ANALYSIS OF VSTOL AIRCRAFT CONFIGURATIONS FOR SHORT HAUL AIR TRANSPORTATION SYSTEMS

R. Gallant
M. Scully
W. Lange

FT-66-1
Analysis of VSTOL Aircraft Configurations for Short Haul Air Transportation Systems

R. Gallant
M. Scully
W. Lange

This report was performed under Contract C-136-66 for the Department of Commerce, Office of High Speed Ground Transport.
CONTENTS

Page

1   I. Introduction

5   II. Discussion of Results

8   III. Aircraft Design Procedures

8   III-1. General

8   a) Atmospheric Model

9   b) Turbo-Shaft Engines

10  c) Cruise Turbofan Engines

12  d) Direct Operating Cost Program

16  III-2. Tilt Wing Jet Lift and STOL Aircraft

16  a) Tilt wing, Jet Lift and STOL Aircraft Computer Design Programs

45  b) Component Weight Formulas

50  III-3. Rotary Wing Aircraft

50  a) Rotary Wing Aircraft Design Studies

55  b) Computer Design Procedures

66  c) Performance - Fuel Burn

74  d) Component Weights

81  IV. Discussion of Aircraft Designs

81  a) Optimization of Tilt Wing, Jet Lift, and STOL Aircraft

82  b) Off Design Operation

85  c) Fuselage Parametric Analysis

86  d) Tilt Wing Aircraft Transition

98  e) Design Variations of the Tilt Wing Aircraft

108 f) Tilt Wing STOL

110  g) Multiple Hops Operations

112  h) Rotary Wing Aircraft

114  V. References

115  VI. Figures
I. INTRODUCTION

The potential of air transportation as a means of filling the growing need for a mass short haul trans-
portation system was investigated in Ref. 1 where all aspects of short haul air transportation systems were examined in some detail. It was concluded that air transportation could provide a promising means of relieving the congestion associated with the heavy vehicular ground traffic encountered on our urban access routes and at a cost which could well be com-
petitive in the 1970 period with surface transportation systems. This conclusion was postulated on the basis of existing developments in the aircraft industry not yet put into practice on operating airlines but whose feasibility has been well demonstrated with experimen-
tal units.

Among the many aspects of the total system which must be examined in arriving at such a conclusion are the flight vehicle characteristics. The direct operating costs (DOC) of these vehicles was chosen as a measure of their effectiveness. In short haul operations the direct costs are frequently less important than indirect
costs in determining total transportation costs and hence ticket price. However the DOC is a convenient measure for estimating the relative performance of different vehicle configurations and of the penalties associated with operation off optimum conditions. Furthermore, the previous study (Ref. 1) had quantified the almost obvious need in short haul transportation for a vehicle capable of operating from highly congested areas and requiring a minimum in take-off, landing or cleared approach areas. This need directed attention to the newer concepts of vertical take-off and landing (VTOL) aircraft which would not have the speed limitation of present day helicopters, the only VTOL aircraft currently in commercial operations. Because present day experience with these aircraft indicates their direct operating costs to be several times that of comparable fixed wing aircraft, there has been a natural reluctance to predict future operating costs for these vehicles at a level which would make them effective other than in a high-priced specialized operation such as an airline feeder system.

Consequently, in Ref. (1), a study of the costs and operating procedures of the existing helicopter
airlines was conducted in some depth and by this means the predicted direct operating costs were removed from the realm of discussion and opinion and reduced to a matter of statistics and analysis. Maintenance costs and lost time in air and ground maneuvering were, as expected, important aspects of the cost problem and these were therefore analyzed on a quantitative basis. While there may still be room for disagreement on the predicted DOC for the various vehicles considered in this study, the quantitative information on which these are based have been carefully documented in Ref. (1) and are further substantiated in Ref. (2) for the maintenance aspects and in this report for the vehicle characteristics such as weight, fuel burned and block speeds. This additional documentation has been considered desirable not only to confirm the previous results and to explore other promising configurations, but also to provide a basis for rational discussion of the relative merits and potentials of different vehicle configurations which all too often in the past has been conducted on a subjective rather than an objective basis.
It may be well to reiterate the conclusion of Ref. (1) that, in face of the high indirect costs inherent in short haul systems, the actual vehicle configuration is not a dominant factor in determining total operating costs. Any well engineered configuration capable of safe all weather operations would probably prove satisfactory. However the need for direct access to city centers with a minimum of land taking does indicate the desirability of VTOL. These vehicles have the potential for appreciably reducing block times, and hence costs, in the shorter legs, below 50 miles, of interest in intra urban or suburban travel, providing present concepts of control and navigation currently under intensive development for military applications can be reduced to practice in the more legalistic environment of commercial operations.
II. DISCUSSION OF RESULTS

This report has investigated the effects of further refinements to the computer programs as used for determining the vehicle characteristics and hence DOC, and also includes studies in greater depth than was possible in Ref. (1) of the stopped and stowed rotor configuration and of a new concept of a tilt wing configuration where both the wing and rotor/propeller can be individually tilted. This permits lower disc loading hence reduced weights and lower costs, but in turn introduces problems associated with the accommodation of large rotors on a wing of optimum area while still maintaining sufficient stiffness to prevent aeroelastic instabilities. The net result is however a tilt wing of appreciably better performance than that predicted in Ref. (1).

Studies of the stowed rotor continue to show this aircraft to be a competitive configuration with DOCs of the same order as those of the helicopter at ranges above 50 miles. Unknown problems associated with stopping, retracting and stowing the rotor continue to make the prediction of the actual DOC of these
vehicles difficult. The sensitivity of the results to these assumptions is discussed in the body of the report.

An investigation of the effects of multiple hops, typical of a line haul type operation as compared to a shuttle type operation over a given stage length indicated that the average DOC was not appreciably affected by the multiple stop operation when compared with an aircraft operating over the same shorter distance.

Studies of the effects of operating the aircraft off optimum speed and altitude similarly showed relatively little effect on the DOC indicating the high degree of operational flexibility inherent in such an air transportation system. Operation at different altitudes in order to reduce turbulence or minimize noise as weather conditions, passenger comfort and community requirements may dictate is therefore entirely feasible.

An investigation was conducted to determine the capabilities of a tilt wing operating in an STOL mode and, as expected, this aircraft is a superior STOL aircraft although the advantage at the shorter range remains with the tilt wing operated as a VTOL aircraft.
Finally, a study of the effects of increasing fuselage width to increase the number of seats abreast while maintaining constant passenger load showed no appreciable effect on DOC. This study was conducted in an attempt to optimize fuselage weight and aerodynamic drag and indicates the relative insensitivity of DOC to the aircraft fuselage dimensions, thus permitting the use of a configuration with maximum passenger appeal and which would minimize time for loading or unloading.

In summary, the results of these studies substantiate the cost estimate prepared in Ref. (1) as being somewhat conservative. Continued refinement of the computer programs and more complete optimization studies have lowered the estimated direct operating costs by small amounts. The general conclusions of Ref. (1) remain substantially unchanged.
III. AIRCRAFT DESIGN PROCEDURES

III-1 General

a) Atmospheric Model

The ICAO standard atmosphere as presented in NACA TR 1235 is used. For hot days the temperature is increased by the same amount at all altitudes.

The formulas used are as follows:

\[ T = T_{SL} - 0.003566h \]  \hspace{1cm} (°R)

\[ p = \frac{1.233}{T_{SL}} \left( \frac{T}{T_{SL}} \right)^{4.2561} \text{(slug/ft}^3) \]

\[ a = 1117. \sqrt{\frac{T}{519.}} \hspace{1cm} (\text{ft/sec}) \]

where: \( T = \) temperature (°R)

\( T_{SL} = \) sea level temperature

\( T_{SL,STD} = 519^\circ \text{ R std. day} \)

\( T_{SL,HOT} = 550^\circ \text{ R hot day} \)

\( h = \) altitude (ft)

\( a = \) speed of sound (ft/sec)

\( p = \) pressure (slug/ft\(^3\))
b) **Turbo-Shaft Engines**

The power to weight ratio of the turbo-shaft engines is taken as 7.5 hp/lb at a 30-minute maximum continuous power rating and sea level, standard conditions. The installation factor is taken as 1.5, exclusive of fuel system. The 30-minute rating is assumed to be 1.2 times normal rated power at all times. The variation of normal rated power (NRP) with temperature and altitude is approximated by:

\[
\text{NRP} = K_c \times (\text{NRP})_{SL, STD} \left(1 - \frac{0.55h}{30000} - \frac{0.15 (T_{SL-520})}{30}\right)
\]

where \( K_c = 30000 \). 

The specific fuel consumption (\(\text{SFC}_{\text{NRP}}\)) is taken as 0.55 lb fuel/hp-hour at normal rated power and sea level 90\(^\circ\) F conditions. No reduction in specific fuel consumption (SFC) with altitude and speed is assumed since an investigation of various advanced free turbine engine concepts indicated that the increase in SFC associated with the reduction in turbine rpm required to obtain reasonable rotor and propeller efficiencies in forward flight approximately cancels the effects of speed and altitude. The variation of SFC
with power is approximated by:

\[
SFC = SFC_{NRP} \left( \frac{NRP}{\text{Power used}} \right)^{0.36}_{\text{SL, STD}}
\]

c) **Cruise Turbofan Engines**

Cruise turbofan engines with a by-pass ratio of 2 (bypass mass flow equal to twice the gas generator mass flow) are taken to have a thrust to weight ratio of 10 lb thrust/lb. of weight exclusive of thrust deflectors, at 30-minute maximum continuous rating and sea level standard conditions. The installation factor is taken as 2 including thrust deflectors. The 30-minute rating is taken at 1.2 times normal rated thrust at all times. The variation of normal rated thrust \( (NRT_{T}) \) with temperature and altitude is approximated by:

\[
NRT_{T} = K_{TC} (NRT)_{SL, STD}
\]

where

\[
K_{TC} = \left[ 1 - \frac{0.55h}{30000} - \frac{0.08(TSL-520)}{30} \right]
\]

The thrust specific fuel consumption is taken as .55 lb fuel/(lb thrust-hr) at normal rated thrust and sea level - static conditions. The variation of TSFC
with speed and altitude is approximated by:

\[ TSFC = TSFC_{SL,ST} + .45M - \frac{.05h}{30000} \]

where \( M \) = aircraft Mach number, \( h \) = altitude (ft), and \( ST \) implies \( M=0 \) conditions.

The jet lift engines are assumed to have a bypass ratio of 2 with a thrust to weight ratio of 25 at 30 minute maximum continuous rating and sea level conditions. The installation factor is taken as 2.

The variation of normal rated thrust with temperature and altitude is the same as the cruise turbofan engines. The static sea level specific fuel consumption is taken as 0.7. The variation of specific fuel consumption of the lift engines with speed and altitude is taken as

\[ TSFC = TSFC_{SL,ST} + \frac{0.12V}{200} - \frac{0.1h}{30000} \]

where \( V \) = aircraft velocity, mph
d) Direct Operating Cost Program

The direct operating cost program calculates the DOC in dollars per aircraft mile for an airborne vehicle as estimated by the methods described in Ref. (1) and further developed in Ref. (2).

It was found early in the analysis of the operating costs of these vehicles that maintenance costs would represent a major portion of the DOC and it was further established that these costs were highly dependent on the state of development of the aircraft and its components. Consequently an analysis was made of the subsystem maintenance costs for both existing fixed wing and rotary wing aircraft, as reported in Ref. (1). From this, estimates were made for the 1970 technology VTOL aircraft maintenance costs. These estimates were substantiated by a detailed analysis of the cost and causes of all maintenance items over a one year period for two existing short haul airlines operating rotary wing aircraft as reported in Ref. (2).

The vehicles are powered by gas turbine engines, all of which are normally operating continuously (ex-
cept for the lift engines of the jet lift aircraft).

The total direct operating costs are broken down into the following costs:

**Total Flight Operations**
- Pilot
- Co-pilot
- Fuel and Oil
- Insurance

**Total Direct Maintenance**
- Airframe Maintenance
  - Labor
  - Materials
- Engine Maintenance
  - Labor
  - Engines

**Applied Maintenance Burden**

**Total Depreciation**
- Airframe
- Engines
- Airframe Spares
- Engine Spares
- Propellers and/or rotors
- Electronic Equipment
Assumptions of the input coefficients were as follows:

a) Operating crew consists of one pilot and one copilot, paid at US domestic rates.

b) Fuel is JP4 at 0.11 dollars per gallon and 6.5 pounds per gallon.

c) Oil is 6 dollars per gallon and 8.1 pounds per gallon.

d) Insurance rate is 4% of the aircraft cost per year.

e) Public liability and property damage rate is .0087 $/aircraft mile.

f) Labor rate is $3 per hour.

g) Depreciation is to a residual value of 15% for airframe, engine propellers, and airframe and engine spares.

h) Electronic equipment valued at $150,000 is fully depreciated over five years.

i) Airframe spares are assessed at 10% of the empty airframe cost.

j) Engine spares are assessed at 50% of the total engine cost with a spare parts price factor of 1.5.

k) The cost of the airframe is related to the production run through an 85% learning curve, with an initial cost of $150 per pound and a development cost of $5000 per pound.
Normal values of input parameters for DOC cost studies were:

1) Distances: 10, 20, 30, 40, 50, 60, 80, 100, 200, 400 miles

2) Cost of engines: turbo shaft 300 $/lb
turbo jet 150 $/lb

3) Engine time between overhauls = 4000 hrs.

4) Airframe depreciation period = 12 years

5) Vehicle utilization = 3000 hours per year

6) Production run of airframe = 300

7) Maintenance cost factor: helicopter = 1.3
   (See Refs. 1 and 2)
   tilt wing = 1.3
   STOL = 1.1

The DOC program accepts as input from the design programs the fuel, burn, and average block speed for the set of distances less than design range. Any two parameters can be selected as variable in one run. For example, DOC versus distance can be given for utilizations of 1000, 2000, 3000, etc. hours per year. Typical breakdowns of the DOC for all aircraft are given in Figures F4a, F5a, F6a and Figures R20, R21, and R22.
The design procedure for the conventional tilt wing, jet lift and STOL aircraft is discussed in this section. Since the design techniques are all basically the same, the three vehicles will be discussed simultaneously. Differences in techniques will be appropriately noted. The design information presented here supplements that given in Ref. (1).

The aircraft design is an iterative procedure. For a given payload and design range, etc., the general method is to

i) estimate a gross weight

ii) design aircraft components based upon the assumed gross weight and specified operational requirements.

iii) determine the fuel used over the design range

iv) calculate a new gross weight based upon the fuel used and component weights.

v) repeat the procedure until changes in the assumed and calculated gross weights are no more than 10 pounds.
The design of all aircraft was based upon the requirement that the aircraft was capable of taking off on a 90°F hot day at sea level. These design requirements are considered to be conservative designs for the Northeast Corridor environment. The sequential steps in the computer program are now given.

1) Aircraft Sizing

The fuselage length was based upon the number of passengers, the specified number of seats abreast, the number of main entrance doors, the number of toilets, and a fixed length which allowed for the cockpit, nose section and tapered tail section.

