A Method of Determining Propulsion System Requirements for Long-Range, Long-Endurance Aircraft

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A method of evaluating analytically the matching problems of engine design vs airplane design to obtain maximum airplane range or endurance is described and demonstrated. The method uses a presentation of airplane performance which facilitates the determination of range or endurance as a function of the primary engine performance and engine weight requirements of the airplane. The method is demonstrated by matching a series of theoretically defined regenerative and nonregenerative turbofan engines with varying bypass ratios to both laminar and turbulent aircraft of typical design at a typical cruise speed. All of the engines are front fan configurations with fans geared to the gas generators at bypass ratios greater than 4.0. Regenerative engines are designed with liquid metal coupled heat exchangers, which remove heat from the exhaust stream and add heat to the burner inlet. Engine designs are based upon thermodynamic cycles defined for minimum specific fuel consumption at cruise and cycles at sea-level static, consistent with constant fan and gas generator exhaust areas. Bypass ratios from 7 to 11 at takeoff conditions result in maximum range.

Nomenclature

$\mathcal{R}$ = aspect ratio
B = bypass ratio, fan exhaust to gas generator airflow ratio = $\dot{W}_{af}/\dot{W}_{agg}$
C_D = drag coefficient
C_L = lift coefficient based on wing planform area
D = diameter, ft
F = net thrust, lb
K = const
L = length, ft
M = Mach number
S = wing planform area, ft^2
S_W = wetted area, ft^2
SFC = specific fuel consumption, lb fuel/hr/lb thrust
V = velocity of flight, knots
W = weight, lb
$\dot{W}_a$ = air weight flow, lb/sec
e = natural logarithmic base
e_w = wing efficiency factor
h = height, ft
n = ultimate load factor
q = dynamic pressure, lb/ft^2
t = radial thickness of airstream, ft
π = 3.1416

Subscripts

cl = climb at cruise flight condition
cr = cruise flight condition and power setting
lam = laminar flow control, laminar
max = maximum value
min = minimum value
nor = normal power setting (maximum speed and maximum continuous turbine inlet temperature)
to = takeoff (sea-level static conditions)
turb = turbulent
req = required
str = structure
a = air
b = boattail
crit = critical
des = design point at reduced power cruise condition
e = engine
f = fan
fus = fuselage
g = gross, ground run
gg = gas generator
i = installed
j = exhaust stream
n = nacelle
0 = freestream
pl = payload
s = suction system
t = total
w = wetted

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Introduction

THIS paper describes a method of evaluating analytically some of the interface and matching problems of engine design vs airplane design to obtain maximum range or endurance. The method uses a simplified airplane design study that is presented in a manner designed to facilitate the determination of range or endurance as a function of the primary engine performance and engine weight requirements of the airplane. Use of the method described simplifies the problem of defining desirable propulsion system design characteristics and allows more rapid comparison of either theoretical or existing engine designs in terms of airplane range or endurance.

The airplane design is considered only to the extent that it affects requirements to be satisfied by the propulsion system in order to achieve maximum airplane range or endurance Wing loading is the primary airplane design parameter, which is varied because of its effect on engine size requirements and matching characteristics. Since laminar flow control is considered as a means of meeting desired airplane performance, the wing loading parameter assumes increased importance because of the lower drag per unit area of the wing compared to turbulent airplanes.

Some comparisons are made of the range and endurance of typical theoretical airplanes with and without laminar flow control. These comparisons are made using consistent series of both regenerative and nonregenerative turbofan engines specifically designed for the airplane. Thus, the method of studying matching problems is demonstrated. The analyses

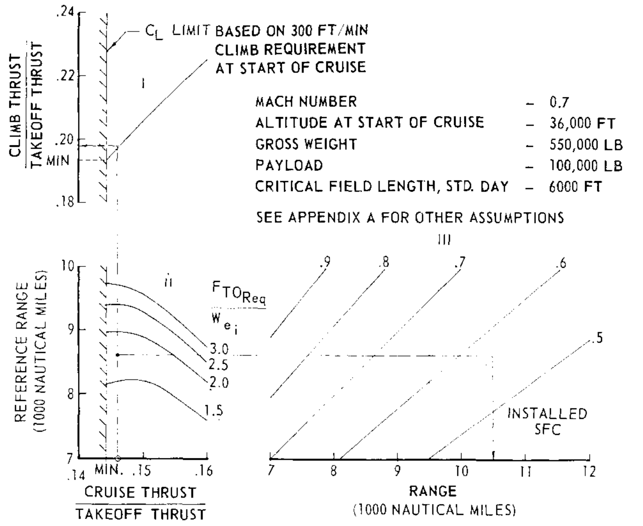

Fig. 1 Range in terms of propulsion requirements, laminar airplane.

are based upon a complex set of design assumptions, listed in Appendix A, which must be considered when evaluating results. The results indicated do not necessarily show the longest possible range or endurance for a given airplane or engine type. Furthermore, other engine types such as the regenerative turboprop engine can give longer endurance and longer range than is shown in this paper. The method of analysis, however, can be used with other engine types as well. Turbofan engine design parameters of principal concern for the comparisons shown are fan bypass ratio, fan pressure ratio, and compressor pressure ratio as well as regenerative and nonregenerative cycles.

Studies of advanced engine and aircraft concepts have been previously published to outline the potential of long-range, long-endurance aircraft and some of the problems associated with the design and operation of this airplane type.[1-4] Propulsion systems considered have included regenerative[4,5] and nonregenerative turbofan[3,5] engines as well as other types of propulsion such as turboprops,[1,3,4] tip turbine turbofans,[2] and even piston engines.[3] The comparisons in this paper supplement these studies.

Method of Analysis

The method used to analyze airplane-engine matching for optimum airplane performance is divided into three parts.

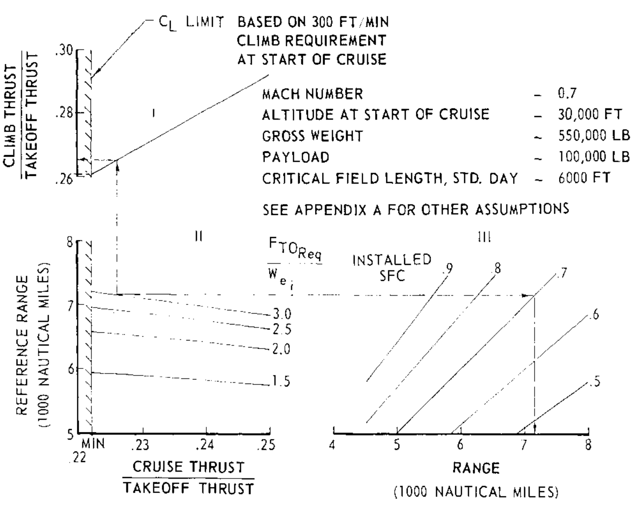

Fig. 2 Range in terms of propulsion requirements, turbulent airplane.

1. Airplane Range or Endurance Presented in Terms of Propulsion Requirements

Range or endurance resulting from varying airplane wing loading, installed engine weight, and specific fuel consumption is calculated and presented in terms of propulsion parameters required to obtain that range or endurance. The presentation is useful as a means of rapidly performing studies of airplane-engine matching and the effect of engine design on airplane performance. The airplane performance is obtained by fixing all of the other airplane design and mission parameters such as gross weight, payload, design Mach number, cruise Mach number, critical field length, altitude at start of cruise, climb rate capability at start of cruise, load factor, and cargo space required. Engine performance and weight requirements are delineated as installed cruise thrust-to-takeoff thrust ratio, installed engine takeoff thrust-to-installed weight ratio, and installed specific fuel consumption.

