, but this trend diminishes rapidly at the higher values of turbine temperature. This characteristic results from the increasing η_g trend with turbine temperature which also exhibits diminishing improvements at high turbine temperatures.

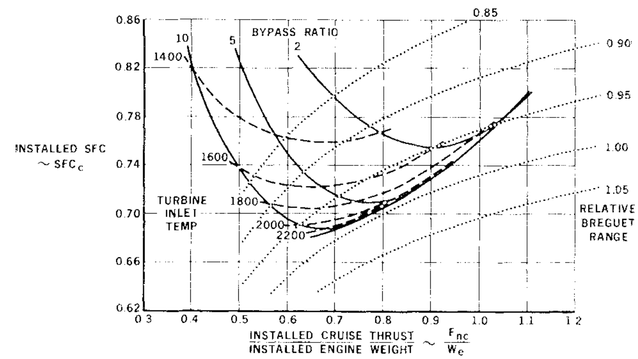
3) The range change with bypass ratio is relatively flat, and the bypass ratio can vary significantly from the optimum value without serious range penalty.

4) A general increase in range occurs with increasing cycle pressure ratio, and the effect is more pronounced at higher turbine temperatures.

This discussion is intended to provide an insight into the thermodynamic and aerodynamic basis for the behavior of turbofan engines in long-range aircraft, and to describe a method by which the optimum combination of cycle variables can be approximated for an assumed transport mission.



a) cycle pressure ratio = 15



b) cycle pressure ratio = 20

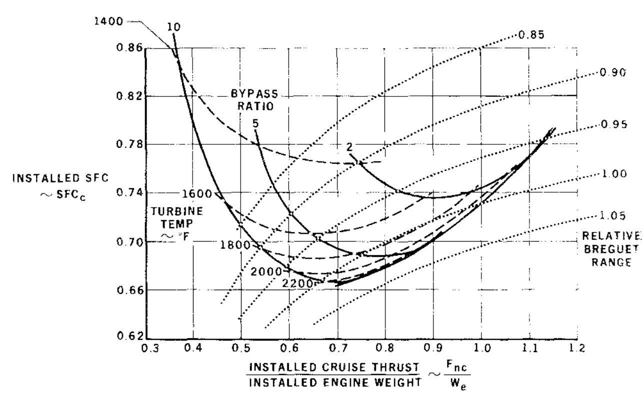


FIGURE 9C

c) cycle pressure ratio = 25

Fig. 9 Effect of engine cycle on range (M_n = 0.79 at 30,000 ft).

The absolute values presented in the foregoing curves are not necessarily representative of an actual range problem, but rather are intended to illustrate the relative influence of the variables affecting performance. The study of any particular range problem using the procedures discussed herein requires that appropriate inputs be made which represent the technologies of engine and airframe designs being considered.