The relationship used was

\[
\text{fuselage length} = \frac{\text{no. of passengers}}{\text{no. of seats abreast}} \times 2.833 + \text{no. of doors} \times 3.7 + \text{no. of toilets} \times 4.5 + 27.5 \text{ (ft)}
\]

The length of fuselage allotted to the doors allows for clearance area. The fuselage area opposite each main door is assumed to be clear for the purpose of an emergency door and sufficiently allows for the satisfaction of FAA emergency door requirements. The length
of fuselage allotted for the toilet is considered conservative since the area opposite the toilet is not utilized. It could be used for a coat rack and package storage area. Generally, for the 80 passenger aircraft, 6 seats abreast, 2 entrance doors and one toilet were assumed.

For the jet lift aircraft additional fuselage length is required for the installation of lift engines in two engine bays located in forward and aft sections of the fuselage. It was assumed that the lift engines had an effective loading of 1000 lbs. per square foot over the horizontal cross-section of this engine bay. It was also assumed that 80% of the breadth of fuselage (inside diameter) was available for lift engine installations. This would permit the installation of two engines side by side with an aisle for walking from the cockpit to the passenger compartment. Therefore the additional length of fuselage necessary for the installation of lift engines is

\[
\text{fuselage} = \frac{\text{lift engine thrust}}{1000 \times 0.8 \times (\text{breadth of fuselage})}
\]

The breadth of fuselage was based upon the number of seats abreast and a fixed value which allowed for one
aisle and seat clearances at the fuselage walls as
follows.

\[
\text{Breadth of fuselage} = 1.6 \times (\text{no. of seats abreast}) + 2.3, \text{ feet}
\]

The fuselage outside diameter is considered to be eight inches greater than the inside diameter. All aircraft were designed with a maximum of six seats abreast.

On the first pass of the design loop the wing area is determined from a guess of gross weight and wing loading. Subsequent wing areas are then determined by the most recent values of gross weight and wing loading.

2) Drag Calculations

The profile drag of the aircraft, \( C_{D_0} \), is then determined. The drag coefficient of each major aircraft component was determined independently and then summed to obtain the total profile drag coefficient. The drag accumulation takes into account interference drag, roughness and leakage. The Reynolds number used was based on cruise altitude and speed.

The drag models used are as follows:
(i) **Fuselage Drag**

\[
\frac{C_{D\text{ wet}}}{C_f} = 1 + \frac{1.5}{(1/d)^{3/2}} + \frac{7}{(1/d)^3}
\]

where \( l \) = fuselage length

\( d \) = fuselage diameter

\( C_{D\text{ wet}} \) = drag coefficient based on wetted area

and \( C_f \) = friction drag coefficient based on flat plate theory

The friction drag coefficient is taken as:

\[
C_f = \frac{0.030}{(R_{n\text{ fus}})^{1/7}}
\]

where \( R_n \) is the Reynolds number at cruise flight conditions

\[
R_{n\text{ fus}} = (1) \cdot (R_n/\text{foot})
\]

\( S_{\text{wet}} \approx \pi (l) \cdot (d) \) = wetted fuselage area

\( S_0 = \frac{\pi (d)^2}{4} \) = frontal area of fuselage

\[
\therefore \frac{C_{D_0}}{C_f} = \frac{C_{D\text{ wet}}}{C_f} \cdot \left( \frac{S_{\text{wet}}}{S_0} \right) = \frac{C_{D\text{ wet}}}{C_f} \cdot (4l/d)
\]
where $C_{Do}$ = total fuselage drag coefficient based on frontal area.

These are reduced to:

$$C_{Do} = C_f \left[ 4 \left( \frac{1}{d} \right) + \frac{6}{\left( \frac{1}{d} \right)^{1/2}} + \frac{28}{\left( \frac{1}{d} \right)^2} \right]$$

based on frontal area.

(ii) **Wing Drag**

$$C_{Do} = 2C_f \left[ 1 + 2 \cdot \frac{t}{c} + 60 \cdot \left( \frac{t}{c} \right)^4 \right]$$

where:

$C_{Do}$ is based on projected wing area, $S_w$.

$C_f$ is based on wetted areas of the wing

$t/c = \text{thickness/chord ratio.}$

If we take $t/c = 0.12$

$$C_{Do} = 2.48 \ C_{f\text{wing}}$$

$$C_{f\text{wing}} = \frac{.030}{(R_{n\text{wing}})^{1/7}}$$

$$R_{n\text{wing}} = \bar{C} \cdot (R_{n/foot})$$
where $C$ is the mean chord length.

These are reduced to:

$$C_{D_{0\text{wing}}} = \frac{0.0744}{(R_{n\text{ wing}})^{1/7}} \text{ based on } S_w$$

(iii) **Empennage** - (includes interference drag)

$$C_{D_{0\text{emp}}} = 0.6 \ (C_{D_{0\text{wing}}}) \text{ based on } S_w$$

(iv) **Nacelles** - (for 4 nacelles)

$$C_{D_{0\text{nac}}} = 1.1 \ C_{D_{0\text{fuse}}} \left(\frac{S_o}{S_w}\right) + C_{D_{0\text{wing}}} + C_{D_{0\text{emp}}} + C_{D_{0\text{nac}}} \text{ based on } S_w$$

(v) **$C_{D_{0\text{Total}}}$**

$$C_{D_{0}} = 1.1 \ C_{D_{0\text{fuse}}} \left(\frac{S_o}{S_w}\right) + C_{D_{0\text{wing}}} + C_{D_{0\text{emp}}} + C_{D_{0\text{nac}}} \text{ based on } S_w$$

The constant of 1.1 takes into account leakage and roughness of the aircraft.

The $C_{D_{0}}$ for several current commercial aircraft was determined using this model to verify its validity.
and were found to compare quite well.

3) Wing Area and L/D for Cruise

Using the profile drag coefficient, the coefficient of lift, $C_L$, which would result in maximum lift to drag ratio, $L/D$, for the tilt wing and jet lift aircraft in cruise was determined by:

$$C_L = \left( C_{D0} \cdot \pi \cdot e \cdot AR \right)^{1/2}$$

where we have specified as input parameters:

- $e$ = efficiency factor of wing
- $AR$ = wing aspect ratio

The maximum $L/D$ was then determined by

$$L/D_{\text{max}} = \frac{C_L}{2 \cdot C_{D0}}$$

Determining the $C_L$ which results in the maximum $L/D$ will yield the optimum use of the wing at a given speed. However, designing the wing solely on this criteria may result in too large a $C_L$, resulting in the possibility of inadvertent wing stall due to vertical gusts. Therefore, whenever the calculated value
of $C_L$ is greater than 0.5, the $C_L$ is set at 0.5 and
the corresponding $L/D$ is determined by

$$L/D = \frac{C_L}{C_{D_0} + C_{D_i}} = \frac{C_L}{C_D}$$

where $C_{D_i} = \text{induced drag} = \frac{C_L^2}{\rho e \ AR}$

For the STOL aircraft the coefficient of lift
was determined by

$$C_L = \frac{WL}{1/2 \ \rho \ V_{cr}^2}$$

where

- $WL = \text{wing loading is fixed by takeoff considerations}$
- $\rho$ is the density at cruise altitude
- $V_{cr}$ is cruise velocity.

Whenever $C_L$ is greater than 0.5 it is reset to 0.5.
Since the wing loading is fixed, the cruise velocity
is then increased to the value which will satisfy the
$C_L$ equation above. The lift to drag ratio is then com-
puted by

$$L/D = \frac{C_L}{C_{D_0} + C_{D_i}}$$
4) Thrust Requirements

The cruise drag equals cruise thrust for the STOL, therefore,

\[ \text{Thrust}_{cr} = \frac{1}{2} \rho \frac{V_{cr}^2}{C_D S_w} \text{ where } C_D = C_{D0} + C_{Di} \]

An improved value of wing area and wing loading for the tilt wing and jet lift aircraft is now computed by

\[ S_w = \frac{\text{Gross Weight}}{1/2 \rho V_{cr}^2 C_L} \]

\[ WL = \frac{\text{Gross Weight}}{\text{Wing Area}} = \frac{W_g}{S_w} \]

For the jet lift aircraft, the cruise thrust requirement is simply determined by the cruise drag.

\[ \text{Thrust}_{cr} = \frac{1}{2} \rho \frac{V_{cr}^2}{C_D S_w} \]

For the tilt wing aircraft, a wing loading to disc loading ratio (DL/WL) of 0.6 is an important design requirement. Its function is to insure a high velocity propeller slipstream over the wing which improves the stall characteristics of the wing during accelerating and decelerating transitions. Therefore, with this
restriction the disc loading becomes:

\[ \text{disc loading} = (\text{DL/WL}) \times \text{(Wing Loading)} \]

The propeller diameters are then determined by

\[ D = 2 \left[ \frac{W_g}{\text{NE} \times \text{DL} \times \Omega} \right]^{1/2} \]

where

\[ \begin{align*}
\text{NE} & = \text{Number of Engines} \\
\text{DL} & = \text{Disc loading, lbs/ft}^2
\end{align*} \]

The rotational speed of the propellers (\( \Omega \)) is determined by the propeller diameters and the requirement that the propeller tip Mach number equal 0.75 on a 90\(^\circ\) F sea level day. This Mach number was chosen for noise consideration. It was felt that rotor noise at higher Mach numbers would not be tolerable at city center locations. The propeller thrust coefficient at hover is then determined by

\[ C_{T_{\text{hover}}} = \frac{W_g}{(\text{NE}) \rho \Omega^2 (D/2)^4} \]

The propeller solidity (\( \sigma \)) is

\[ \frac{C_{T_{\text{hover}}}}{(C_T/\sigma)} = \frac{\text{propeller area}}{\text{disc area}} \]
where

\[ \frac{C_T}{\sigma} = 0.12, \text{ fixed input} \]

The value of $\sigma$ is constrained to a maximum value of .25 since that is considered to be the upper limit in the design of a practical propeller. Whenever the calculated value of $\sigma$ violates this constraint, it is reset to .25, and the propeller disc loading, diameter, thrust coefficient, and rotational speed must be recalculated. Consequently, a new wing loading, wing area, aircraft profile drag coefficient ($C_{D_0}$), lift coefficient, and cruise L/D must also be recalculated since the wing design is coupled to the propeller design through the $(DL/WL)$ ratio.

5) Power Requirements

The horsepower required for the tilt wing is determined by hover requirements:

\[
HP_{\text{hover}} = \frac{\text{Gross Weight}}{550 \ C_{T_{\text{hover}}}} \cdot V_{\text{tip}} \cdot \left[ \frac{C_{D_{\text{prop}}}}{\sqrt{2}} + \frac{C_{T_{\text{hover}}}}{\sqrt{2}} \right]^{3/2}
\]
The engine shaft horsepower required is

\[ \text{SHP}_{\text{hover}} = \frac{\text{HP}_{\text{hover}}}{\eta_{\text{hover}} \cdot \eta_{\text{trans}}} \]

where

\[ \eta_{\text{hover}} = .9 = \text{propeller efficiency in hover} \]

\[ \eta_{\text{trans}} = .9 = \text{transmission efficiency} \]

The maximum shaft horsepower available (SHP\(_0\)) is determined by the requirement that the engines have 15 percent extra power available to be used as control power. An additional 33 1/3% power is also made available for engine-out emergency operation. The tilt wing is assumed to have four engines. The maximum shaft horsepower is considered to be 120 percent of normal rated power (NRP). Therefore, in summary, the tilt wing normal rated power is

\[ \text{NRP} = \frac{1.533 \text{ HP}_{\text{hover}}}{1.2\eta_{\text{hover}} \cdot \eta_{\text{trans}}} \]

The installed power of the STOL aircraft is
determined by cruise requirements. Using the thrust needed in cruise, the horsepower used is determined by

\[ \text{HP}_{cr} = \frac{(\text{Thrust}_{cr}) \cdot V_{cr}}{550 \eta_{prop} \cdot \eta_{trans}} \]

where

\[ \eta_{prop} = 0.875 \]

\[ \eta_{trans} = 0.90 \]

The equivalent sea level power is then determined by the correction factor presented in Section IIIb. Also, the engine is assumed to operate at 90 percent of NRP in cruise. Therefore, the NRP is 1.11 times the equivalent sea level cruise power. No extra horsepower is considered necessary for engine-out capability.

The propellers were sized by assuming that the ratio of horsepower to propeller diameter squared was 7.5. A propeller solidity of .165 was assumed. These relationships were established as a result of propeller optimization studies as being reasonably representative. Their use considerably simplified the computational process. Corresponding efficiencies were centered
around 87½ percent, which was used in all computations.

The jet lift aircraft is designed with separate cruise and lift engines. The normal rated thrust (at sea level (NRT) of the cruise engines is determined by the thrust required in cruise corrected for altitude effects plus the assumption that the engines operate at 90 percent of their NRT in cruise.

The lift engine installed thrust is based on the requirement that the total available thrust (lift plus cruise engines) is equal to 1.5 times the gross weight at sea level, 90° F. This thrust to weight ratio of 1.5 is required for engine-out hover capability with a second engine shutdown in order to maintain thrust symmetry. It also leaves sufficient excess thrust to insure control capability amounting to an acceleration of approximately ½ radian per second squared in pitch, and a margin for deceleration of the aircraft.

6) Fuel Requirements

The next phase of the design process was to determine fuel requirements. This was accomplished by flying the aircraft over the design range. The fuel consumed
during each segment of the flight profile was based on the power or thrust used and the specific fuel consumption (SFC). The SFC was determined by the relationship presented in Sections IIIb and IIIc.

The fuel consumed by the tilt wing aircraft will be considered first. The fuel consumed during engine start and checkout was assumed equivalent to normal rated power for 0.6 minutes. The aircraft had a total hover time of 12 seconds consisting of the time to ascend vertically to 50 feet from take-off, and the reverse when landing. The fuel consumed during the hover mode was based on the engine shaft horsepower used and the associated SFC.

\[
\text{Fuel (hover)} = (\text{SFC}_{\text{hover}}) \times (\text{HP}_{\text{hover}}) \times t_h
\]

where

\[ t_h = \text{time in hover} \]

For simplicity, and with no loss in overall accuracy, the acceleration and transition to the climb velocity at 1500 feet altitude and the acceleration from the termination of climb to cruise velocity were handled as one total acceleration period.
The total time for this acceleration was computed to be typically 0.7 minutes. The fuel consumption was based on the conservative assumption that the engines were operating at normal rated power.

\[
\text{Fuel (acceleration)} = (\text{SFC}_{\text{NRP}}) (\text{HP}_{\text{NRP}}) \times t_{\text{acc}}
\]

where \( t_{\text{acc}} \) = acceleration and transition time = 0.7 minutes.