2. Installed Engine Performance

Installed specific fuel consumption and takeoff thrust-to-weight ratio are calculated as functions of cruise-to-takeoff thrust ratio and bypass ratio. Nacelle surface area and drag coefficient are found for each installed engine design as bypass ratio and cruise-to-takeoff thrust ratio are varied.

The engines used to demonstrate the method are designed at a partial power cruise design point where the design compressor pressure ratio, fan pressure ratio, and turbine inlet temperature are chosen to give minimum specific fuel consumption for each design bypass ratio at or below operating temperature limits.

3. Airplane-Engine Matching for Maximum Range or Endurance

The range or endurance study in terms of propulsion parameters is combined with installed engine performance to determine the best combination of airplane and engine parameters to give maximum range or endurance. The maximum range or endurance obtainable with each engine bypass ratio is plotted vs bypass ratio to find the effect of bypass ratio on maximum range. The maximum range at each bypass ratio is dependent on altitude at start of cruise, airplane wing loading and cruise power setting (or cruise-to-takeoff thrust ratio), engine size, installed engine weight, and installed cruise specific fuel consumption. The method of matching airplanes with engines is in general agreement with the rules described in Ref. 2.

Range in Terms of Propulsion Requirements

General Discussion

An analysis of airplane performance using the Breguet range equation can show the relationship among the three principal engine performance parameters at cruise which is required to obtain maximum range or endurance. These three engine performance parameters are 1) installed cruise thrust-to-sea-level static takeoff thrust ratio, 2) installed engine takeoff thrust-to-installed weight ratio, and 3) installed specific fuel consumption at cruise. The relationships among these parameters are presented in Figs. 1 and 2, as obtained from the equations in Appendix A. Such a study shows the relative importance of these engine performance parameters and provides a rapid means of matching engines with airplanes. By combining the airplane performance study results with engine designs, a tradeoff of range or endurance vs an engine design parameter such as bypass ratio is possible after engine designs are defined.

Discussion of Curves

Figures 1 and 2 are shown only at typical altitudes at start of cruise, 36,000 ft for the laminar airplane and 30,000 for the turbulent airplane. These altitudes are chosen because, for most reasonable combinations of airplane and engine parameters used with the listed assumptions in Appendix A, near maximum range is achieved at these particular altitudes for the two airplane types.

The minimum climb-to-takeoff thrust ratio required to satisfy a 300-fpm climb requirement at start of cruise is obtained from segment I of Figs. 1 and 2 when the cruise-to-takeoff thrust ratio required in level flight is known. This curve, then, indicates the minimum allowable thrust ratio at normal power and at start of cruise for any given wing loading or cruise thrust ratio. (The relation between cruise thrust ratio and wing loading is shown in Fig. 3). The segment I curve is used to find engine power setting at start of cruise and determines engine size[2] when the engine is sized at the cruise condition as explained in the numerical example and the engine sizing sections.

Segment II of Figs. 1 and 2 shows the effect of wing loading or cruise-to-takeoff thrust ratio upon range at a reference value of specific fuel consumption (0.7 lb fuel/hr/lb thrust). Cruise thrust (a function of airplane drag at cruise) is varied by changing wing loading at constant gross weight, thus varying wing area [Eq. (A5) in Appendix A shows this relationship]. Takeoff distance also is held constant as wing area is varied by varying static takeoff thrust. The resulting cruise thrust and takeoff thrust then define the cruise-to-takeoff thrust ratio. Takeoff engine thrust-to-weight ratio defines installed engine, nacelle, and strut weight and allows calculation of fuel available for cruise and reference range by use of the Breguet range equation. The curve includes the tradeoff of wing area vs wing and tail weight, the effect of induced drag on lift-to-drag ratio, the effect of propulsion weight on wing weight, and the effect of laminar flow control on wing, tail, and pumping system weight. The curves are used to find the cruise power setting and wing loading combination, which give the maximum range as described in the example.

The effect of installed engine specific fuel consumption when different from its reference value upon range or endurance is found by the use of segment III obtained from the Breguet range equation. Of course, the curve can be plotted in terms of endurance instead of range if desired, since endurance is range divided by flight speed. A lower limit on both cruise and climb thrust requirement for a given takeoff thrust is set by a lift coefficient limit to avoid excessive shock losses.

Effect of Lift Coefficient Limit

As altitude at start of cruise increases at a constant lift coefficient, wing loading decreases because wing loading must vary proportionally with dynamic pressure to hold C_L constant. So at the constant lift coefficient limit, wing loading decreases with increased altitude at start of cruise. Cruise-to-takeoff thrust ratio as a function of wing loading is shown in Fig. 3. On this curve the minimum value (at C_L limit) of cruise thrust-to-takeoff thrust ratio is seen to increase with increasing altitude at start of cruise. Likewise, minimum climb thrust-to-takeoff thrust ratio at the lift coefficient limit varies in the same way, since Eqs. (A4) and (A5) of Appendix A differ only by a constant term.

Portions of segment II of Figs. 1 and 2 are plotted in Fig. 4 to show the effect of altitude, lift coefficient limit, and laminar flow control on reference range. Reference range decreases with increasing altitude at the lift coefficient limit for both laminar and turbulent airplanes. In the case of turbulent airplanes, the lift coefficient limit defines the maximum reference range. On the other hand, the curves for laminar airplanes show increased reference range at reduced lift coefficients (lower wing loading and higher cruise thrust ratio). The marked peaking of the curves in the laminar case is the result of the cleanness of the laminar airplane. With increasing altitude, wing loading at the peak of the curves decreases

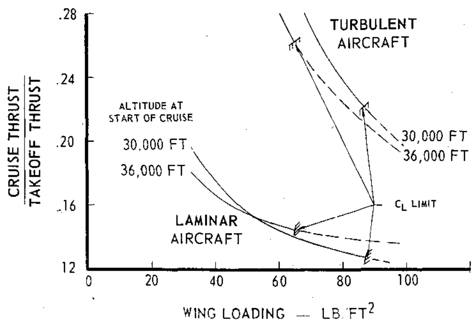

Fig. 3 Cruise-to-takeoff thrust ratio vs wing loading.

substantially. Because of these characteristics, the problem of matching the engine to the laminar airplane for maximum range or endurance becomes more difficult, since the longest ranges are obtained near the peak of the curves of Figs. 1 and 4. Maximum range varies with thrust ratio, altitude, and wing loading, as well as installed specific fuel consumption and installed thrust-to-weight ratio.

Thrust Ratio Requirements

Maximum range occurs at a lower installed cruise-to-takeoff thrust ratio and lower wing loading for the laminar airplane than for a turbulent airplane. In other words, the laminar airplane requires less thrust to cruise at a given condition than a turbulent airplane with the same gross weight and takeoff thrust. The lift coefficient limit (set at $C_{L\max\,cr} = 0.40$ for this study) allows use of the lower cruise thrust ratio desired for laminar aircraft and applies a reasonable limit to both airplane types.

Engine Size

An engine, which has a lower thrust ratio at normal power setting and at the cruise flight condition than the minimum climb thrust ratio required by the lift coefficient limit, is sized at the climb condition required at the start of cruise. In this case, engine thrust (and size) is increased by the ratio of minimum required climb thrust to cruise thrust at normal power. Engine weight is increased proportionally with the thrust increase.