The climb phase was performed at maximum rate of climb at NRP. The horizontal velocity which results in the maximum rate of climb for a given altitude is determined by

\[
V_{C1} = \left[ \frac{2Wg}{\rho S_w \sqrt{3\pi e AR C_{D_0}}} \right]^{1/2}
\]

The average forward velocity for maximum rate of climb was determined by averaging \( V_{C1} \) at 1500 feet altitude and at cruise altitude. The average rate of climb \( (R/C) \) is determined by

\[
R/C = \frac{33,000 \ \text{HP}_{\text{Cl}} \eta_{\text{prop}} \cdot \eta_{\text{trans}} - 60 \cdot V_{C1} \cdot \text{Drag}}{\text{Gross Weight}} \ (\text{ft/min})
\]

where
\( HP_{Cl} \) = average shaft horsepower available, i.e., NRP corrected for average altitude

\( V_{Cl} \) = average horizontal velocity during climb

Drag = average drag

Since NRP is used during climb the specific fuel consumption \( (SFC_{Cl}) \) is taken to equal \( SFC_{NRP} \). The fuel consumed during climb was

\[
\text{Fuel (Climb)} = (SFC_{NRP}) (HP_{Cl}) (t_{Cl})
\]

where

\( t_{Cl} = \text{time to climb} = \frac{\text{cruise altitude} - 1500}{R/C} \) (min.)

The thrust required for the tilt wing aircraft for the cruise phase of the profile is equal to the drag. The cruise shaft horsepower is then calculated by

\[
\text{SHP}_{cr} = \frac{\text{Thrust}_{cr} \ V_{cr}}{550 \ \eta_{\text{prop}} \ \eta_{\text{trans}}}
\]

where

\( \eta_{\text{trans}} \times \eta_{\text{prop}} = \text{overall efficiency} \)

\( = .81 \) for \( V_{cr} \leq 400 \) mph

\( = .72 \) for \( V_{cr} = 450 \) mph
The sea level equivalent horsepower, specific fuel consumption and the fuel consumed per minute are determined as in the climb phase. The total fuel consumed during cruise is determined from the fuel consumed per minute, and the cruise velocity and distance. The cruise distance is determined by subtracting the distances covered in the acceleration, climb, descent and deceleration phases from the design range (see Fig. F1a). Therefore the fuel consumed in cruise is

\[ \text{Fuel (cruise)} = (\text{SFC})_{cr} \times (\text{HP}_{cr}) \times (t_{cr}) \]

where

\[ t_{cr} = \frac{\text{cruise distance}}{V_{cr}} \]

The descent phase of the tilt wing aircraft profile is performed at the cruise velocity. The rate of descent (R/D) is approximated by

\[ \frac{V_{cr}}{(L/D)_{avg}} \]
where

\[
L/D_{avg} = \text{the average of the lift to drag ratio at cruise and 1500 feet altitude.}
\]

It is assumed that no thrust is available from the engines. However, a fuel consumption equivalent to the engine operating at 25% of NRP and SFC equal to SFC\text{NRP} is assumed, since the engine is in flight idle.

Fuel (descent) = (SFC\text{NRP}) \times (\text{NRP}/4) \times (t_d)

where

\[
t_d = \text{time of descent} = \frac{\text{cruise altitude} - 1500}{R/D}
\]

The deceleration and re-transition phase of the tilt wing flight profile is assumed to require an average power level equal to 0.75 of the horsepower required in hover. The weight of fuel during deceleration is

Fuel (deceleration) = (SFC\text{dec}) \times (\text{HP}_{\text{dec}}) \times (t_{\text{dec}})

where

\[
t_{\text{dec}} = \text{time of deceleration} = 1 \text{ minute}
\]

\[
\text{HP}_{\text{dec}} = \text{average deceleration power}
\]

\[
\text{SFC}_{\text{dec}} = \text{specific fuel consumption in deceleration based on } \text{HP}_{\text{dec}}.
\]
Reserve fuel is based on 20 minutes of reserve time at cruise power and specific fuel consumption.

At this point, the weight of various aircraft components such as engines, wings, propellers, fuel tanks, etc. can be re-estimated using the weight formulae of Section IIIf. The fuel weight, payload, and this new estimate of aircraft empty weight combine to give a new gross weight, and the computer design process will return to step (1) if the new gross weight differs by more than 10 lbs. In this way, an aircraft will be designed for specified inputs of payload-range, aspect ratio, seats abreast, cruise speed, cruise altitude, etc. which will have properly sized engines, propellers, wings, etc. to perform the specified mission. Investigation of the sensitivity to all of the design input parameters (such as cruise speed) can be quickly carried out to choose optimum parameter values. This sensitivity investigation gives an aircraft design optimized for any criterion such as direct operating cost, minimum gross weight, etc. In this study, the computer output of the design program was linked to the DOC program, and aircraft were optimized for minimum DOC.
The jet lift and STOL are designed in much the same way as the tilt wing aircraft. For the jet lift aircraft, the fuel consumed during the start and check-out phase of all engines is based on a thrust specific fuel consumption (TSFC) of 0.7 for 0.6 minutes of normal rated thrust. The fuel consumed in the hover mode was based on a TSFC of 0.7 for 0.2 minutes at a total thrust level of 1.2 of the gross weight. The weight of fuel used in the acceleration period is based on a TSFC of 0.7 for the lift engine and 0.55 for the cruise engines. The cruise engines are considered to operate at normal rated thrust and the lift engine at 0.6 normal rated thrust for the acceleration phase. The acceleration time is the time taken to accelerate the aircraft from hover to cruise velocity at an average acceleration of \( \frac{1}{2} g \). Unlike the tilt wing all the acceleration takes place at 1500 feet altitude and therefore the climb to cruise altitude is at cruise velocity. In the climb phase, the thrust of the lift engines is

\[
\text{thrust of lift engines} = \frac{(W_g)(R/C)}{V_{C1}}
\]
where \( W_g = \) Gross Weight

\[ \text{R/C} = \text{rate of climb} = 5000 \text{ ft/min} \]

\[ V_{Cl} = \text{climb velocity} = \text{cruise velocity} \]

The thrust of the cruise engines is taken to be the average of the normal rated thrust at 1500 feet altitude and at cruise altitude.

The thrust specific fuel consumption of the lift and cruise engines in climb is the average of the TSFC at 1500 ft. and cruise altitude. Hence, the fuel consumed in climb is based on the average TSFC and thrust of the lift and cruise engines and the time to climb. The time to climb is determined from the R/C and cruise altitude minus 1500 feet.

The weight of fuel consumed in cruise is determined by the thrust in cruise of the cruise engines alone, the time in cruise and the TSFC in cruise. The time is calculated as in the tilt wing case and the TSFC in cruise is determined by

\[
\text{TSFC}_{cr} = \text{TSFC} + .45 \, \text{Mn} - \frac{.05 \, h_{cr}}{30,000}
\]

as described in Section IIIc.
where \( \text{TSFC} = .55 \)

\[ \text{TSFC}_{\text{cr}} = \text{cruise TSFC} \]

\[ M_n = \text{cruise Mach No.} \]

\[ h_{\text{cr}} = \text{cruise altitude} \]

The lift engines are shut down during cruise. The rate of descent is determined as in the tilt wing case. The fuel used in descent is based on the time in descent, 25 percent of normal rated thrust and the thrust specific fuel consumption used in cruise.

The fuel consumed in the deceleration conversion is based on the lift engines operating at .6 normal rated thrust. The reserve fuel is based on the operation of the cruise engine at cruise normal thrust for 20 minutes.

An additional amount of fuel is consumed in the restart and checkout of the lift engines prior to the landing of the jet lift aircraft. The restart fuel consumed is based on .5 minute operation at normal rated thrust of the lift engines and at cruise specific fuel consumption.

For the STOL aircraft the fuel consumed during engine start is determined as in the tilt wing case.
However, additional fuel required for taxiing is based on 25 percent normal rated power at a specific fuel consumption of .55 for 6 minutes.

Three acceleration periods are considered in the STOL aircraft operation, 1) ground roll acceleration from zero velocity to lift off velocity, 2) acceleration from lift off velocity to velocity of maximum rate of climb at 1500 ft. and 3) acceleration from maximum rate of climb at cruise altitude to cruise velocity. The acceleration periods are described below.

The velocity at lift off is considered to be 118.5 fps, which assumes a take-off lift coefficient of 3.0 or .9 $C_L$ max at a wing loading of 50 psf. The average thrust during the take-off roll is determined by the approximation

$$\text{Thrust}_{\text{roll}} = \frac{1.2 \ \text{NRP} \ \eta_{\text{prop}} \cdot \eta_{\text{trans}} \cdot 550}{0.7 \ V_{\text{lift}}}$$

where

$\text{NRP}$ = normal rated power

$\eta_{\text{prop}} = 0.875$ = propeller efficiency

$\eta_{\text{trans}} = 0.90$ = transmission efficiency

$V_{\text{lift}} = 118.5$ fps = lift off velocity
The average acceleration is then determined by

\[ \text{Accel}_{\text{roll}} = \left[ \frac{\text{thrust}_{\text{roll}} - \text{drag}_{\text{roll}}}{W_g} - \mu \right] g \]

where

\[ \mu = 0.1 \text{ = ground roll friction based on the requirement for takeoff on a smooth runway but in slush.} \]

\[ W_g = \text{gross weight, lbs.} \]

\[ \text{drag}_{\text{roll}} = \frac{1}{2} \rho_{\text{SL}} \left( \cdot 7 V_{\text{lift}} \right)^2 C_{\text{D}_0} S_w \text{ (lbs)} \]

\[ \rho_{\text{SL}} = \text{sea level air density, slugs/ft}^2 \]

\[ C_{\text{D}_0} = \text{parasite drag coefficient} \]

\[ S_w = \text{wing area, ft}^2 \]

\[ g = \text{gravity} = 32.17 \text{ ft/sec}^2 \]

The time of the roll acceleration is the time it takes to accelerate to \( V_{\text{lift}} \) at the average acceleration \( \text{Accel}_{\text{roll}} \).

The second acceleration phase is that from \( V_{\text{lift}} \) to the velocity at maximum rate of climb at \( 1500 \) \( (V_{\text{C1}_{1500}}) \). The maximum rate of climb is determined as in the tilt wing case. The average thrust is calculated as
Thrust\textsubscript{accel\textsubscript{2}} = \frac{1.2 \text{ NRR} \cdot n_{\text{prop}} \cdot n_{\text{trans}}^{550}}{(\frac{1}{2}) (V_{\text{lift}} + V_{\text{Cl}1500})}

The associated average acceleration is determined by

\text{Accel}_2 = \frac{(\text{Thrust}_{\text{accel\textsubscript{2}}} - \text{drag}_2)}{m}

where

\text{drag}_2 = \frac{1}{2} \rho_{1500} \left[ \frac{V_{\text{lift}} + V_{\text{Cl}1500}}{2} \right]^2 C_D S_w

\rho_{1500} = \text{density at 1500 ft altitude}

S_w = \text{wing area}

C_D = C_{D_0} + C_{D_i} \text{; aircraft drag coefficient evaluated at the average velocity}

The corresponding acceleration time is the time it takes to accelerate from \( V_{\text{lift}} \) to \( V_{\text{Cl}1500} \) at \text{Accel}_2.

The third acceleration phase is that from the velocity at maximum rate of climb at cruise altitude to cruise velocity and is determined in essentially the same way as for the 1500 feet altitude acceleration case. Also the cruise and descent phases were determined as in the tilt wing case. The fuel consumed in the three acceleration phases are handled as a group. It is assumed that the engines operate at normal rated power at a specific fuel consumption.
.55 for the total time of the three acceleration phases. The fuel consumed during climb cruise and descent is calculated the same as for the tilt wing.

It is assumed that the total deceleration from cruise velocity to zero velocity is at an average of 8 feet per second squared at 0.25 normal rated power and a specific fuel consumption (from Section IIIb).

\[ SFC_{dec} = 0.55 \ (NRP/0.25 \ NRP) \]

An additional amount of fuel is also required for four minutes of maneuvering in approach patterns. The power is conservatively assumed to be the same as the cruise power and the corresponding specific fuel consumption. Reserve fuel is based on one-half hour reserve time at cruise conditions.

When a converged solution for any aircraft type is finally achieved, a set of fuel burned and blocks speeds is computed for various ranges less than the design range. These calculations assume that the cruise velocity and fuel flow rate are the same as for the design range case. This information along with the gross weight, empty weight, number of engines, utilization, etc. is then used to calculate direct
operating costs versus range using the DOC program
described in Section IIId.
b) **Component Weights Formulas**

The empirical relations used to determine the components weights of the tilt wing, jet lift, and STOL aircraft are as follows:

**Fuselage**  
\[ W_F = \text{Fus. Wt.} = .8 \ (LF)^{1.5} \ (OD)^{.25} (N \times W_g)^{.15} \]

where
- \( LF \) = length of fuselage (ft)
- \( OD \) = fuselage outside diameter (ft)
- \( N \) = structural load factor = 4.5
- \( W_g \) = gross weight (lbs.)

**Wing**  
\[ W_W = \text{Wing Wt.} = .015 \ N^2 \ W_g^{1.15} W_L^{-0.6} \cdot \left( \frac{1 + \lambda}{t/c} \right)^{0.4} \cdot \frac{1}{\cos \Lambda} \cdot (AR)^{0.5} \]

where
- \( N \) = structural load factor = 4.5
- \( W_g \) = gross weight (lbs)
- \( W_L \) = wing loading - psf
- \( \lambda \) = \( \frac{\text{tip chord}}{\text{root chord}} \)
  - = .5 for tilt wing and STOL
  - = .25 for jet lift
- \( t/c \) = thickness to chord ratio = .1
- \( \Lambda \) = mean sweepback angle
  - = 0 deg. for tilt wing and STOL
  - = 30 deg. for jet lift
AR = aspect ratio
= 9.5 for tilt wing
= 7.0 for STOL
= 6.0 for jet lift

Empennage WTE = Empennage Wt. = .025 Wg, for tilt wing and jet lift
= .035 Wg, for STOL

Engines WE = Engine Wt. = $\frac{1.2 \text{ NRP}}{6.375}$ for the tilt wing and STOL turbo prop engines, lbs.
= $\frac{1.2 \text{ NRTT}}{9.2}$ for jet lift, cruise engines, lbs.
= $\frac{1.2 \text{ NRTT}}{23.}$ for jet lift, lift engines, lbs.

where NRP = normal rated power of the turbo prop engines

NRTT = normal rated thrust of turbofan engines

1.2 NRT = engine 30 minute rating

Propellers WP = Propeller Wt. = (NE). (14.2) .

\[
\left[ \frac{D^2}{10} \left( \frac{P_i}{NE} \right)^{0.5} \frac{V_{tip}}{1000} \frac{T}{D^2 \sigma} \right]^{0.67}
\]
where \( \text{NE} \) = no. of engines

\[ D = \text{propeller diameter} \]

\( \text{Pi} \) = power input to propeller

\[ V_{\text{tip}} = \text{propeller tip velocity, fps} \]

\( \sigma \) = solidity

Nacelles \( W_N \) = Nacelle Wt. = 0.5 \( (W_E) \), for tilt wing

and STOL

= cruise engine and lift engine weight, for jet lift

Engine Oil \( W_O \) = Engine Oil Wt. = 35 \( (W_E) \), for all aircraft

Undercarriage \( W_{LG} \) = Undercarriage Wt. = .03 \( (W_g) \),

for all aircraft

Transmission - \( W_T \)

The tilt wing transmission weight equals the larger

of

\[ W_{T1} = 60 \left[ \frac{1.2 \text{ NRP}}{V_{\text{tipHover}}} \right] \cdot D/2 \]

\[ W_{T2} = 60 \left[ \frac{\text{HP}_{\text{cr}}}{V_{\text{tipCr}}} \right] \cdot D/2 \]

where \( D \) = propeller diameter

\[ V_{\text{tipHover}} \] = Propeller tip velocity in hover, fps

\[ V_{\text{tipCr}} \] = Propeller tip velocity in cruise, fps
The tip velocity in hover is limited to Mach 0.75; in cruise to Mach 0.9. Under most flight conditions $W_{T_1}$ is the larger value.

The STOL transmission weight is

$$W_T = 60 \left[ \frac{1.2 \text{ NRP} \cdot D/2}{V_{\text{tip}}} \right]^{0.8}$$

where $V_{\text{tip}}$ is limited to Mach 0.9 at sea level.

**Furnishings - WFE**

Furnishings and equipment were assumed to weigh 400 pounds plus 50 pounds for each crew member, plus 40 pounds per passenger. The crew consists of pilot, co-pilot and steward at 200 lbs. each.

**Air Conditioning - WAC**

Provision for air conditioning and anti-icing was taken to be 500 pounds plus 13 pounds per passenger.

**Hydraulics - WH**

The weight of the hydraulics was

$$\text{Hydr. Wt.} = 5 \times 10^{-4} (W_g)^{1.28}$$

**Electrical Equipment - WEL**

Electrical Wt. $= 1.61 (W_g)^{0.55}$
Electronic Equipment - $W_{ES}$

Assumed value of 642 lbs.

Flight Controls - $W_{FC}$

Flight Control Weight = $0.02 \ (W_g)$

Fuel Tanks - $W_{FT}$

Fuel Tank Weight = $0.3 \left( \frac{\text{Fuel Wt.}}{6.7} \right)$

Payload - $W_{PL}$

$W_{PL} = 200$ (no. of passengers)

Crew - $W_{CR}$

$W_{CR} = 200 \ (\text{No. of crew}) = 600 \ lbs.$
a) **Rotary Wing Aircraft Design Studies**

A computer program is used to design these aircraft and to compute their DOC vs range performance. This approach allows parametric studies of the effects of aircraft size, design range, engine performance, etc. to be run off easily, as in Reference (1). It also makes direct optimization of parameters such as block speed, cruise altitude, aspect ratio, etc. for minimum DOC possible instead of the more conventional optimizations to minimize gross weight or drag which do not necessarily minimize DOC. Computer results are more easily analyzed and evaluated if an understanding of the program which generated them is available. Therefore a detailed discussion of the rotary winged aircraft design program is presented below. This program designs and computes the DOC vs range performance of helicopters, compound helicopters, and stowed rotor aircraft. The program also prints out weight and drag breakdowns and a table of time, distance covered, and fuel burned in the various phases of flight (acceleration, climb, cruise, etc.). This program is a development of the program used
to calculate the helicopter and compound helicopter performance for Reference (1).

The stowed rotor performance presented in Ref. (1) was calculated by modifying the program which calculated jet lift performance. This gave a first estimate. The decision was made to conduct a more detailed analysis of the stowed rotor since other studies indicate this to be a promising configuration. The current stowed rotor analysis is sufficiently detailed to show the effects of different choices of transition speed, rotor solidity, etc.

Various refinements have been made to the analysis of the helicopter and the compound helicopter since the work reported in Ref. (1). These are summarized below, see pages 54 - 63 for a detailed discussion of the program used for this report.

1) The rotor transmission and drive system is designed to a torque limit which is optimized by a parametric analysis for minimum DOC.

2) The parasite drag calculation has been refined with the fuselage and wing drag calculated individually. The effects of different choices of fuselage length and diameter are then more clearly indicated.
3) The equation for fuselage length has been changed to allow more room for doors and carry-on baggage, and a tradeoff between extra fuselage length (fairing at the tail) and drag has been investigated.

4) A parametric study of the cost of more comfortable seating (larger seat pitch and seat width) has been conducted.

5) The estimate of high speed rotor performance ($C_T/\sigma$ and $L/D_E$ vs $\mu$, Table I, page 56) has been refined. In addition $\sigma$ (the rotor solidity) has been chosen by trying various values of $\sigma$ in the computer calculation and choosing the value of $\sigma$ which results in the lowest DOC. Previously a typical value of $\sigma$ was obtained from other studies.

6) The acceleration distance and fuel burn has been calculated in greater detail, using numerical integration, instead of assuming an average velocity and fuel flow rate.

7) The tip Mach number in hover of the rotor has been limited to obtain improved noise characteristics. This condition sizes the rotor for the stowed rotor aircraft and for compound helicopters.
8) A more detailed calculation of the auxiliary propulsion equipment weight and performance has been conducted for the compound. This included a detailed study of propeller performance in order to choose an optimum propeller solidity ($\sigma_p$). The more detailed study has resulted in a considerable reduction in the auxiliary propulsion system weight estimates and hence an improvement in DOC for the compound.

9) Instead of choosing a wing $L/D$ and wing loading based on typical values from other studies, these are calculated from the aspect ratio ($AR$) and lift coefficient ($C_L$). The AR and $C_L$ are optimized by a parametric analysis for minimum DOC.

10) Instead of unloading the rotor 75% in cruise for the compounds the rotor is sized by a tip Mach number limit in hover and operated at design $C_T/\sigma$ in cruise (see Table I). The wing is then sized to carry the remaining lift.

11) The maximum speed capability of the compound (as limited by aeroelastic effects) has been increased in the light of more recent information.
12) The download on the wings and fuselage during hover is now calculated instead of being estimated. This is necessary for the optimization of $\sigma$ since a high $\sigma$ implies a high disc loading and hence a high down-load.
b) **Computer Design Program**

The design procedure is an iterative one. Initially a gross weight is estimated based on the payload required. This gross weight is used to calculate the fuel burn for the design mission and the component weights. From these results an improved estimate of the gross weight is generated which is used to recalculate the fuel burn and component weights, etc. This process continues until successive estimates of the gross weight agree within 10 lbs.

Since the wing loading is independent of gross weight, it is chosen first.

\[ W_L = \frac{1}{2} \rho V^2 C_L \]

For the stowed rotor aircraft the wing is designed for transition at 5000 ft. and \( V_{TR} = 160 \) mph with \( C_L = 1.5 \). For the effect of varying \( V_{TR} \) and \( C_L \) see Figs. R1 and R2. The wing of the compound is designed for cruise with the \( C_L = 1.0 \). An aspect ratio of 8 is used for both the compound and the stowed rotor. These are practical limits, the DOC decreases slightly for higher values of both \( AR \) and \( C_L \). The DOC of the compound is not very
sensitive to $C_L$ and AR variations.

The rotor is designed next. For advance ratios

$$\mu = \frac{\text{flight speed}}{\text{rotor tip speed}} \geq 0.3$$

the following table of design thrust coefficient/solidity ($C_T/\sigma$) and equivalent lift to drag ratio of the rotor, less hub, ($L/D_E$) is used:

**TABLE I: High Speed Rotor Performance**

<table>
<thead>
<tr>
<th>$\mu$</th>
<th>.30</th>
<th>.35</th>
<th>.40</th>
<th>.45</th>
<th>.50</th>
<th>.55</th>
<th>.575</th>
<th>.60</th>
<th>.65</th>
<th>.68</th>
</tr>
</thead>
<tbody>
<tr>
<td>$C_T/\sigma$</td>
<td>.070</td>
<td>.068</td>
<td>.0655</td>
<td>.058</td>
<td>.054</td>
<td>.052</td>
<td>.051</td>
<td>.0505</td>
<td>.0485</td>
<td>.047</td>
</tr>
<tr>
<td>$L/D_E$</td>
<td>8.5</td>
<td>9.5</td>
<td>9.7</td>
<td>9.5</td>
<td>8.1</td>
<td>7.4</td>
<td>7.2</td>
<td>6.9</td>
<td>6.5</td>
<td>6.3</td>
</tr>
</tbody>
</table>

The rotor tip speed in cruise ($V_T$) is chosen to give a Mach number at the advancing blade tip of .95, $V_T = \frac{.95A}{1 + \mu}$, where $A =$ speed of sound. The rotor blades are 12% thick inboard tapering to 6% at the tip to avoid excessive compressibility losses. For the helicopter the rotor thrust in cruise is taken equal to the gross weight and the rotor is sized based on the design ($C_T/\sigma$) (see Table I), an input $\sigma$, and $V_T$. 

-56-
\[ \Pi R^2 = \frac{W}{\rho c_s \left( \frac{C_T}{\sigma} \right) (V_T)^2} \]

where \( R \) = rotor radius
\( W \) = gross weight

For the compound and the stowed rotor the rotor is sized by a limit on the tip Mach number in hover of \( M = 0.6 \) (see below). The wing of the compound is then sized by requiring the rotor to operate at design (\( C_T/\sigma \)) (see Table I) in cruise.

\[
\text{Wing Lift} = L_w = W - \rho \Pi R^2 (V_T)^2 \sigma \left( \frac{C_T}{\sigma} \right)
\]

The rotor solidity \( (\sigma = \frac{\text{blade area}}{\text{disc area}}) \) is chosen to minimize DOC (See Figs. R3, R4, and R5).

In hover \( C_{TH}/\sigma = 0.10 \) is used and the thrust is taken as 1.1 times gross weight to allow for control requirements. Therefore the hover tip speed = \( V_{TH} = \sqrt{\frac{1.1w}{\rho \Pi R^2 \sigma}} \). If the tip Mach number in hover \( (V_{TH}/A) \) > .6 then the DL = \( W/\Pi R^2 \) is reduced to make \( (V_{TH}/A) = .6 \) in order to satisfy the noise criterion. The power
required in hover is calculated as

\[
\text{HP}_H = \frac{\rho \pi R^2 (V_{TH})^3}{550 \eta_H} \left[ \frac{\sum C_{D_0}}{8} + \frac{C_{TH}^{3/2}}{\sqrt{2}} \right]
\]

where \( C_{D_0} = 0.012 \) and \( \eta_{TH} = 0.86 - D_1/W \) for single rotor aircraft (including allowance for tail rotor power) or \( \eta_{TH} = 0.92 - 0.15 (OL) - (D_1/W) \) for tandem rotor aircraft. \( D_1 \) is the download on the wings and fuselage calculated assuming \( C_D = 0.8 \) for the fuselage, \( C_D = 1.2 \) for the wings, and a fully developed uniform rotor downwash and \( OL = 1.0 - (LF-10.0)/(2R) \) is the overlap. For the engine out hover case the equivalent, installed normal rated power required, corrected to sea level, standard conditions, is:

\[
\text{HP}_{HN} = \left[ \text{HP}_H \cdot (\text{NE}/1-\text{NE}) \cdot \frac{1}{K_C \cdot K_{EP}} \right]
\]

where \( \text{NE} \) = number of engines, \( K_{EP} = 1.2 \), a factor to correct 30 min. rating to normal rated power, and \( K_C \) is a factor to correct power to sea level, standard conditions (see page 9).

The fuselage outside diameter is taken as \( OD = SW \) (SAB) + 3.0, where \( SW = 1.6 \) ft. = seat width and SAB =
number of seats abreast. The fuselage length is taken as
\[ LF = SP \cdot \frac{(PX)}{SAB} + 3.7 \ (DN) + 20 \ (FSL) + 4.5 \]
where \( SP = 2.83 \) (ft) seat pitch and \( PX \) is the number
of passengers (Figure R8 shows the effect of varying
\( SP \) and \( SW \) for a typical aircraft). \( DN \), the number of
large doors per side, is taken as 1 for 40 passengers,
2 for 80, and 3 for 120. \( FSL \), the tail fairing allow-
ance, is 1.0 or 1.5 depending on the degree of tail
fairing in order to allow for a trade-off between fuse-
lage weight and drag. Optimum \( FSL = 1.0 \) for the tandem
rotor aircraft and 1.5 for single rotor aircraft. For
single rotor aircraft it is necessary to ensure that
the fuselage is long enough to accommodate both the main
and tail rotors. Therefore, for single rotor aircraft:

\[ LF \geq 1.12R + 0.5 \left( SP \cdot \frac{(PX)}{SAB} + 3.7 \ (DN) \right) + 12.25 \ (ft) \]

For tandem rotor aircraft: \( LF \geq 1.2R + 10. \ (ft) \)

The parasite drag is built up from the drag of
the various components. The drag of the fuselage is:
(in terms of equivalent flat plate area)

\[ FF = 0.0066 \ (OD)^2 \left[ \frac{LF}{OD} + 1.5 \sqrt{\frac{OD}{LF}} + 7 \left( \frac{OD}{LF} \right)^2 \right] /FSL \]
where FSL is the tail fairing parameter mentioned above.

The wing parasite drag is \( F_W = 0.008 \frac{W}{W_L} \). The drag of the rotor hubs and pylons, the engine nacelles, and the tail rotor is taken as \( F_R = 0.0096 \frac{W}{W_L} \). For the helicopter and the compound the parasite drag flat plate area is taken as:

\[
F = 1.12 \left( F_F + 1.5 \frac{W}{W_L} + F_R \right)
\]

where the 1.12 allows for roughness, protuberances, and leakage and the 50% increase in \( W \) represents the drag of the tail. For the stowed rotor in cruise configuration the parasite drag (flat plate area) is taken as:

\[
F + 1.12 \left( 0.124 F_R + 1.1 \frac{W}{W_L} + 1.5 \frac{W}{W_L} \right)
\]

and when the rotor is operating: \( F = 1.12 \left( F_R + 1.1 \frac{W}{W_L} + 1.5 \frac{W}{W_L} \right) \). The fuselage drag is increased by 10% to account for the drag of the enlarged part of the fuselage required to stow the rotor. The wing skin friction drag is increased by 50% to allow for the drag of the tail. The drag of the engine nacelles and tail is 0.095\( R \). It is possible to provide doors to cover...
the retracted rotor hub and first 1/3 of the blades only thus saving 10% on rotor system weight and paying .029FR in drag as compared to completely covering the blades. For the short ranges considered here (200 mi.) leaving the last 2/3 of the blades uncovered results in a slightly lower DOC.

The overall aircraft lift/drag ratio is calculated as follows:

$$\frac{1}{L/D} = \frac{1}{L/D_i} + \frac{\rho V^2 F}{2W} + \left( \frac{L_R}{W} \left( \frac{1}{L/D_E} - \frac{1}{L/D_i} \right) \right)$$

where $L_R/W =$ rotor lift/gross weight, $L/D_E =$ rotor equivalent $L/D$ (see page 55), and $L/D_i$ is the lift/induced drag of the wing.

$$L/D_i = \frac{1.4 AR \rho V^2}{(WL)}$$

The installed power for the helicopter and compound aircraft is determined by either the cruise requirement or the engine out hover condition, whichever requires more power. The equivalent, installed, sea level, standard day, normal rated power required for engine out hover ($HPHN$) was calculated above (page 58). The power required for cruise is:
\[ \text{HP}_{\text{CR}} = \frac{W \cdot V_{\text{CR}}}{550 \cdot \eta (L/D)_{\text{CR}}} \]

where \( \eta = .96 \) for single rotor helicopters, \( \eta = .92 \) for tandem rotor helicopters, and \( \eta = .85 \) for compounds (determined as typical from propeller performance analyses). The \((L/D)_{\text{CR}}\) used is calculated for cruise values of \( V \) and \( \rho \). To ensure a high engine TBO the engine is designed to operate at a maximum of .9 normal rated power in cruise. The equivalent, installed, sea level, standard day, normal rated power required for cruise is therefore:

\[ \text{HP}_{\text{CR N}} = \frac{\text{HP}_{\text{CR}}}{1/ .9 K_c} \]

where \( K_c \) = factor to correct power to sea level, standard conditions as before. The installed, sea level, standard day, normal rated power (NRP) is the largest of \( \text{HP}_{\text{HN}} \) and \( \text{HP}_{\text{CR N}} \).