Thrust-to-weight ratio plotted in Figs. 1 and 2 accounts for installed takeoff thrust required to meet exactly the critical field length condition. If the engines are sized at the climb condition noted previously, the takeoff thrust is greater than required. Then the thrust-to-weight ratio used in reading

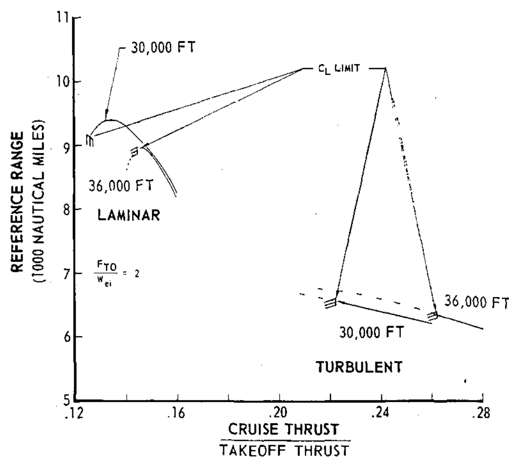

Fig. 4 Effect of altitude at start of cruise and airplane type on reference range vs cruise-to-takeoff thrust ratio at 0.7 Mach number, 550,000 lb gross weight, 100,000 lb payload.

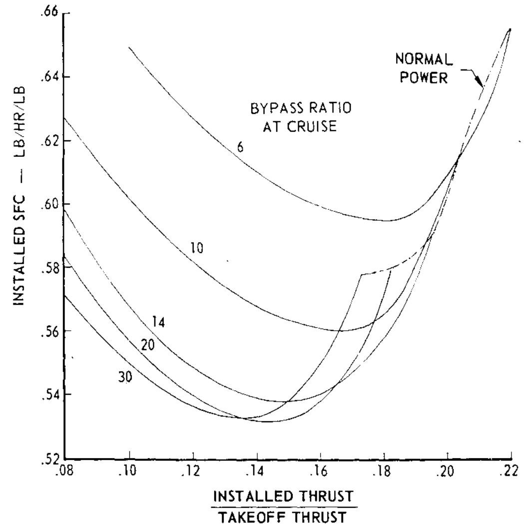

Fig. 5 Installed specific fuel consumption of podded regenerative turbofan engines designed for minimum SFC at cruise design point, 36,000-ft altitude at start of cruise and 0.7 Mach number, engine size variable with cruise power setting.

the curves of segment II of Figs. 1 and 2 is smaller than that originally calculated because of the increased engine weight for the same required takeoff thrust. Therefore, thrust-to-weight ratio used in reading the curve is the installed thrust-to-weight ratio times the ratio of installed normal power cruise to takeoff thrust divided by the ratio of minimum required climb to takeoff thrust. The airplane is then capable of taking off in less distance than required by the given critical field length.

If installed thrust ratio at normal power setting of a given engine at the cruise flight condition is greater than the minimum climb-to-takeoff thrust ratio required by the lift coefficient limit, the engine is sized at takeoff. Thrust-to-weight ratio used in reading Figs. 1 and 2 is then installed takeoff thrust per installed engine and pod weight. Thrust available for cruise in this case is then greater than required by the wing loading, and the engine is throttled back from normal power setting to obtain the required cruise thrust. Wing loading can be varied to give the optimum match of airplane and engine characteristics resulting in maximum range. This optimum occurs near the peak, but not necessarily at the peak, of the curves of reference range vs thrust ratio. The reason for this, as shown in the numerical example of sizing and matching, is the tradeoff between engine weight and installed specific fuel consumption. Thus, the best match depends upon the slope of the thrust vs specific fuel consumption curve (Fig. 5) at the cruise condition.

Assumptions

A number of assumptions must be made, based upon typical airplane requirements in order to calculate range or endurance for presentation in terms of engine requirements. These include cruise Mach number, minimum altitude at start of cruise, airplane gross weight, payload, critical field length, number of engines desired, and structural design requirements such as load factor, landing gear type, and maximum design payload. Assumptions used in the analysis are listed in detail in Appendix A together with the basic equations used for calculation of airplane performance.

A critical field length of 6000 ft on a standard day was assumed for convenience to represent critical takeoff conditions. This assumption approximates a critical field length of 8000 ft on a hot day when the effects of temperature on engine and airplane performance are taken into consideration.

Laminar Flow Control

The range or endurance vs propulsion requirements study (Fig. 1) is based upon consideration of the airplane and the complete laminar flow control pumping system (if used) as separate from the main engine pods and struts. Cruise thrust ratio required of the engines is reduced for the laminar aircraft by the additional cruise thrust produced by the laminar flow control pumping system. Weight of fuel used by the pumping system to reheat engine bleed air required for the pumping system turbines and the pumping system weight both decrease fuel weight that is available to the propulsion engines. Thus, the energy expenditure required for operation of the laminar flow control suction system is reflected in a range decrease for a given propulsion engine specific fuel consumption, as is indicated by Eqs. (A9) and (A10) of Appendix A. The effect of bleed on the performance of main propulsion engines is taken into account by decreasing engine thrust and increasing specific fuel consumption at constant fuel flow. As a result of these assumptions built into the parametric curves of Figs. 1, 3, and 4, the same installed engine performance can be used to compare both laminar and turbulent aircraft.

The effective wetted area of laminar surfaces is assumed to be 1.92 times wing planform area. This figure is representative of laminar flow control applied to both surfaces of the wing and tail.

Installed Engine Performance and Weight

Uninstalled Design Point Engine Performance

An engine parametric study is used at the cruise flight condition to define combinations of engine design parameters at cruise which give minimum cycle specific fuel consumption consistent with turbine inlet temperature limits and reasonable compressor weight and efficiency. Some of the results of the parametric study are plotted in Figs. 6 and 7, and much more detailed results are to be found in Ref. 6.

There are significant differences between the optimum engine cycle design parameters of regenerative and nonregenerative turbofan engines, so much so that to combine a regenerator with components of a nonregenerative engine seldom, if ever, results in even an approximate comparison of the capabilities of each.

The lowest specific fuel consumption of regenerative engines is obtained at higher turbine inlet temperature[1,5,6] and much lower compressor pressure ratio.[1,5,6] The optimum pressure ratio of a nonregenerative engine is set by aerodynamic, mechanical, and weight considerations because specific fuel consumption improves at a decreasing rate, and specific thrust increases as compressor pressure ratio increases.[5,6] On the other hand, the pressure ratio of the regenerative turbofan is set largely by the maximum turbine inlet tempera-

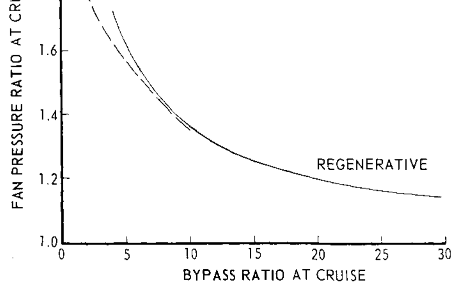

Fig. 6 Fan pressure ratio vs bypass ratio at cruise design point for typical series of turbofan engines.

ture at cruise and regenerator effectiveness, both of which should be as high as possible for minimum specific fuel consumption but consistent with structural, weight, and volume considerations.

Consideration of installed specific fuel consumption tends to increase the pressure ratio, resulting in decrease of engine size and engine plus regenerator weight. Since the pressure ratio of the series of regenerative engines is selected only on the basis of minimum specific fuel consumption, the installed specific fuel consumption could possibly be further reduced by additional analysis based upon installed performance.