The engine used by the stowed rotor aircraft is a convertible turbofan. In this engine the gas generator exhaust can either be used partially to run a fan and partially to produce thrust directly (thus acting as a normal turbofan engine) or the exhaust can be diverted.
to a power turbine where it is expanded to atmospheric pressure producing a maximum of shaft power for use in hover and acceleration to transition speed. Cycle analysis has shown that the ratio of normal rated horsepower to normal rated thrust can be varied at least 20% on either side of 1.0 by varying the design bypass ratio. Since experience has shown that the thrust and power requirements of the stowed rotor are approximately equal, it is assumed that the bypass ratio of the convertible turbofan is chosen to give exactly the required power in each mode of operation. The engine weight is calculated based on 5 hp/lb or 5 lbs. thrust/lb at normal rated power and sea level standard conditions, whichever is greater. An installation factor of 1.5 is used. The SFC of the engine is taken as 0.55 lb/hp-hr at normal rated power and sea level, 90°F conditions. The TSFC is taken as:

\[
\text{TSFC} = 0.55 \frac{\text{NRP}}{\text{TNRP}_{\text{SL,STD}}} \text{ lb fuel/lbs. thrust-hour}
\]

at normal rated power and sea level, standard conditions. The design shaft power is the power required for engine-out hover (HP_{HN}) as calculated above (page 58).
design thrust is determined from the cruise power required:

\[ T_{\text{NRP}_{\text{SL, STD}}} = \frac{W}{(L/D)} \left[ \frac{1}{0.9K_{TC}} \right] \]

where cruise is at 0.9 normal rated power and \( K_{TC} \) corrects the thrust to sea level standard conditions.

Since the rotor transmission and drive system is much heavier than the engines it is not always desirable to design it for the maximum power of the engines (a 30 min. rating of 1.2 normal rated power). While the installed power required is determined by either cruise at 0.9 normal rated power or engine out hover at 1.2 normal rated power, the maximum power actually put through the transmission and drive system occurs during acceleration and climb. Thus there is a trade-off between transmission and drive system weight and block speed. The rotor transmission and drive system is designed for a maximum power (\( HP_{GB} \)) at hover tip speed (\( V_{TH} \)) which is either \( HP_H \) or \( K_{GB} HP_{CR} \) whichever is larger, where \( K_{GB} \) is a parameter which is chosen to minimize DOC. The maximum power which can be used at cruise tip speed (\( V_T \)) is then \( HP_M = HP_{GB} \frac{V_T}{V_{TH}} \) for the heli-
copter (this is because the transmission limit is on torque, not power) and $HP_M = 1.2$ NRP for the other aircraft, since the rotor power is less than $HP_M$. Since $K_{GB} = 1.0$ gives good performance in climb and low speed acceleration increasing $K_{GB}$ to 1.2 for example does not pay off enough in block time to overcome the weight penalty. At the high speed end of the acceleration performance is marginal and $K_{GB} = 1.2$ results in a significant improvement in helicopter block times and hence a small DOC improvement, especially at short ranges. Thus a typical $K_{GB} = 1.2$ for the helicopter and 1.0 for the other aircraft.
c) **Performance - Fuel Burn**

Having determined the basic design of the aircraft it is now necessary to calculate the gross weight as a first step in the iteration. First performance, and hence the fuel required, is calculated. Then the weight empty is calculated.

The fuel burned at hover power is:

\[ F_H = t_H (SFC)_H HP_H \text{ lb.} \]

where \( t_H = 0.2 \) minutes is the hover time, which allows for both takeoff and landing.

The acceleration calculation is broken up into several parts. First is a rotor-born acceleration phase at hover altitude, up to an advance ratio of \( .3 \) or up to transition speed (VTR) for the stowed rotor. The calculation starts at \( \mu = 0 \) and proceeds in steps of \( .05 \) in \( \mu \). The nondimensional inflow through the rotor \( (\lambda) \) is calculated by:

\[ \lambda = \tan i + \frac{C_{TH}}{2 \sqrt{\lambda^2 + \mu^2}} \]
where \( \tan i = \left[ \frac{F \mu^2}{2 \pi R^2} + \frac{\sigma C_{\text{Do}}}{2} \right] / C_{\text{TH}} \)

\( i \) = incidence of the tip path plane

The acceleration is determined from:

\[
\frac{a}{g} = \frac{1}{\mu C_{\text{TH}}} \left[ \frac{550 \eta \text{HP}_G}{\rho \pi R^2 (V_{\text{TH}})^3} - \frac{C_{\text{TH}}^2}{2 \sqrt{\lambda^2 + \mu^2}} - \frac{\sigma C_{\text{Do}}}{8} \right] (1 + 4.6 \mu^2) - \mu \tan(i) C_{\text{TH}}
\]

where \( a = \text{acceleration (ft/sec}^2) \), \( g = 32 \text{ (ft/sec}^2) \)

If \( \frac{a}{g} > \frac{(a)}{g_{\text{max}}} \), then \( \frac{a}{g} = \left( \frac{a}{g} \right)_{\text{max}} = 0.25. \)

An \( \left( \frac{a}{g} \right)_{\text{max}} = 0.5 \) was also tried but was found to make negligible difference in DOC because much of the acceleration is power limited to \( \left( \frac{a}{g} \right) < 0.1. \) So much time is spent in low acceleration that the difference in time between limits of 0.5 and 0.25 \( g \) in the high acceleration part makes a negligible difference in overall acceleration time.

For \( \mu > .3 \), in the case of the helicopter and the compound, the second part of the acceleration calculation
is based on the table of $L/D_E$ vs $\mu$ on page 56 and
again proceeds in steps of .05 in $\mu$.

\[
\frac{a}{g} = \frac{1}{VW} \left[ 550 \, \eta \, HPM - \frac{VW}{L/D} \right].
\]

where $V = \mu (V_T) =$ flight velocity, $L/D$ is recalculated at each $\mu$ and is calculated at cruise altitude,

$HP_M = HPM_{GB} \left( \frac{V_T}{V_TH} \right)$ for the helicopter and $HP_M = 1.2$ NRP (at cruise altitude) for the compound, and $\frac{a}{g} \leq \left( \frac{a}{g} \right)_{max}$ = 0.25 as before.

For $V > V_TR$ in the case of the stowed rotor the acceleration is conducted at hover altitude up to climb speed and then at cruise altitude from climb speed to cruise speed.

\[
\frac{a}{g} = \left[ \frac{1.2 \, K_{TC} \, (T_{NRP})_{SL, STD}}{W} - \frac{1}{(L/D)} \right]
\]

which is calculated at 50 ft/sec intervals in $V$.

The acceleration time ($t_A$), distance ($d_A$), and fuel burn ($F_A$) are now calculated.

\[
t_A = \left( \frac{V}{3600 \, (32.17) \, (a/g)} \right)^{hr}
\]
The fuel burns for the various aircraft are calculated by summing acceleration time multiplied by fuel flow rate for the various parts of the acceleration.

The stowed rotor goes through a transition phase where the rotor is stowed or unstowed. The total transition time (t_{TR}) for both stowing and unstowing is taken as one minute and transition speed (V_{TR}) is taken as 160 mph. Figs. R1 and R6 show the variation of DOC with V_{TR} and t_{TR}. Although the wing is sized for transition at 5000 ft, the thrust required for transition at hover altitude is calculated to ensure that the installed power is large enough to allow transition anywhere between hover altitude and 5000 ft. For convenience of calculation the transition distance (d_{TR}) and fuel burn (F_{TR}) are calculated at hover altitude although noise considerations will force transition at a higher altitude in actual operation.

For the helicopter and the compound, climb is conducted under rotor power at maximum rate of climb. The
rate of climb \((R/C)\) is calculated from energy consider-
ations:

\[
R/C = \frac{V_{TH}}{C_{TH}} \left[ \frac{550 \text{ HP}_{GB}}{\rho \frac{\pi R^2}{2} \mu_c^2} - \frac{(C_{TH})^2}{2} \mu_c - \frac{C_{D_0}}{8} \right] (1 + 4.6 \mu_c^2) - \frac{F \mu_c^3}{\pi R^2}
\]

where \(\mu_c\) is determined by setting \(\frac{d(R/C)}{d \mu_c} = 0\) and solving the resulting quartic for \(\mu_c\). \(V_c\) and \(\mu_c\) are calculated at three different altitudes: hover altitude \(h_H\), cruise altitude \(h_{CR}\), and their average \(\frac{h_H + h_{CR}}{2}\).

The three results are then averaged to obtain an average \(R/C\) and \(\mu_c\) which are used to determine time to climb \((t_c)\), distance in climb \((d_c)\), and fuel burned in climb \((F_c)\).

\[
t_c = \left[ \frac{h_{CR} - h_H}{3600 (R/C)} \right] \text{ hr}, \quad d_c = t_c \mu_c V_{TH} \left[ \frac{3600}{5280} \right] \text{ mi}.
\]

\[
F_c = t_c (SFC) \text{ HP}_{GB} \text{ lb}.
\]

For the stowed rotor aircraft climb is calculated only for the stowed configuration at the speed for maximum rate of climb \((V_{CF})\) or at transition speed \((V_{TR})\) whichever is larger.
\[ R/C = V_{CF} \left[ \frac{1.2 \, K_{TC} \, (T_{NRP})_{SL, STD}}{W} - \frac{1}{(L/D)} \right] \]

where \( V_{CF} \) is found by setting \( \frac{d(R/C)}{dV_{CF}} = 0 \) and solving for \( V_{CF} \) (note that \( L/D \) is a function of \( V_{CF} \)) and an average altitude \( \frac{h_H + h_{CR}}{2} \) is used.

\[ t_c = \frac{h_{CR} - h_H}{3600} \, (R/C) \, hr. \]

\[ d_c = t_c \, V_{CF} \left[ \frac{3600}{5280} \right] \, mi. \]

\[ F_c = 1.2 \, t_c \, K_{TC} \, (T_{NRP})_{SL, STD} \]

For descent, cruise velocity \( (V_{CR}) \) is maintained and the engines are reduced to idle power. The rate of descent is

\[ V_D = \frac{V_{CR}}{L/D} \]

where \( L/D \) is taken as \( (L/D)_{CR} \) for the helicopter and the compound and dive brakes are used to reduce the \( L/D \) to 6.0 for the stowed rotor.

\[ t_d = \frac{h_{CR} - h_H}{3600} \, V_D \, hr. \]

\[ d_D = t_d \, V_{CR} \left[ \frac{3600}{5280} \right] \, mi. \]
\[ F_D = t_D \frac{(TSFC)_{NRP}}{4} \left[ \frac{T_{NRP}}{4} \right] \text{SL, STD lb.} \]

since the idle fuel flow rate is approximately one quarter that at normal rated power.

Deceleration is assumed to be at an average of 0.20 \( \text{g} \), and at idle power, for all rotary winged aircraft.

\[ t_{DC} = \left[ \frac{V_{CR}}{3600 \times 32.17 \times 0.2} \right] \text{hr,} \]

\[ d_{DC} = t_{DC} \left[ \frac{V_{CR}}{2} \times \frac{3600}{5280} \right] \text{mi.} \]

\[ F_{DC} = t_{DC} \frac{(TSFC)_{NRP}}{4} \left[ \frac{T_{NRP}}{4} \right] \text{SL, STD lb.} \]

The cruise distance \( (d_{CR}) \) is the trip distance \( (D) \) minus the sum of the distance covered in the other phases of the flight.

\[ d_{CR} = D - d_A - d_{TR} - d_C - d_D - d_{DC} \]

\[ t_{CR} = \frac{d_{CR}}{V_{CR}} \]

\[ F_{CR} = t_{CR} \frac{(SFC)_{CR}}{H_{CR}} \text{ or } F_{CR} = t_{CR} \frac{(TSFC)_{CR}}{T_{CR}} \]
The total fuel carried by the aircraft (FTOT) includes an allowance for \( \frac{1}{2} \) minute at normal rated power and \( \frac{1}{2} \) minute at idle power plus a fuel reserve sufficient for 20 minutes of cruise.

\[
F_{\text{START}} = SFC_{\text{NRP}} \left( \frac{0.5}{60} + \frac{0.5}{4(60)} \right)
\]

\[
F_{\text{TOT}} = F_H + F_A + F_{\text{TR}} + F_C + F_D + F_{\text{DC}} + F_{\text{START}} + F_{\text{CR}}
\]

The total fuel burn is:

\[
F_B = F_H + F_A + F_{\text{TR}} + F_C + F_D + F_{\text{DC}} + F_{\text{CR}} + F_{\text{START}}
\]
d) **Component Weights**

Statistical component weight equations are used in calculating the empty weight of the rotary wing aircraft.

The wing weight (for the rotary wing aircraft which have fixed wings) is given by the same formula used for the fixed wing aircraft, assuming no taper or sweep, and modified to use wing lift:

\[
W_W = 0.015 \sqrt{n} K_S \left[ \frac{\sqrt{AR}}{(WL) \cdot 6} \right] \cdot \left( \frac{L_W}{W} \right)^{1.15}
\]

where \(K_S\) is a structural technology factor (normally \(K_S = 1.0\)), \(n = 4.5\) is the ultimate load factor, \(WL = \) wing loading (lb/ft\(^2\)), \((t/c)_W = \) wing thickness/chord ratio (typically 0.12), \(W = \) gross weight (lb), \(L_W = \) wing lift (lb), and \(AR = \) aspect ratio.

The rotor system weight for single rotor aircraft is:

\[
W_R = 14.2 \cdot KR \left[ \left( \frac{R}{5} \right)^{0.25} \sqrt{\frac{HP_{GB}}{\sqrt{\frac{\Pi R^2 \sigma}{10000} K_D}}} \right]^{0.67}
\]

where \(KR\) is a rotor technology factor (normally \(KR = 0.9\) for helicopters and compounds and \(KR = 0.9 KST\) for stowed rotor aircraft where \(KST\) is the weight penalty for stowing...
the rotor, figure R7 shows how DOC varies with $K_{ST}$,
\[ R = \text{rotor radius (ft)}, \quad \sigma = \text{rotor solidity}, \quad HP_{GB} = \text{maximum hp put into the rotor}, \quad V_{TH} = \text{rotor tip speed in hovering (ft/sec)}, \quad \text{and } K_D = \frac{R^{1.6}}{(1200) \ c(t/c)_R} \] is the rotor droop factor which must always be $\geq 1.0$, where $c = \text{rotor chord (ft)}$ and $(t/c)_R = \text{rotor thickness/chord ratio at 25\% radius (typically .12)}$. For tandem rotor aircraft:

\[
W_R = 28.4 \ K_R \left[ \left( \frac{R}{5} \right)^{0.25} \sqrt{\frac{0.6 \ HP_{GB}}{\frac{V_{TH}}{10000} \ \pi \ R^2 \ \sigma}} \ K_D \right]^{0.67}
\]

\[ K_D = \frac{R^{1.6}}{(960) \ c(t/c)_R} \] and must be $\geq 1.0$.

The transmission and drive system weight for single rotor aircraft is:

\[
W_T = 40 \ K_T \left[ \frac{R}{V_{TH}} \ \frac{HP_{GB}}{} \right]^{0.8}
\]

where $K_T$ is a transmission technology factor (typically 0.9). For tandem rotor aircraft:

\[
W_T = 50 \ K_T \left[ \frac{R}{V_{TH}} \ \frac{HP_{GB}}{} \right]^{0.8}
\]

The weight of the tail rotor system (single rotor aircraft only) is

\[ W_{TR} = .088 \ W_R \]
The weight of the tail rotor plus the propeller and the drive system of the compound (single rotor only) is

\[ W_{TR} = 0.088 \, w_R + 16 \, k_R \left[ \frac{\left( \frac{R_p}{5} \right)^{25}}{v_{Tp}} \frac{\rho_p \pi R_p^2}{10000} \right]^{2/3} \]

where \( \sigma_p \) = propeller solidity \( (\sigma_p = 0.25 \text{ typically}, \) lower \( \sigma_p \) results in lower DOC but the propeller becomes too large to fit in the aircraft, \( v_{Tp} \) = propeller tip speed (ft/sec) \( (\text{typically selected to give a tip Mach number in cruise of } 0.82) \), and \( R_p \) = propeller radius (ft.). The propeller disc loading \( (DL_p) \) is based on \( a \, (C_T/\sigma)_\text{prop} = 0.1 \)

\[ DL_p = 0.1 \, \sigma_p \, \rho_{CR} \, v_{Tp}^2 \]

where \( \rho_{CR} \) = air density at cruise altitude. The propeller thrust \( (T_p) \) is determined from the cruise L/D:

\[ T_p = \frac{W}{(L/D)_{CR}} , \quad \pi R_p^2 = T_p / DL_p \]

\[ R_p = \left[ \frac{w}{0.1 \, (L/D)_{CR} \, \sigma_p \, \rho_{CR} \, v_{Tp}^2} \right]^{1/2} \]

All aircraft which have wings are equipped with tail surfaces and the tail system weight (including the
tail rotor where applicable) is increased by 2% of the gross weight to account for the tail. Therefore:

\[ W_{TS} = W_{TR} + 0.02W \]

for winged aircraft.

The fuselage weight for single rotor aircraft is:

\[ W_F = 0.9 \ K_S \ (LF)^{1.5} \ (OD)^{0.25} \ (nW)^{0.15} \]

where \( n = 4.5 \) is the ultimate load factor, \( LF \) is the fuselage length (ft) (see page 59), and \( OD \) is the fuselage diameter (ft) (see page 59). For tandem rotor aircraft:

\[ W_F = 124 \ K_S \left( \frac{W-W_R-W_W}{1000} \right)^7 (OD)^{2} (LF)^{1.5} \ (0.0183)^{0.508} \]

where a 4.5 ultimate load factor is also used.

The landing gear weight is:

\[ W_{LG} = 0.03 \ W \]

The weight of flight controls is:

\[ W_{FC} = 0.318 \ W^{0.77} \]
The installed engine weight or power plant weight is:

\[ W_p = 1.5 \frac{\text{NRP}}{\text{WHP}} \]

where \( 1.5 \) is the installation factor, NRP is normal rated power at sea level, standard conditions, and WHP is the power to weight ratio (6.25 hp/lb for shaft engines and 5 hp/lb or lb st/lb for convertible turbofans).

The hydraulic and electrical system weights are:

\[ W_H = 0.005 W^{1.28}, \quad W_{EL} = 1.61 W^{0.55} \]

The weight of furnishings and equipment is:

\[ W_{FE} = 400 + 50 \ \text{CR} + 40 \ \text{PX} \]

where \( \text{CR} = \) number of crew = 3 and \( \text{PX} = \) number of passengers.

The weight of the air conditioning and de-icing systems is:

\[ W_{AC} = 500 + 13 \ \text{PX} \]

The weight of the fuel system is:

\[ W_{FT} = 0.075 \ \text{FTOT} \]

where \( \text{FTOT} \) is the total weight of fuel carried (lb).
The empty weight (less engines) is:

\[ W_E = W_w + W_R + W_{TS} + W_F + W_{FC} + W_T + W_{LG} + W_H \]

\[ + W_{EL} + W_{ES} + W_I + W_{FE} + W_{AC} + W_{FT} + W_{pp}/3 \]

The weight of the payload plus the crew is:

\[ W_{PL} = 200 \ (PX + CR) \]

The weight of trapped engine oil is:

\[ W_o = 35 \ n_e \]

where \( n_e \) = number of engines.

The gross weight is:

\[ W = W_E + W_{PL} + FTOT + W_o + 2 \ W_{pp}/3 \]

This gross weight can now be compared with the original estimate. If they do not agree within 10 lb. the program goes back to the determination of the rotor design (page 56), and proceeds from there to calculate an improved estimate of the gross weight.

When a converged solution is finally achieved a set of fuel burns and block speeds is computed for various ranges less than the design range (D) but greater
than $D-d_{CR}$, where $d_{CR}$ is the distance covered in cruise flight. These calculations assume that the cruise velocity and fuel flow rate are the same as for the design range case. This information along with the gross weight, empty weight, number of engines, utilization, etc. is then used to calculate DOC vs range using a modified ATA formula (see part III of Ref. 1).

The optimization for minimum DOC of the various input parameters ($S, AR, C_L$, etc.) is not conducted by calculating all possible combinations of 3 or more different values of each of the parameters. Instead a set of optimum values for these parameters is estimated from past experience and these parameters are varied individually about their estimated optimums to find out if they are true optimums. If not, a new set of estimated optimums is chosen and the process is repeated. The final results of these optimizations is shown in the tables of typical aircraft (Figs. R17-R22). The figures showing the variation of DOC vs range as various parameters are varied are all for the typical aircraft with only the parameter in question varying from the typical aircraft case. The value of the parameter which corresponds to the typical case has been put in a box on each of the figures.
IV. DISCUSSION OF AIRCRAFT DESIGNS

a) Optimization of Tilt Wing, Jet Lift and STOL Aircraft

In light of further refinements of the design procedure since the publication of Ref. (1), the conventional tilt wing, jet lift and STOL aircraft were re-evaluated to determine optimum design configurations. Figs. F1 through F3 present the results of a trade-off analysis of altitude and velocity for these three aircraft. The DOC trends and levels are the same as in Ref. (1). Presented in Figs. F4 through F6 are tabulations of the characteristics of favorable nominal designs for these aircraft. These tabulations include weight breakdowns, aircraft dimensions, power system characteristics, fuel consumptions and flight profile distances and time. Fig. F7 is a plot of DOC versus range for these three nominal design aircraft. Fig. F8 is a plot of DOC versus range for 400 mile design aircraft.
b) Off Design Operation

A parametric analysis of the datum aircraft was conducted in order to determine the effect of off design operation on direct operating cost. Graphs are presented for each aircraft, showing the results of both off speed and off altitude operation (F9-F11).

Before outlining the results for the specific aircraft, it is necessary to look at the assumptions behind this parametric analysis. The aircraft were designed to the datum conditions, and then flown off these conditions. In the case of off design velocity operation, the altitude for all aircraft was held fixed at 20,000 ft. In the case of off design altitude operation, as the altitude increased above 20,000 ft., there was an increase in the velocity as well in order to hold the cruise lift coefficient below .5, as specified to provide a stall margin in gusts.

The STOL aircraft direct operating costs increase when the aircraft is flown below design velocity. Above design velocity, the graphs show a decrease in direct operating cost, of no practical significance since the installed normal rated power is inadequate for these
speeds. Off design altitude operation has a very small effect on direct operating cost, amounting to under one percent in the 10,000 to 25,000 foot range. Above 25,000 feet, flight is again not possible due to power limitations (see Fig. F9).

The tilt wing aircraft behaves in much the same manner as the STOL. Here again there is a marked decrease in direct operating cost with increase in cruise velocity. Because of the large installed power, enabling vertical takeoff, there is no power limitation on operating velocity up to about 500 miles per hour. Again, off design altitude flight changes the direct operating cost approximately one percent in the 10,000 through 30,000 foot range (see Fig. F10).

The jet lift aircraft shows a very flat response to off design velocity operation in the 400 to 460 mile per hour range. Above 460 miles per hour, there is an indicated drop in direct operating cost, but flight in this speed range is not possible due to thrust limitations. There is a slight decrease in direct operating cost with increasing altitude, again impossible to realize due to the thrust limitations in the aircraft (See Fig. F11).
In general, the propeller aircraft have rather severe direct operating cost penalties when flown below design speed. The jet lift aircraft has no such penalty. In the case of off design altitude operation, all aircraft show almost no cost penalty in the altitude range of 10,000 to 20,000 feet, but the jet lift and STOL are power limited at the higher altitudes.
c) **Fuselage Parametric Analysis**

Using the improved models for fuselage design and drag estimation, a parametric analysis was made to determine the number of seats abreast which would minimize direct operating cost for various capacity aircraft. This analysis was performed on the jet lift, tilt wing, and STOL aircraft.

Three plots of cents/seat-mile versus number of seats abreast are presented (Figs. F12-F14). The design range, cruise velocity and cruise altitude were held constant at values considered nominal. The discontinuity of the plot between 6 and 7 seats abreast is caused by the addition of a second 18 inch aisle. The minimum cost point (s) for each curve is circled.

After these minimum cost seats abreast configurations were determined, they were incorporated into the design program, so that all aircraft are optimally designed with respect to seats abreast. However, no configuration with seats abreast greater than 6 was employed since the reduction of DOC was insignificant.
d) **Tilt Wing Aircraft Transition**

The tilt wing aircraft was analyzed to determine what aerodynamic problems would be encountered during the transition (acceleration) and retransition (deceleration) phases of flight. Of particular interest was whether or not the transitions could be performed at reasonable acceleration levels.

Transition was analyzed from the point of view that the aircraft should have the capability of performing both transition and retransition while maintaining steady level flight. The important aspect that was investigated was to determine whether or not adequate total lift could be maintained without stalling the wing while maintaining adequate horizontal forces to appropriately accelerate or decelerate the aircraft. It is important that wing stall and buffetting be avoided so that the aircraft control and structural integrity could be maintained. The determination of the flow characteristics over the wing of a tilt wing aircraft with the wing at an angle in forward speed is best determined by wind tunnel tests for particular aircraft design in question. However, the theoretical technique
presented below works quite adequately for determining these flow characteristics.

The technique proposed works well if the aircraft wing is assumed to be fully immersed in the fully developed slipstream of the propeller. Also it is assumed that the propellers fully span the length of the wing with little or no overlap. It is also assumed, for generality, that the propeller and wing can be controlled independently as shown in Fig. F15. The position is measured with respect to the horizon since the aircraft is assumed to be flying in level horizontal flight. However, in the transition analysis of the conventional tilt wing aircraft the wing and propeller axis were taken to be fixed with respect to one another with the wing zero lift line at a nominal 4.6 degrees greater than the propeller axis. This allowed the wing to maintain an adequate lift coefficient in cruise with the propeller axis in a horizontal position.

The nominal tilt wing aircraft analyzed had a gross weight of 57,750 pounds, a wing area of 721 sq. ft., and four propellers, each with a diameter of 19.55 ft. This
aircraft had a nominal design capability of 200 miles range and 400 mph in cruise with an 80 passenger pay-
load. The transition was assumed to be at 1500 ft.
altitude. The profile drag, $C_{D_0}$, was broken into
wing ($C_{D_{ow}}$) and fuselage-empennage ($C_{D_{os}}$) component
parts of .0112 and .0141, respectively.

The wing was taken to have a lift slope of $C_L = 4.9$ per radian. The addition of flaps and leading
edge slots to the wing was assumed to displace and
extend the lift curve, but not to change the slope.

As shown in Fig. F16 the propeller induced velocity
is designated as $v$ while the velocity component parallel
to the propeller axis in the fully developed slipstream
is $2v$. The freestream velocity is designated as $v$.
The velocity through the propeller disc is the vector
sum of $V$ and $v$ and is designated by $V'$ as is also shown
in Fig. F16. Similarly the velocity relative to the
wing, $V''$, is the vector sum of $V$ and $2v$ as shown in
Fig. F17. The relative angle of attack of the wing,
$\alpha''$, is calculated by

$$\alpha'' = \beta'' - (i_n - i_w)$$

(1)
where

$$\tan \beta'' = \frac{V \sin (i_n)}{V \cos (i_n) + 2v}$$

or

$$\sin \beta'' = \frac{V \sin (i_n)}{V \cos (i_n) + 2v}$$

(2)

Thrust is calculated from momentum theory as

$$T = 2 \rho A (V \cos i_n + v) v$$

where

$$A = \text{total propeller area}$$

The component velocity $V \sin i_n$ is assumed to be zero for determining mass flow. This assumption is reasonable since when $i_n$ is large the forward velocity, $V$, will be small and vice versa. The assumption greatly simplifies the theoretical development of the technique. Solving for $2v$ in the above equation we have

$$2v = -v \cos i_n + \sqrt{v^2 \cos^2 (i_n) + \frac{2T}{\rho A}}$$

(3)

Also solving for $(V'')^2$ from Fig. F17

$$(V'')^2 = (V \sin i_n)^2 + (V \cos i_n + 2v)^2$$

(4)
Substituting $2v$

$$(V'')^2 = (V \sin \theta)^2 + (V \cos \theta)^2 + \frac{2T}{\rho A} \quad (5)$$

$$= V^2 + \frac{2T}{\rho A}$$

defining $q''$ as the dynamic pressure in the slipstream we have

$$q'' = \frac{1}{2} \rho (V'')^2 = \frac{1}{2} \rho V^2 + \frac{T}{A} = q + \frac{T}{A} \quad (6)$$

where $q$ is the free stream dynamic pressure. We also define the thrust coefficient, $T_c''$, to be based on the slipstream conditions as

$$T_c'' = \frac{T}{q''A} \quad (7)$$

Appropriate substitution of Equation 6 results in

$$T_c'' = 1 - \frac{q}{q''} \quad (8)$$

It can also be shown that

$$\frac{V''}{V} = \sqrt{\frac{1}{1 - T_c''}} \quad (9)$$

Substitution of Equation 9 into 2 results in
\[ \tan \beta'' = \frac{\sin i_n}{\sqrt{(\cos i_n)^2 + \frac{T_c''}{1-T_c''}}} \] (10)

or

\[ \sin \beta'' = \frac{\sin i_n}{\sqrt{\frac{1}{1-T_c''}}} \]

The total lift generated by the wing is proportional to the total mass flow through a cylinder of air whose diameter equals the wing span (Fig. F18). The wing lift is based on \( V'' \). Therefore, since the mass flow per unit cross-sectional area of the slipstream of the propeller is greater than the remaining mass flow, the total lift must be appropriately reduced. This is accomplished by introducing a corrected lift coefficient, \( C'_L \), as follows

\[ C'_L = \frac{m'}{m} \cdot C_L \] (11)

where

\[ m = \text{the actual mass flow.} \]

\[ m'' = \text{the mass flow if the total cylinder mass flow was assumed to be at } V''. \]
That is,

\[ m = \rho (V + v)A + \rho V \left[ \frac{\pi b^2}{4} - A \right] \]  