For minimum specific fuel consumption, an increase in bypass ratio must be accompanied by a decrease in fan pressure ratio[4,5] of both regenerative and nonregenerative turbofan engines as indicated in Fig. 6. The fan pressure ratio of a regenerative turbofan engine should be lower than that of a comparable nonregenerative engine at minimum specific fuel consumption, since the regenerative engine has less energy in its exhaust stream and requires lower fan exhaust velocity to balance exhaust velocities as required for best propulsive efficiency. However, the much higher turbine inlet temperature of the regenerative engine has the opposing effect of adding to the gas generator exhaust energy and offsets this characteristic to the extent that optimum design fan pressure ratio of the nonregenerative engine is only slightly lower than that of the regenerative engine.[5] The same result is found when design point specific thrust is similarly plotted vs bypass ratio for the two engine types.[6] The reasons for this result are the same as those for the similarity in fan pressure ratio explained previously.

Figure 7 defines the uninstalled specific fuel consumption at the cruise design points for each engine type vs bypass ratio. This curve is largely independent of the engine configuration because no off-design performance is shown. For example, the same data could be used for either a forward or aft fan arrangement and for fixed or free turbine engines.

Uninstalled Engine Performance at Off-Design Conditions

Cycle calculations at sea-level static takeoff and for normal power setting at the cruise flight condition are required in order to define specific thrust (thrust per air flow) and pressure ratio at these conditions so that specific weight and takeoff thrust may be estimated. Some knowledge is required of the engine control mode, so that engine size will be minimized. Previous studies of regenerative turboprop and turbofan engines[1,4] have shown that the regenerative turbofan engine should be operated at essentially constant turbine inlet temperature with thrust being varied by varying speed and, therefore, airflow. The gas generator, then, is designed to operate at constant turbine inlet temperature except at takeoff power setting where the temperature is allowed to increase from 1990° to 2060°F. At sea-level static, the fan must be unloaded to reduce work required from the turbine

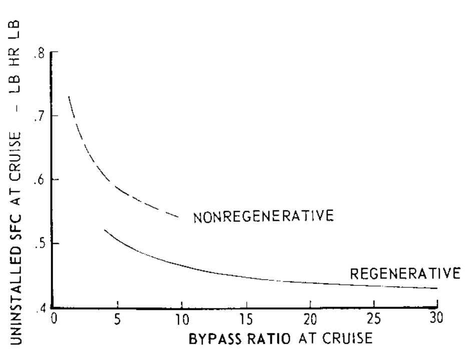

Fig. 7 Uninstalled design point cruise specific fuel consumption vs bypass ratio at cruise for typical series of turbofan engines, 36,000-ft altitude at start of cruise, 0.7 Mach number.

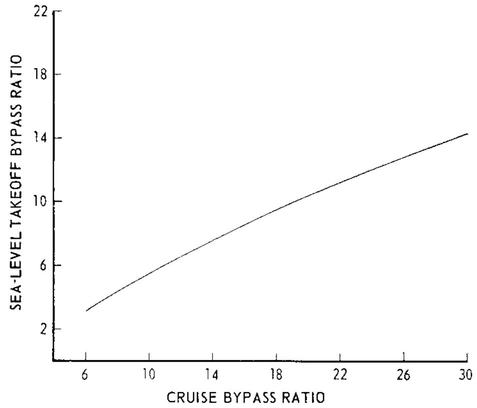

Fig. 8 Sea-level takeoff bypass ratio vs cruise bypass ratio for regenerative turbofan engines.

so as to match fan work to that available with the gas generator exhaust nozzle area held constant. Fan exhaust area is held constant, and both bypass ratio and fan pressure ratio are allowed to vary in order to match fan work to that required and to match fan exhaust area calculated from the fan exhaust flow to the required exhaust area. Airflow through the fan is varied by use of variable inlet guide vanes. A fixed turbine design was assumed for the regenerative engines used to demonstrate an engine comparison. Figure 8 shows the relation between bypass ratios at the cruise design point and at sea-level static takeoff power.

The control mode of the nonregenerative engines studied is more like current practice. A free turbine configuration is assumed with fixed gas generator and fan exhaust areas. Turbine inlet temperature decreases with speed and airflow of the gas generator as power setting is reduced with fixed turbine nozzle area. Fan turbine work is defined by its fixed nozzle area and fixed exhaust area, whereas fan speed and pressure ratio are determined by the airflow required to match the fan work to the fan exhaust area.

As an example of the method of obtaining engine off-design performance using bypass ratio as the primary variable for study, a part power design point at cruise is selected for each of several bypass ratios with compressor pressure ratio, fan pressure ratio, and turbine inlet temperature selected to give minimum cycle specific fuel consumption for that bypass ratio within established limits.

Performance is then calculated at sea-level takeoff and normal power cruise for several gas generator airflows along the engine operating line as determined by engine geometry and control mode. Takeoff thrust, size, and weight are calculated for each gas generator airflow assumed. Reference 7 lists the equations used in the calculation. Each point thus calculated represents an engine designed for minimum specific fuel consumption at cruise but at a different cruise power setting. The method is discussed in more detail in Ref. 6.

Regenerator Design

The size and weight of the liquid metal finned tubular recuperator (regenerator) are calculated for given effectiveness, airflow, gas flow, pressure losses, and gas thermodynamic properties at the cruise design point by applying data given in Refs. 8–10 to basic heat-transfer analysis.

The specific regenerator design point chosen for the study is based upon an effectiveness of 0.80 at the cruise design point of the engine using liquid metal finned tubular heat-transfer surfaces. Pressure losses at the cruise part power design point are 2 and 4% for the cold air and hot gas heat exchangers, respectively. The choice of these design conditions was based upon judgment rather than upon a tradeoff study of the effect of pressure losses and effectiveness on installed performance and weight or on range.[1] Further im-

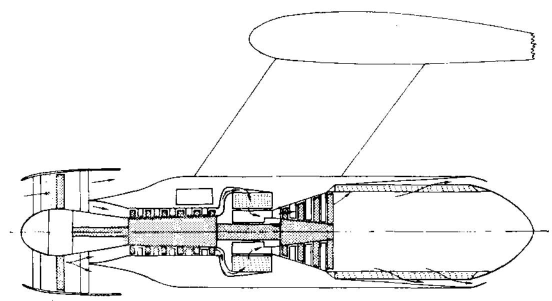

Fig. 9 Installation schematic of regenerative turbofan with gear-driven fan and finned tubular liquid metal heat exchangers at burner entrance and turbine discharge.

provements in range with fully optimized regenerator designs appear to be likely. For the engines considered, the regenerator core specific weight was 49 lb/lb airflow/sec, and specific volume was 0.86 ft^3/lb airflow/sec.

Figure 9 shows a schematic of the regenerative engine configuration used. One heat exchanger is outside the burner area, and the other is located aft of the turbine. The regenerator transfers heat from the turbine exhaust to the burner inlet.

Nacelle Drag

Nacelle and strut drag estimation is outlined in Appendix B. Engines with bypass ratios below 3.0 are designed with long fan ducts and have no scrubbing losses at the fan exit stream. Therefore, the value of M_{jf}/M_0 is zero when bypass ratio is less than 3.0.

Installed Engine Weight

Installed engine thrust-to-weight ratio is calculated as

$$F_{to}/W_{ei} = F_{to}/(1.3\ W_e + 5.0\ S_{wtn}) \tag{1}$$

where F_{to} includes inlet duct and exhaust duct pressure losses at takeoff.