(12)

\[ m'' = \rho (V + v) \frac{\pi b^2}{4} \]  

(13)

where

\[ b = \text{wing span} \]

Therefore,

\[ \frac{m}{m''} = \frac{2 + \frac{2v}{V} \frac{A}{\pi b^2/4}}{2 + \frac{2v}{V}} \]  

(14)

Using Equation 3 and assuming the angle of attack effect is small when calculating the mass flow (since \( i_n \) is small for large \( T'' \)), we have

\[ 2v = -V + \sqrt{V^2 + \frac{2T}{A}} \]  

(15)

Rearranging terms

\[ \frac{2v}{V} = -1 + \sqrt{1 + \frac{1}{q} \frac{T}{A}} \]
Substituting equations 7 and 8

\[
\frac{2v}{V} = -1 + \sqrt{1 + \frac{T_A}{q''}} \frac{q'''}{q}
\]

Substituting equation 17 into 14 results in the useful equation

\[
\frac{m}{m''} = \frac{2 + \frac{A}{\Pi b^2/4}}{1 + \sqrt{1 - \frac{1}{1 - T_{C''}}}}
\]

The effect of the mass flow correction on \( C_L \) is illustrated in Fig. F19. The fuselage-empennage drag is determined by

\[
D_f = q S C_{Dfuselage}
\]

The lift and drag of the wing are calculated in the \( V'' \) axis system as

\[
L_w = q'' S C_{L''} = \text{Wing Lift}
\]

\[
D_w = q'' S C_{D''} = \text{Wing Drag}
\]
where

\[ C_D'' = C_{D_{0w}} + \frac{C_{L}^2}{e \cdot AR} \]

\[ e = \text{wing efficiency factor} \]

\[ AR = \text{wing aspect ratio} \]

The vertical and horizontal acceleration of the aircraft is then determined by

\[ a_y = \frac{1}{m} (T \sin i_n + I_{wz} \cos \varphi - D_{wz} \sin \varphi) - g \quad (22) \]

= vertical acceleration, positive up

where

\[ \varphi = i_{w} - \alpha'' \]

\[ m = \text{aircraft mass} \]

\[ a_x = \frac{1}{m} (T \cos i_n - I_{wz} \sin \varphi - D_{wz} \cos \varphi - D_f) \quad (23) \]

= horizontal acceleration, positive forward

Determining the transition performance of the tilt wing was based on the iterative solution of all these equations. The aircraft was evaluated at several repre-
sentative forward velocities, and wing and propeller angles while maintaining level flight.

It was found that for the conventional tilt wing designs there were no transition (acceleration) problems. The angles of attack of the wing were always below stall and at the thrust levels available, acceleration equal to or greater than that considered adequate were always obtainable.

However, retransition did pose some problems. Alternate thrust levels were investigated while the wing incidence was maintained at an angle (relative to the fuselage) which would result in a high angle of attack, but less than $\alpha''$ stall (about 15 degrees). The maximum aerodynamic forces tending to decelerate the aircraft were then determined. It was concluded that to maintain level flight with reasonable deceleration levels, high lift devices on the wings, such as flaps and leading edge slats, would be necessary for a conventional tilt wing aircraft. The major problem is that for a conventional wing without high lift devices, adequate lift from the wings and props cannot be obtained to maintain level flight without the thrust
levels becoming so large that the aircraft is accelerated rather than decelerated. This is due to the fact that the thrust vector is at a relatively small angle to the horizontal, even though the wing is near the stall angle.

The addition of slats alone allows the wing and thrust line to be maintained at a higher angle before encountering wing stall, resulting in a lower required thrust. The resulting horizontal thrust will then be smaller, and therefore, deceleration will be larger. The addition of flaps will increase wing lift and, hence, also decrease the thrust level required, therefore, again increasing deceleration. Even larger benefits of increased deceleration are obtained by adding both flaps and slats.

It is also quite apparent that the deceleration wing stall problem becomes more severe for a tilt wing aircraft which is descending while in the retransition mode because of the resultant higher angle of attack encountered. In this case, the requirement for high lift devices becomes even more necessary.
Analysis of a hybrid tilt wing-tilt propeller was performed to determine its advantage during transition over the conventional tilt wing configuration. It was found that reasonable deceleration levels for this hybrid configuration could be obtained without the use of any high lift devices on the wing. This is the result of always being able to 1) maintain the wing at an angle of attack but, below stall and 2) maintain the thrust line at a large angle from the horizontal. Therefore, a major portion of the vertical lift can be obtained from the propellers without adverse horizontal acceleration.
e) **Design Variations of the Tilt Wing Aircraft**

One of the principle shortcomings of the conventional tilt wing VTOL aircraft is the problem of wing stall during transition modes, particularly retransition (deceleration). To improve the stall characteristics of the wing during this critical portion of the flight, it is desirable to maintain a high propeller slipstream velocity over the wing. For the conventional tilt wing aircraft considered in this report, this was accomplished in the design by requiring a propeller disc loading to wing loading ratio of 0.6. However, the main ratio is that the wing is not then optimally designed for cruise flight conditions except at the higher propeller disc loadings which are not generally the optimum ones. Consequently, the wing is usually larger than it would be if optimally designed.

One alternate technique for insuring that the wing will not stall during the transition period is to control the attitude of the wing and the propeller independently. Controlling the attitude of the propeller could be accomplished by either allowing the propeller-
engine combination to rotate about the wing or by controlling only the attitude of the propeller. As presented in the section "Tilt Wing Aircraft Transition", the operational advantages of this configuration are improved deceleration capabilities and better wing stall control. This configuration also results in slightly lower direct operating costs (DOC) because of the advantage of being able to design the wing for cruise conditions. This cost advantage is shown in Fig. F20. Presented is direct operation cost versus stage length for the conventional tilt wing and a hybrid tilt wing - tilt propeller. The design disc loading of the hybrid aircraft was fixed at the same value as the conventional aircraft (50 psf) but, the calculated optimal wing loading is higher due to its cruise design. Both aircraft are equipped with four engines and four propellers.

It was also found that reductions in the disc loading of the hybrid design resulted in further reductions of the direct operating cost, Fig. F21. This is due to increased hover propulsion efficiency and therefore lower required hover horsepower. Since, for
high disc loading, the installed power is based on hover power requirements, the reduction in DOC is the direct result of lower installed power. The reduction of DOC holds true up to the point where the installed power is determined by the specified cruise conditions.

It can also be seen from Fig. F21 that at the higher disc loadings the wing is designed for cruise conditions. As the disc loading is decreased the diameter of the propellers increase. Therefore at the low disc loadings the propeller diameter is so large that the wing size must then be based on the requirement that the propellers fit on the wings. This results in a heavier and more costly wing as well as higher drag. However, in the region where the installed power is still based on hover requirements, the savings in power cost due to reduction in the disc loading outweighs the cost of increased wing size and drag. At very low disc loadings power is determined by cruise requirements, therefore, any further reduction in disc loading only results in an increased wing and propeller weight. The direct
operating cost becomes quite large at this design point. Therefore, the minimum cost design point is at the boundary between power determined by hover and cruise. For the case where the wing was designed to accommodate large propellers, the design criteria used was that 1) the tip of the propeller would be no closer than two feet from the fuselage, 2) overlap of adjacent propellers would be no more than 20 percent of the propeller diameter, and 3) the outboard propeller could be placed at or near the wing tip. Also presented in Fig. F21 is a $\sigma = .25$ line. Only disc loadings which result in propeller solidity ($\sigma$) less than .25 are considered to be practical propeller designs.

Figs. F22, F23 and F24 present the results of a cruise speed and altitude tradeoff study for the four propeller hybrid tilt wing-propeller aircraft. The cruise velocity was held fixed at 350, 400, or 450 mph while the cruise altitude was varied from 10,000 to 30,000 feet. The plots illustrate that the design optimized at the lower cruise altitude. This is in contrast to the conventional tilt wing which optimizes at higher altitudes. For Figs. F22, F23 and F24 the
wing was designed on the basis of accommodating the propellers for all levels of disc loading shown.

A two propeller-four engine hybrid tilt wing aircraft design was also analyzed. The results of a cruise altitude-velocity trade-off study are presented in Figs. F25 through F27. The same design criteria and comments apply here as applied to the four propeller hybrid case. It can be seen that the reduction of DOC with decreasing cruise altitude is more pronounced here than in the four propeller case.

A comparison of direct operating cost for the four propeller and two propeller hybrid tilt wing aircraft is shown in Fig. F28 which is a plot of DOC versus disc loading. The cruise altitude and velocity chosen is for minimum direct operating cost of each aircraft.

Presented in Fig. F29 is a plot of DOC versus range for the nominal designs of the conventional tilt wing and the 2 and 4 propeller hybrid tilt wing. The design characteristics for the two hybrid aircraft is presented in Figs. F30 and F31. The characteristics
of the conventional tilt wing are presented in an earlier section.

The data presented for the 2 and 4 propeller hybrid tilt wing-propeller aircraft were for wing designs with an aspect ratio of 9.5. Figs. F32 and F33 illustrate the effect on DOC due to changes in wing aspect ratio for these same aircraft. It is observed that the DOC decreases as the aspect ratio increases due to the reduction in wing area and weight. However, the cost advantage of designing high aspect ratio wings is quite small, especially for the two propeller case. Aeroelastic problems associated with the propeller whirl mode could force a maximum aspect ratio in the vicinity of 6 to 9 depending on the design characteristics which determine rotor and mounting stiffnesses. Clearly no major penalty will result from this constraint, should it exist.

Fig. F32 presents DOC versus range for the nominally designed conventional tilt wing aircraft with various wing aspect ratios.
It is observed that in this case the cost decreases with decreasing aspect ratios due to reduced wing weight.

As previously stated the trend of DOC with changes in design cruise altitude is different for the conventional and hybrid tilt wing aircraft. A discussion of these trends is presented below.

Tabulation of some of the important design and operation characteristics of these aircraft types is shown in Figs. F35 through F37. Design altitudes of 10,000, 20,000 and 30,000 feet are considered for each case. The disc loading for the 4 propeller and 2 propeller hybrid aircraft are 30 psf and 15 psf, respectively. These disc loadings are at or near the optimal for each design case.

The major factors of particular interest in these tabulations are the gross weight (3), wing area (5), total cost in dollars per hour (12), total cost less fuel cost in dollars per hour (13), block time (17) and fuel burned (4).

The following operational relationships are
1. Utilization (hrs/yrs) is assumed to remain constant.

2. The total cost less the fuel cost in dollars per hour (13) is an approximate measure of the aircraft cost (purchase plus maintenance). The larger this cost, the larger the DOC in cents per seat mile (2).

3. The lower the block time (17), the lower the DOC since more miles are flown per year.

4. The larger the gross weight (3), the larger the total cost less fuel cost, in dollars per hour.

5. The higher the aircraft flies, the higher the block time and the lower the fuel cost, except in one case, that of the 2 propeller hybrid aircraft at 30,000 feet.

The reversal of the cost trend from the conventional to the hybrid tilt wing hinges on the design of the wing,
This is dramatically brought out in the comparison of the conventional tilt wing and the two propeller hybrid tilt wing. For the hybrid, the wing area increases significantly with increases in altitude since the wing is nearly optimally designed for cruise except for the restriction of $C_{l_{\text{max}}} = 0.5$. This results in large increases in gross weight and therefore aircraft purchase and maintenance cost. The higher cruise altitude has also resulted in larger block time while fuel burned decreased only slightly. The trend of each of these three factors contributes to the increase in DOC. There are no factors which would significantly contribute to a reduction in DOC.

In contrast, the wing area of the conventional tilt wing aircraft remained almost constant since it is designed on the wing loading - disc loading ratio requirements. For the same reason the wing area is much larger than optimal and this explains the high overall DOC. Despite the increase in block time, the reduction in fuel burned was large enough to result in a slight decrease in DOC with increasing
Similar analysis of the 4 propeller hybrid aircraft reveals that the aircraft exhibits cost trend characteristics of both other aircraft discussed above. This is because, being a hybrid tilt wing, its wing sizing is determined by other factors, such as propeller diameter. The result is that DOC does decrease from 10,000 to 20,000 feet altitude but then increases again at 30,000 feet.
f) Tilt Wing STOL

Some preliminary study of a tilting wing STOL aircraft has been carried out to compare this type of aircraft with the standard tilt wing and the standard STOL. The method used is essentially that discussed in Section II except that the assumption involving the neglect of the cross flow component $V \sin(i)$ in determining mass flow through the rotor has been eliminated since this component is now appreciable.

From an economics standpoint, one of the most obvious advantages of the tilting wing STOL over the standard STOL aircraft is the decrease in ground acquisition costs. Depending upon the wing angle of incidence, takeoff ground run distances of from 100 feet to 150 feet were obtained for this tilting wing STOL. This is substantially less than the 500 feet allowed for the standard STOL. A second advantage of the tilting wing STOL when compared to the standard STOL is in the wing design. For the standard STOL, the wing loading was held at 50 psf in order to achieve
takeoff in under 500 feet. This gives, however, a non-optimum wing for cruise. For the tilting wing STOL, the combination of the vertical component of thrust and the lift generated by the wing make it unnecessary to constrain the wing loading to any particular value. The wing is thus designed for cruise, where the optimum wing loading is 83 psf, while still achieving a ground run of only 250 feet at .5 acceleration, as compared to the 500 feet required for the STOL. This higher wing loading, and hence smaller wing, lead to a 7 percent decrease in direct operating cost, as seen in Fig. F38. This decrease is mainly in maintenance costs due to component weight decreases, the fuel burn being little affected.

In a comparison with the standard tilt wing, the tilting wing STOL is more economical from the standpoint of direct operating costs at the longer ranges, above 150 miles, but less economical at the shorter ranges. This is also clearly shown in Fig. F38. Looking again at the ground acquisition problem, the standard tilt wing is still more economical than the tilting wing STOL, requiring less land.
Multiple Hops Operations

Each vehicle type was evaluated to determine its potential capability for multiple hop operation without refueling. The results of both the 200 and 400 mile design range vehicles of each type are shown in the accompanying plots.

For all aircraft types, ground operation consisted of 5 minutes of idle at each stop with all engines operating at idle, except for the jet lift aircraft. For this vehicle, preflight equipment checkout with engines operating at normal rated power for .6 minutes was performed only at the beginning of the first hop. Decrease in weight due to fuel burned was taken into account at the end of each hop, before the next hop was embarked upon.

The results for 200 and 400 mile design aircraft are presented graphically in Fig. F39 and Fig. F40. The jet lift aircraft is penalized by the high fuel burn in takeoff. This makes the jet lift aircraft
a poor vehicle for short hop operations. The STOL is also penalized in short hop operation by the air maneuver time requirement. This added fuel burn decreases the multiple hop capability. The best winged aircraft for multiple hops is the tilt wing. The tilt wing possesses the dual assets of low fuel burn in take-off and no air maneuver requirements. These factors combine to give the tilt wing the best multiple hops capabilities.
h) **Rotary Wing Aircraft**

Three different types of rotary winged aircraft have been considered:

1) A tandem rotor helicopter. An equivalent single rotor helicopter gives about the same performance, the choice was arbitrary.