Installed engine thrust to weight ratio is assumed constant as size of the same engine design varies in spite of the fact that bare engine thrust-to-weight ratio is usually nonlinear with change in engine size. An illustration of the small error involved is found in Appendix C of Ref. 6.

Bare engine weight is estimated from studies of weight of engine components. The weights of fans, gear boxes, and gas generators are estimated from required pressure ratios and airflows. Constants for the following general equation are found by studying a series of similar engines with known component weights. The equation with known constants is then used to estimate the weight of study engines.

The equation is

$$W_e = K_1 D_f^2 + K_2 D_{gg}^2 L_{gg} + K_3 \dot{W}_{gg\text{des}} \tag{2}$$

where $\dot{W}_{gg\text{des}}$ is airflow through the gas generator at the reduced power cruise design point. The constant K_1 accounts for gear box weight as well as fan weight, and K_3 accounts for regenerator weight.

Both installed and uninstalled thrust-to-weight ratios at takeoff are plotted in Fig. 10 vs bypass ratio at takeoff. Regenerative engines are represented by solid lines and nonregenerative by dashed lines, as is the practice in all of the figures except Figs. 1–5.

Results of Installed Engine Performance

Results are shown in Figs. 5 and 11 that relate installed specific fuel consumption at cruise vs installed cruise-to-takeoff thrust ratio for various bypass ratios at the cruise design point. Each cruise bypass ratio line is based upon a specific engine thermodynamic cycle at the cruise power setting that minimizes uninstalled specific fuel consumption at the flight condition. For a given cruise bypass ratio, engine size varies with thrust ratio, thus causing the variation of installed cruise specific fuel consumption. Minimum installed specific fuel consumption continues to decrease with bypass ratio increases even up to ratios of 25 to 1 at cruise.

Matching of Airplane-Engine Characteristics

Data from the installed engine performance curves (Figs. 5 and 11) are combined with the airplane performance (Figs. 1 and 2) to find the cruise-to-takeoff thrust ratio, engine size, and airplane wing loading, which give maximum range or endurance at each design bypass ratio and altitude at start of cruise. By repeating this procedure at various bypass ratios and altitudes, the cruise bypass ratio, which gives maximum range, is established. Figure 12 shows airplane range as a function of the equivalent takeoff bypass ratio at the combinations of altitude at start of cruise, engine size, and airplane wing loading resulting in the longest range. Cruise Mach number (0.7), gross weight (550,000 lb), and payload (100,000 lb) were selected in order to demonstrate the use of the method described in the paper. The selection of a different cruise speed, a minimum altitude at start of cruise less than 30,000 ft, a different basic airplane or engine design, any variation in the engine design assumptions, or the use of another engine type such as a turboprop could change the results shown in Fig. 12 significantly. The assumptions may be

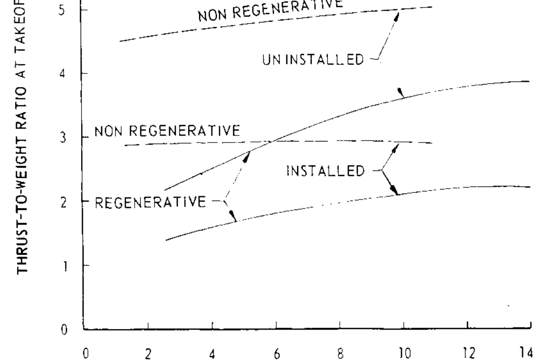

Fig. 10 Installed and uninstalled takeoff thrust-to-weight ratios vs bypass ratio at takeoff for typical series of turbofan engines.

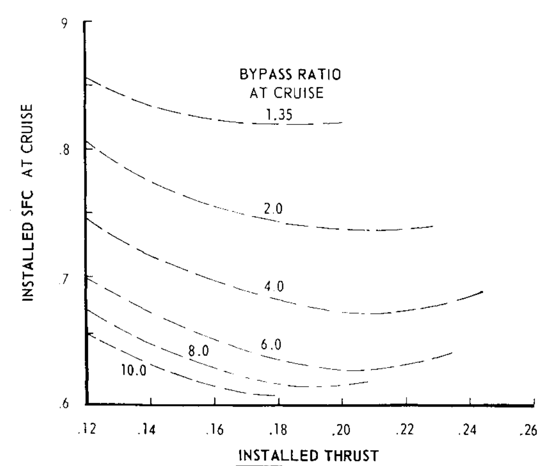

Fig. 11 Installed specific fuel consumption at cruise of a typical series of nonregenerative turbofans at 36,000-ft altitude at start of cruise and 0.7 Mach number.

varied by appropriate changes to the equations listed in Appendix A without altering the method described.

Of the three propulsion characteristics that cause the change in maximum range with variation in bypass ratio, installed specific fuel consumption and installed engine weight have been discussed previously. The effect of the third propulsion characteristic, namely, thrust match of the engine with that required by the airplane, is described in Fig. 13. In this figure, optimum thrust match of the engine and airplane at a given altitude at start of cruise occurs at the intersection of the line denoting the required thrust with the line denoting the actual engine thrust. At points above this intersection, the engine is sized by the climb requirement at start of cruise, and the airplane can exceed takeoff thrust requirements. However, at points below the intersection, the engines are sized at takeoff and throttled back both during cruise and at the maximum required climb. Engine size required by the airplane design is represented by the required thrust line. Power setting required at cruise tends to decrease at higher altitudes with the same airplane type. This power setting tends to decrease when comparing a laminar airplane relative to a turbulent airplane at the same bypass ratio. Laminar airplane designs tend toward lower wing loading, higher bypass ratio engines, or cruise at higher altitude to match thrust of the engines to that required by the airplane.

Bypass ratio for maximum range in Fig. 12 is higher than the bypass ratio that matches thrust requirements in Fig. 13. The bypass ratio for which this thrust match takes place does not result in the best tradeoff between installed specific fuel consumption and engine weight. Fan outer diameters are within a range that allows the engines to be installed in underwing pods of airplanes designed with the wing at the top of the fuselage.[6]

Numerical Examples

The following numerical examples demonstrate use of the curves to determine maximum range at given bypass ratios and given altitude at start of cruise. Assume a regenerative engine with design bypass ratio at cruise of 10.0 at 36,000-ft altitude and 0.7 Mach number to be matched to a laminar airplane.

Example 1

Minimum cruise-to-takeoff thrust ratio $(F_{cr}/F_{to})_{\min} = 0.144$ at C_L limit is from Fig. 1, segment II. The C_L limit is constant, being a function of wing design parameters (namely, taper ratio, sweep angle, thickness, and section), which are fixed. Minimum required climb-to-takeoff ratio $(F_{cl}/F_{to})_{req} = 0.194$ is from Fig. 1, segment I at C_L limit.

Installed engine normal power thrust-to-takeoff thrust ratio $(F_{nor}/F_{to})_e = 0.203$ is from Fig. 5 at cruise bypass ratio of 10.0 and at normal power (maximum value of cruise-to-takeoff thrust ratio). Normal power thrust ratio $(F_{nor}/F_{to})_e$ exceeds minimum required climb thrust ratio $(F_{cl}/F_{to})_{req}$. Therefore, the engine is sized at takeoff, and $F_{cr}/F_{to} = (F_{cr}/F_{to})_{\min} = 0.144$. Installed *SFC* at cruise is 0.566 lb/hr/lb from Fig. 5 at $F_{cr}/F_{to} = 0.144$, and $B = 10.0$. Thrust-to-weight ratio of installed engine, pod, and pylon from Fig. 10 is 1.77 based upon nominal engine size and takeoff bypass ratio of 5.55 (Fig. 8), which is equivalent to 10.0 cruise bypass ratio. Reference range from Fig. 1, segment II is 8640 naut miles at $F_{to\ req}/W_{ei} = 1.77$ and cruise thrust ratio of 0.144. Then from segment III at *SFC* of 0.565, actual range is 10,560 naut miles.

Table 1 Range calculation

F_{cr}/F_{to}	0.144	0.146	0.148	0.150
Reference range	8,640	8,670	8,680	8,640
SFC	0.566	0.565	0.564	0.564
Range, naut miles	10,560	10,590	10,620	10,580
F_{cr}/F_{nor}	0.711	0.719	0.729	0.738

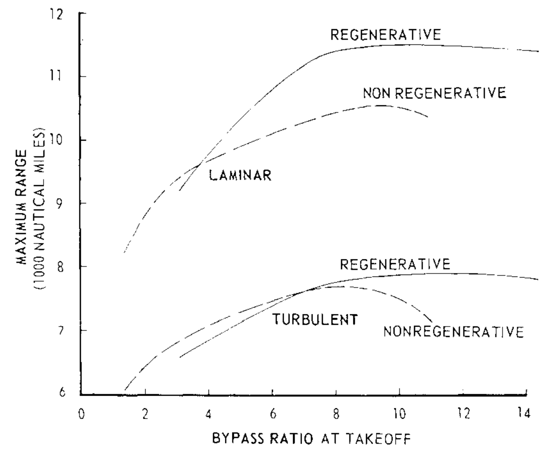

Fig. 12 Range vs bypass ratio for optimum wing loading, optimum engine size and optimum altitude at start of cruise, for 550,000 lb gross weight, 100,000 lb payload, and 0.7 Mach number.

The ratio of cruise to normal thrust is

$$F_{cr}/F_{nor} = (F_{cr}/F_{to})/(F_{nor}/F_{to}) = 0.144/0.203 = 0.711 \quad (3)$$

It is possible that higher cruise thrust ratio values may improve the attainable range even though reference range is reduced. This possibility is tested in Table 1 by using the procedure noted previously but assuming several higher values of cruise thrust ratio with no change in thrust to weight ratio.

The longest range for the assumed altitude at start of cruise, therefore, occurs at a cruise thrust ratio of 0.148 and at a cruise power setting of 72.9% of normal because of the combination of reduced specific fuel consumption and increased relative range at this condition. Thus, Fig. 12 shows a range of 10,620 naut miles at 5.55 bypass ratio, and cruise-to-normal thrust ratio is 0.729 in Fig. 13.

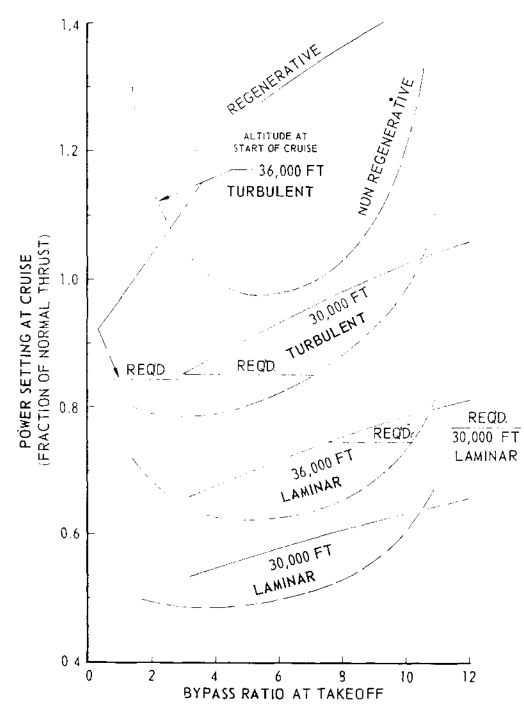

Fig. 13 Effect of bypass ratio, altitude at start of cruise and airplane design on power setting at cruise and thrust match of engine to airplane.

To verify that the calculated range is the maximum, other altitudes at start of cruise can be selected. With these altitudes, basic curves like Figs. 1 and 2 can be generated, and ranges for these altitudes can be found by following the procedures of this example.

Example 2

Assume that cruise bypass ratio is 20.0 at the same altitude and Mach numbers as example 1. In Fig. 1, segment II, $(F_{cr}/F_{to})_{min} = 0.144$ at C_L limit, and the minimum $(F_{cl}/F_{to})_{req} = 0.194$ is from segment I at C_L limit.

Installed engine normal power thrust-to-takeoff thrust ratio $(F_{nor}/F_{to})_e = 0.182$ from Fig. 5 at a cruise bypass ratio of 20.0. This is less than the 0.194 climb-to-takeoff thrust ratio required, so that the engine must be scaled up to meet the climb requirement. The engine before scale-up still exactly meets the takeoff and cruise thrust requirements. Thus, the scaled-up engine is oversized for takeoff and cruise. The scale-up is such that engine normal power thrust is increased to equal the thrust required for climb.

The engine cruise-to-takeoff thrust ratio for the scaled-up engine (normal power thrust = climb thrust required) is

$$\begin{aligned}(F_{cr}/F_{to}')_e &= (F_{cr}/F_{to})_{min}\,(F_{nor}/F_{to})_e/(F_{cl}/F_{to})_{req} \\ &= (0.144)\,(0.182)/0.194 = 0.135\end{aligned} \tag{4}$$

where the prime refers to the scaled-up quantity. The $(F_{cr}/F_{to}')_e = 0.135$ is used to find the engine installed *SFC*, which at cruise bypass ratio of 20 on Fig. 5 is 0.533 lb/hr/lb.

The reason that the cruise thrust ratio based upon the scaled-up engine (0.135) is less than the minimum cruise thrust ratio (0.144) is that the availabe thrust at takeoff power is increased by scaling up engine size, whereas the cruise thrust is held constant because it is defined by the airplane requirements. The scaled-up engine has more takeoff thrust available (F_{to}') than thrust required to meet the critical takeoff field length requirement (F_{to}). Required takeoff thrust is identical to its value prior to scale-up, because it is defined by the airplane critical field length requirement. Since the cruise and required takeoff thrust remain constant as the engine is scaled up, their ratio remains constant and equal to 0.144.

Installed thrust-to-weight ratio of the engine, pod, and strut calculated for a nominal sized engine with cruise bypass ratio of 20 and takeoff bypass ratio of 10.44 is 2.15 from Fig. 10. Engine thrust available at takeoff for the engine that is scaled up to meet the climb requirement may be calculated from

$$F_{to}' = (F_{to})(F_{cl}/F_{to})_{req}/(F_{nor}/F_{to})_e \tag{5}$$

The scaled-up engine is heavier in proportion to the thrust increase, assuming that engine thrust-to-weight ratio remains constant with the scale factor.[6] Thus

$$W_{ei}' = (W_{ei})(F_{cl}/F_{to})_{req}/(F_{nor}/F_{to})_e \tag{6}$$

Since the $F_{to\ req}/W_{ei}$ ratio used to plot Fig. 1 was based on the thrust required to meet the critical field length and the true installed engine weight, the thrust-to-weight ratio used in reading the curve is

$$\begin{aligned}F_{to\ req}/W_{ei}' &= (F_{to}/W_{ei})(F_{nor}/F_{to})_e/(F_{cl}/F_{to})_{req} \\ &= 2.15\ (0.182/0.194) = 2.02\end{aligned} \tag{7}$$

Reference range from Fig. 1, segment II is 8960 naut miles at $(F_{cr}/F_{to})_{min} = 0.144$, and $F_{to\ req}/W_{ei}' = 2.02$. The final range is 11,510 naut miles when adjusted for installed *SFC* of 0.533 in segment III of Fig. 1 (plotted in Fig. 12). The ratio of cruise to normal thrust plotted in Fig. 13 is, for this case,

$$(F_{cr}/F_{nor}) = (F_{cr}/F_{to})_{min}/(F_{cl}/F_{to})_{req} = 0.144/0.194 = 0.742 \tag{8}$$

or, from engine parameters,

$$(F_{cr}/F_{nor}) = (F_{cr}/F_{to}')_e/(F_{nor}/F_{to})_e = 0.135/0.182 = 0.742 \tag{9}$$

Because the engine is sized at the climb condition, this thrust ratio is equal to the required cruise power setting in Fig. 13 and remains constant as long as the engine is scaled up to meet the climb requirement. The amount by which engine matching (power setting) lines exceed the corresponding required power setting line indicates the amount of scaling required instead of power setting itself. This explains why "power setting" appears to exceed normal power in Fig. 13.

The scale factor on engine thrust is

$$F_{to}'/F_{to} = (F_{cl}/F_{to})_{req}/(F_{nor}/F_{to})_e = 0.194/0.182 = 1.066 \tag{10}$$

and the ratio of unscaled engine power setting at cruise to required power setting from Fig. 13 gives the same ratio. Thus

$$(F_{cr}/F_{nor})_{unscaled}/(F_{cr}/F_{nor})_{req} = 0.144/0.182/0.742 = 0.792/0.742 = 1.066 \tag{11}$$

Loiter Endurance

The selection of optimum airplane wing loading for maximum loiter endurance is simplified by use of curves similar to Figs. 1 and 2 at the speed and altitude desired for endurance. In such a case, the airplane is designed for maximum loiter endurance rather than maximum range.

If the airplane is designed for maximum range, the loiter endurance of that airplane is found as follows. Altitude and lift coefficient or speed for endurance are prescribed, and endurance is calculated from installed engine weight and available fuel. Specific fuel consumption is found from engine power setting at an installed thrust, which is equal to airplane drag at the endurance condition. Engine specific fuel consumption vs thrust is calculated at the endurance flight condition from cycle analysis.

An example of loiter endurance obtainable from some of the airplane-engine combinations is shown in Fig. 14. Loiter endurance is defined as the time spent on station. Thus, an 11,500-mile range means that no time is spent on station for the laminar airplane with regenerative engine. With the same airplane, a 6000-naut-mile range means that about 17 hr are spent on station. Longer endurance would be possible by reduction of payload, reduction of endurance speed and altitude, or design of the airplane wing loading to better match engine and airplane for endurance at reduced maxi-

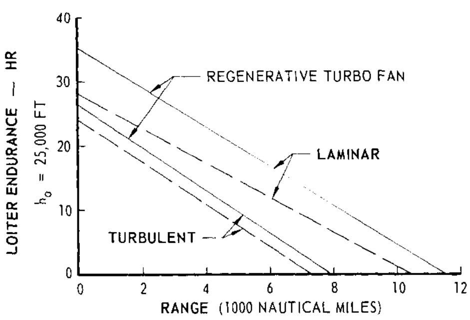

Fig. 14 Loiter endurance at 25,000 ft and at best endurance speeds vs range at 0.7 Mach cruise speed of 550,000 lb gross weight, 100,000 lb payload airplane, with 10.9 bypass ratio engines.

mum range. The slopes of endurance vs range curves vary between laminar and turbulent airplanes in a manner that improves the endurance of laminar airplanes with regenerative engines. This effect of regenerative engines on endurance would be more pronounced at lower payloads because increased fuel is available.

Conclusions

The method of matching airplane requirements with engine characteristics contained in this paper has proven useful for quickly making comparisons or analyses, which require the matching of many engine designs to a basic airplane design whose chief design criterion is maximum range or maximum endurance. The method lends itself to the study of a wide range of engine design and installation design variables and can handle the effect of change in takeoff altitude, nonstandard day performance, and climb rate requirements.

The analytical study is based upon a large number of assumptions, which define both the engine and airplane designs. Of course, variations in these assumptions would alter the results. Some conclusions may be derived from the results of the analysis used to demonstrate the airplane-engine matching method. The analysis gives an indication of the potential increase in airplane range and endurance that can result from the following design advancements: 1) increasing design bypass ratio of the turbofan engine, 2) designing a regenerative turbofan engine, 3) designing the airplane for laminar flow on wings and tail surfaces, and 4) combining the preceding design features so as to obtain the greatest improvements in airplane range or endurance.

Appendix A

Equations

$$W = 0.97\, W_g \tag{A1}$$

where 0.97 accounts for the fuel allowance for takeoff and climb

$$C_L = W/q_0S \tag{A2}$$

$$C_D = (C_L{}^2/e_w\pi \mathit{AR}) + (0.003\, S_{w\text{turb}}/S) + (0.00011\, S_{w\text{lam}}/S) \tag{A3}$$

where 0.003 = profile drag coefficient for turbulent flow, based on wetted area, and 0.00011 = profile drag coefficient for laminar flow, based on wetted area:

$$F_{cl} = [C_Dq_0S + (300\, W/101.3\, V_0) - F_s] \times (SFC_{\text{with bleed}}/SFC) \tag{A4}$$

where 300 = rate of climb at start of cruise, fpm, and 101.3 = conversion factor for knots to fpm

$$F_{cr} = (C_Dq_0S - F_s)\, SFC_{\text{with bleed}}/SFC \tag{A5}$$

Let

$$A = (1.21/C_{L\max to})\{0.067 + 0.025[(C_{L\max to}/1.21) - 1.13]\} \tag{A6}$$

$$\frac{F_{to}}{W_g} = \frac{A\exp\left[\dfrac{L_{\text{crit}}\,(A - 0.025)}{(L_{\text{crit}}/L_g)(W_g/S)(15.79/C_{L\max to})}\right] - 0.025}{\exp\left[\dfrac{L_{\text{crit}}\,(A - 0.025)}{(L_{\text{crit}}/L_g)(W_g/S)(15.79/C_{L\max to})}\right] - 1.0} \tag{A7}$$

where the constants 1.21 = correction to $C_{L\max to}$ for takeoff at 10% above stall speed, 0.067 = drag coefficient in the 3-point attitude during ground roll, 0.025 = rolling coefficient of friction, and 1.13 = lift coefficient in 3-point attitude during ground roll:

$$W_g15.79/SC_{L\max to} = 1.21(\rho V^2_{\text{stall}}/2)$$

where

$$V_{\text{stall}} = 29\,(W_g/SC_{L\max to})^{1/2},\ \text{fps}$$

$$\rho = 0.00238,\ \text{slug/ft}^3$$

$$\Delta W = W_g - W_{pl} - W_{ei} - W_{\text{crew}} - W_{\text{wing}} - W_{\text{horiz tail}} - W_{\text{vert tail}} - W_{\text{fuse}} - W_{\text{gear}} - W_{\text{controls}} - W_{\text{furn}} - W_{\text{air cond}} - W_{\text{anti ice}} - W_{\text{lam str}} - W_{\text{lam pump}} - W_{\text{fuel syst}} \tag{A8}$$

$$SFC_t = (SFC\, F_{cr} + \dot{W}_{s\,\text{fuel}})/C_Dq_0S \tag{A9}$$

Table 2 Airplane design assumptions

Symbol	Subject	Value
...	Minimum altitude for endurance	25,000 ft
$h_{0\min}$	Minimum altitude at start of cruise (used herein only as a constraint)	30,000 ft
W_g	Airplane gross weight	550,000 lb
W_{pl}	Payload	110,000 lb
$C_{L\max to}$	Maximum C_L at takeoff	1.93
$C_{L\max}$	Maximum C_L at cruise	0.40
...	Wing sweep	20°
...	Wing root thickness ratio	0.175
...	Wing taper ratio	0.3
AR	Aspect ratio	10
...	Vertical fin structure span-to-root thickness ratio	8.58
...	Vertical fin area	0.13 (S)
...	Horizontal tail structure span-to-root thickness ratio	35.9
...	Horizontal tail area	0.17 (S)
	Friction drag coefficient based on wetted area	
...	turbulent	0.003
...	laminar	0.00011
e_w	Wing efficiency factor	
...	turbulent	0.90
...	laminar	0.94
	Number of engines	
...	turbulent	6
...	laminar	4
n	Ultimate load factor	3.75
$W_{\text{lam str}}$	Weight *LFC* structure	0.40 ($S_{w\text{ lam}}$) lb
$W_{\text{lam pump}}$	Weight *LFC* pumping system	0.44 ($S_{w\text{ lam}}$) lb
...	Cargo compartment length	90 ft
...	Fuselage diameter	17 ft
L_{fus}	Fuselage length	159 ft
...	Maximum design payload	120,000 lb
...	Aft fuselage ramp	none
$L_{\text{moment tail}}$	Tail moment arm	88 ft
W_{furn}	Weight instruments, furnishings, and equipment	6910 lb
W_{crew}	Weight of crew, gear, and oil	2390 lb
...	Fuel flow service tolerance	5%
...	Takeoff, climb fuel, and fuel reserves per takeoff weight	0.04 or 22,000 lb fuel
$W_{\text{air cond}}$	Weight of air conditioning system	3,700 lb
$W_{\text{anti-ice}}$	Weight of anti-ice system	0.20 (S) lb
W_{gear}	Landing gear weight	0.035 W_g
W_{fus}	Fuselage weight	35,980 lb
$W_{\text{fuel syst}}$	Fuel system weight	0.00636 W_g
$S_{w\text{ lam}}/S$	Wetted laminar to planform area ratio	1.92
$F_s/S_{w\text{ lam}}$	Suction system net thrust per laminar wetted area, 36,000-ft altitude	0.113 lb/ft^2
$\dot{W}_{s\text{ fuel}}/S_{w\text{ lam}}$	Suction system fuel flow per laminar wetted area, 36,000-ft altitude	0.042 lb/hr/ft^2
L_{crit}/Lg	Critical field length per ground run distance	1.12 (6 engines) 1.18 (4 engines)
...	Fuselage wetted area	7000 ft^2
exp(x)	Exponent x to the e	e^x

$$\text{range} = [V_o(C_L/C_D)/1.05\ SFC_t] \ln 1/(1 + K_8 - \Delta W/W_g) \quad \text{(A10)}$$

where $K_8 = 0.040$ and accounts for the fuel allowance for takeoff, climb, and reserves.

Weight Equations

$$W_{\text{horiz tail}} = 11.32[2.1177 \times 10^{-6} W_g n (S)^{3/2}/L_{\text{moment tail}}]^{0.667} \quad \text{(A11)}$$

$$W_{\text{vert tail}} = 0.0393(W_g n)^{0.432}(0.13\ S)^{0.791} \quad \text{(A12)}$$

$$K_5 = f[(W_{pl\ \max} - W_{ei})/W_g] \quad \text{(A13)}$$

$$W_{\text{wing}} = 13.53\ K_5(W_g n/100\ S)^{0.60}(S/1000)^{1.25} \quad \text{(A14)}$$

$$W_{\text{controls}} = 0.745\ W_g^{0.623}[L_{\text{fus}} + (AR\ S)^{0.5}]^{0.224} \quad \text{(A15)}$$

$$W_{\text{fuel system}} = 0.00636\ W_g \quad \text{(A16)}$$

Airplane Design

Table 2 lists the design assumptions used for the airplane design.

Appendix B

Nacelle Drag

Nacelle and strut drag coefficient, based upon nacelle and strut wetted area, is defined as the sum of skin friction and pressure drag coefficients. In order to account for interference drag, the total nacelle and strut drag coefficient is multiplied by a factor of 2.0 which is typical of either forward or aft of wing pod and strut installations with 4 or 6 engines[11]:

$$C_{D_n} = 2.0[0.003 + (2.0\ S_{wbn}/S_{wen})(h_b/L_b)^2\ \{0.05 + 2.0\ t_{jf}/L_b\ [(M_{jf}/M_0)^2 - 1]\}] \quad \text{(B1)}$$

where S_{wbn} is boattail wetted area, and nacelle wetted area is

$$S_{wen} = 0.8\ \pi D_f^2 + \pi D_{gg} L_{gg} \quad \text{(B2)}$$

Total nacelle and strut wetted area is

$$S_{wtn} = 0.78\ D_f L_{gg} + S_{wen} \quad \text{(B3)}$$

Installed net thrust is then calculated from bare engine net thrust F_e:

$$F = F_e - C_{D_n} S_{wtn} q_0 \quad \text{(B4)}$$

Installed specific fuel consumption is main propulsion engine fuel flow in pounds per hour divided by installed net thrust of main propulsion engines in pounds.

References

[1] Brock, J. E. and Gershon, I. J., "Application of regenerative engines to projected Air Force missions," Society of Automotive Engineers Paper 807C, pp. 1–8 (January 1964).

[2] Boyles, R. Q. and Barnett, L., "Aircraft for long range and long endurance missions," Society of Automotive Engineers Paper 869D, pp. 1–11 (April 1964).

[3] Amsler, R. C. and Newton, J. S., "Long endurance aircraft, airplane design analysis," Norair Rept. NOR 63-109, Northrop Corp., Norair Div. (June 1963).

[4] Chapman, G. E., "The aircraft regenerative turbine engine: Where now?," Society of Automotive Engineers Paper 869-B, pp. 3–14 (April 1964).

[5] "Heavy logistics transport preliminary engine studies," Rept. 64-957, Pratt and Whitney Aircraft Div., United Aircraft Corp. (1964).

[6] Bogdanovic, J. A., Feder, A., and Wheaton, R. J., "A method of determining propulsion system requirements for long range-long endurance aircraft," AIAA Preprint 64-783 (September 1964).

[7] Wheaton, R. J. and Osmon, R. V., "Turbojet and turbofan general cycle IBM program," Norair Rept. NOR 62-206, Northrop Corp., Norair Div. (September 1962).

[8] Lambardo, S., "Component development of a liquid metal coupled regenerator system for aircraft engines," Society of Automotive Engineers Paper 807B, pp. 3, 4, 17 (January 1964).

[9] Cos, L. R., Gill, R., and Saco, F. W., "Advanced heat transfer elements for liquid metal regenerator systems," Wright Aeronautical Serial Rept. CTR. 00-272, Curtiss-Wright Corp., Wood-Ridge, N. J., pp. 1–95 (April 30, 1963).

[10] Kays, W. M. and London, A. L., *Compact Heat Exchangers* (McGraw-Hill Book Co., Inc., New York, 1958), Chap. II, pp. 16–23.

[11] Kuchemann, D. and Weber, J., *Aerodynamics of Propulsion* (McGraw-Hill Book Co., Inc., New York, 1953), Chap. IX, p. 232.