2) A single rotor stowed rotor aircraft using convertible turbofan engines which can provide thrust for cruise or shaft power to the rotor for hover and low speed flight.

3) A single rotor compound with a wing and using a tail mounted propeller for high speed forward flight. A tandem rotor compound with a wing but no propeller was also investigated but the results are not presented here because they are so similar to the single rotor compound.

The standard design mission is to carry 80 passengers (at 200 lbs. per passenger) and a 3 man crew.
(2 flight crew and 1 steward-baggage handler) over a 200 mile stage with a 20 minute cruise fuel reserve and sufficient installed power to allow one engine-out hover at sea level on a 550°F hot day. Cruise velocity and altitude were selected to minimize the direct operating cost (DOC), see Figs. R9-R14. For the short stage lengths of interest here the advantage of a higher L/D obtained by cruising at altitude does not compensate for the associated block time penalty. For a detailed discussion of the optimization of various design parameters and the design procedures and assumptions used see Rotary Winged Aircraft Design Procedure, page 50.

Figure R15 compares the DOC vs stage length for the rotary winged typical aircraft (as presented in Figs. R17-R22). The stowed rotor aircraft is not operated below 40 miles since it takes 36.5 miles to accelerate, transition, climb, descend, transition and decelerate. Figure R16 compares the DOC vs. stage length for selected aircraft all designed to carry 80 passengers for 200 miles.
REFERENCES


TILT WING - ALTITUDE AND CRUISE SPEED OPTIMIZATION

80 Passengers
Design Range = 200 mi

VCR = 300 mph
VCR = 350 mph
VCR = 400 mph
VCR = 450 mph

DOC ($$/seat mi)

ALTITUDE (ft)

FIGURE F1
JET LIFT - ALTITUDE AND CRUISE SPEED OPTIMIZATION

80 Passengers  Design Range = 200 mi

VCR = 350 mph
VCR = 400 mph
VCR = 450 mph
VCR = 500 mph
VCR = 550 mph

FIGURE F2
2.1

VCR = 350 mph

VCR = 300 mph

VCR = 250 mph

15,000

STOL - ALTITUDE & CRUISE SPEED OPTIMIZATION

80 Passengers

Design Range = 200 mi

DOC (\$ / seat mi)

FIGURE F3

ALTITUDE (ft)

80 Passengers Design Range = 200 mi

VCR = 400 mph

15,000

1.4

1.5

1.6

1.7

1.8

1.9

2.0

2.1

5,000

25,000
FIG. F4. CHARACTERISTICS OF THE CONVENTIONAL TILT WING AIRCRAFT

80 Passengers - 200 mi. design range
Cruise Altitude 20,000 ft.
Cruise Speed - 400 m.p.h.

<table>
<thead>
<tr>
<th>Structure</th>
<th>Weight (lbs)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Wing</td>
<td>6,507</td>
</tr>
<tr>
<td>Fuel System</td>
<td>192</td>
</tr>
<tr>
<td>Flight Control</td>
<td>1,222</td>
</tr>
<tr>
<td>Tail</td>
<td>1,528</td>
</tr>
<tr>
<td>Fuselage</td>
<td>6,684</td>
</tr>
<tr>
<td>Landing Gear</td>
<td>1,833</td>
</tr>
<tr>
<td>Propulsion System</td>
<td></td>
</tr>
<tr>
<td>Engines</td>
<td>3,781</td>
</tr>
<tr>
<td>Installation</td>
<td>1,891</td>
</tr>
<tr>
<td>Propellers</td>
<td>3,600</td>
</tr>
<tr>
<td>Transmission</td>
<td>5,374</td>
</tr>
<tr>
<td>Navigation Instruments</td>
<td>200</td>
</tr>
<tr>
<td>Hydraulics</td>
<td>669</td>
</tr>
<tr>
<td>Electrical Equipment</td>
<td>691</td>
</tr>
<tr>
<td>Electronics</td>
<td>642</td>
</tr>
<tr>
<td>Furnishings</td>
<td>3,750</td>
</tr>
<tr>
<td>Air Conditioning &amp; De-Icing Eqt.</td>
<td>1,540</td>
</tr>
<tr>
<td>Weight Empty</td>
<td>40,104</td>
</tr>
<tr>
<td>Payload &amp; Crew</td>
<td>16,600</td>
</tr>
<tr>
<td>Trapped Oil</td>
<td>140</td>
</tr>
<tr>
<td>Fuel</td>
<td>4,282</td>
</tr>
<tr>
<td><strong>GROSS WEIGHT</strong></td>
<td><strong>61,126</strong></td>
</tr>
</tbody>
</table>

(L/D) Cruise = 12.8

<table>
<thead>
<tr>
<th>Aircraft Characteristics</th>
</tr>
</thead>
<tbody>
<tr>
<td>Wing:</td>
</tr>
<tr>
<td>Span = 83.4 ft</td>
</tr>
<tr>
<td>Aspect Ratio = 9.5</td>
</tr>
<tr>
<td>Area = 733 sq. ft.</td>
</tr>
<tr>
<td>Wing Loading = 83.4 psf</td>
</tr>
<tr>
<td>Taper Ratio = .5</td>
</tr>
<tr>
<td>Sweepback Angle = 0</td>
</tr>
<tr>
<td>Fuselage:</td>
</tr>
<tr>
<td>Length = 77.2 ft</td>
</tr>
<tr>
<td>Diameter = 12.6</td>
</tr>
<tr>
<td>Seats Abreast = 6</td>
</tr>
<tr>
<td>Engines = 4 at 5,026 HP each</td>
</tr>
<tr>
<td>30 minute rating</td>
</tr>
<tr>
<td>Propeller:</td>
</tr>
<tr>
<td>Disc Loading = 50.04 psf</td>
</tr>
<tr>
<td>Diameter = 19.7 ft</td>
</tr>
<tr>
<td>Solidity = .25</td>
</tr>
<tr>
<td>4 propellers</td>
</tr>
</tbody>
</table>

Fuel Breakdown: Range

<table>
<thead>
<tr>
<th></th>
<th>Fuel lbs.</th>
<th>(st. Mi.)</th>
<th>Time (hrs.)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Hover &amp; Warmup</td>
<td>142</td>
<td>----</td>
<td>.0033</td>
</tr>
<tr>
<td>Acceleration &amp; Climb</td>
<td>664</td>
<td>13.6</td>
<td>.0721</td>
</tr>
<tr>
<td>Cruise</td>
<td>1642</td>
<td>138.1</td>
<td>.3453</td>
</tr>
<tr>
<td>Descent &amp; Decelerat.</td>
<td>265</td>
<td>48.3</td>
<td>.1292</td>
</tr>
<tr>
<td>Reserves</td>
<td>1569</td>
<td>----</td>
<td>----</td>
</tr>
<tr>
<td><strong>TOTALS</strong></td>
<td><strong>4282</strong></td>
<td><strong>200.0</strong></td>
<td><strong>.5499</strong></td>
</tr>
</tbody>
</table>
**DOC Input Parameters**

- Number of Engines = 4
- Max. Cruise Speed = 400 mph
- HP of One Engine = 5,022 hp (NRP)
- Thrust of One Engine = -
- Weight of One Engine = 945 lb.
- Number of Passengers = 80
- Design Range = 200 mi.
- Propeller (Rotor) Weight = 3,600 lb
- Empty Weight (less engines) = 36,462 lb.
- Gross Weight = 61,126 lb.
- Annual Utilization = 3000 hr.
- Depreciation Period = 12 yr.
- Production Run (Airframe) = 300
- $/lb engines = $300/lb.
- Maintenance Factor = 1.3
- Engine TBO = 4000 hr.

**DOC Output**

<table>
<thead>
<tr>
<th>Stage Length (mi.)</th>
<th>10</th>
<th>20</th>
<th>30</th>
<th>40</th>
<th>50</th>
<th>100</th>
<th>200</th>
</tr>
</thead>
<tbody>
<tr>
<td>Flight Operations ($/mi)</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Pilot</td>
<td>12.8</td>
<td>10.9</td>
<td>10.7</td>
<td>10.3</td>
<td>10.0</td>
<td>10.0</td>
<td>9.6</td>
</tr>
<tr>
<td>Copilot</td>
<td>8.1</td>
<td>6.8</td>
<td>6.6</td>
<td>6.3</td>
<td>6.2</td>
<td>6.2</td>
<td>5.8</td>
</tr>
<tr>
<td>Fuel</td>
<td>82.2</td>
<td>57.1</td>
<td>44.9</td>
<td>41.1</td>
<td>38.8</td>
<td>27.2</td>
<td>24.3</td>
</tr>
<tr>
<td>Insurance</td>
<td>21.3</td>
<td>16.7</td>
<td>16.1</td>
<td>15.1</td>
<td>14.5</td>
<td>14.6</td>
<td>13.4</td>
</tr>
<tr>
<td>TOTAL</td>
<td>124.4</td>
<td>91.6</td>
<td>78.3</td>
<td>72.8</td>
<td>69.5</td>
<td>58.0</td>
<td>53.1</td>
</tr>
<tr>
<td>Maintenance Burden ($/mi)</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Labor Airframe</td>
<td>9.6</td>
<td>7.5</td>
<td>7.3</td>
<td>6.8</td>
<td>6.5</td>
<td>6.6</td>
<td>6.0</td>
</tr>
<tr>
<td>Materials Airframe</td>
<td>13.1</td>
<td>10.2</td>
<td>9.8</td>
<td>9.2</td>
<td>8.9</td>
<td>8.9</td>
<td>8.1</td>
</tr>
<tr>
<td>Materials Engines</td>
<td>33.1</td>
<td>26.0</td>
<td>25.0</td>
<td>23.4</td>
<td>22.5</td>
<td>22.6</td>
<td>20.7</td>
</tr>
<tr>
<td>Labor Engines</td>
<td>2.3</td>
<td>1.8</td>
<td>1.7</td>
<td>1.6</td>
<td>1.6</td>
<td>1.6</td>
<td>1.4</td>
</tr>
<tr>
<td>TOTAL</td>
<td>58.1</td>
<td>45.6</td>
<td>43.8</td>
<td>41.1</td>
<td>39.5</td>
<td>39.6</td>
<td>36.3</td>
</tr>
<tr>
<td>Depreciation Items ($/mi)</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Aircraft</td>
<td>21.7</td>
<td>17.0</td>
<td>16.4</td>
<td>15.4</td>
<td>14.8</td>
<td>14.8</td>
<td>13.6</td>
</tr>
<tr>
<td>Engines</td>
<td>11.8</td>
<td>9.2</td>
<td>8.9</td>
<td>8.3</td>
<td>8.0</td>
<td>8.0</td>
<td>7.4</td>
</tr>
<tr>
<td>Props (Rotors)</td>
<td>2.6</td>
<td>2.0</td>
<td>1.9</td>
<td>1.8</td>
<td>1.7</td>
<td>1.7</td>
<td>1.6</td>
</tr>
<tr>
<td>Electronics</td>
<td>4.4</td>
<td>3.5</td>
<td>3.3</td>
<td>3.1</td>
<td>3.0</td>
<td>3.0</td>
<td>2.7</td>
</tr>
<tr>
<td>Airframe Spares</td>
<td>2.6</td>
<td>2.0</td>
<td>1.9</td>
<td>1.8</td>
<td>1.8</td>
<td>1.8</td>
<td>1.6</td>
</tr>
<tr>
<td>Engine Spares</td>
<td>8.8</td>
<td>6.9</td>
<td>6.7</td>
<td>6.3</td>
<td>6.0</td>
<td>6.0</td>
<td>5.5</td>
</tr>
<tr>
<td>TOTAL</td>
<td>51.9</td>
<td>40.7</td>
<td>39.1</td>
<td>36.7</td>
<td>35.3</td>
<td>35.3</td>
<td>32.4</td>
</tr>
<tr>
<td>TOTAL DOC ($/mi)</td>
<td>255.8</td>
<td>194.6</td>
<td>177.3</td>
<td>165.7</td>
<td>158.8</td>
<td>147.5</td>
<td>135.1</td>
</tr>
<tr>
<td>($/hr)</td>
<td>566.5</td>
<td>545.3</td>
<td>515.0</td>
<td>511.6</td>
<td>509.4</td>
<td>470.6</td>
<td>468.2</td>
</tr>
<tr>
<td>($/av. seat)</td>
<td>0.32</td>
<td>0.49</td>
<td>0.67</td>
<td>0.83</td>
<td>0.99</td>
<td>1.84</td>
<td>3.38</td>
</tr>
<tr>
<td>($/av. seat-mi)</td>
<td>3.20</td>
<td>2.43</td>
<td>2.22</td>
<td>2.07</td>
<td>1.98</td>
<td>1.84</td>
<td>1.69</td>
</tr>
</tbody>
</table>

**Stage Length (mi)**: 10, 20, 30, 40, 50, 100, 200

**Fuel Burn (lb)**: 467, 649, 762, 929, 1096, 1529, 2728

**Block Speed (mph)**: 227, 290, 301, 321, 334, 334, 364
FIG. F5 CHARACTERISTICS OF THE JET LIFT

80 passengers - 200 mi. design range
Cruise Altitude 20,000 ft.
Cruise Speed 450 mph

<table>
<thead>
<tr>
<th>Structure</th>
<th>Weight (lbs)</th>
<th>Aircraft Characteristics</th>
</tr>
</thead>
<tbody>
<tr>
<td>Wing</td>
<td>3627</td>
<td>Wing:</td>
</tr>
<tr>
<td>Fuel System</td>
<td>252</td>
<td>Span = 49.1 ft.</td>
</tr>
<tr>
<td>Flight Controls</td>
<td>1168</td>
<td>Aspect Ratio = 6</td>
</tr>
<tr>
<td>Tail</td>
<td>1357</td>
<td>Area = 402 sq. ft.</td>
</tr>
<tr>
<td>Fuselage</td>
<td>7523</td>
<td>Wing Loading = 135 psf</td>
</tr>
<tr>
<td>Landing Gear</td>
<td>1628</td>
<td>Taper Ratio = .5</td>
</tr>
<tr>
<td>Propulsion System</td>
<td></td>
<td>Sweepback Angle = 30°</td>
</tr>
<tr>
<td>Engines</td>
<td>4327</td>
<td></td>
</tr>
<tr>
<td>Installation</td>
<td>4327</td>
<td></td>
</tr>
<tr>
<td>Navigation Instruments</td>
<td>200</td>
<td>Fuselage:</td>
</tr>
<tr>
<td>Hydraulics</td>
<td>575</td>
<td>Length = 84.5 ft.</td>
</tr>
<tr>
<td>Electrical Equipment</td>
<td>646</td>
<td>Diameter = 12.6 ft.</td>
</tr>
<tr>
<td>Electronics</td>
<td>642</td>
<td>Seats Abreast = 6</td>
</tr>
<tr>
<td>Furnishings</td>
<td>3750</td>
<td>Engines:</td>
</tr>
<tr>
<td>Air Conditioning &amp; De-icing Eqt.</td>
<td>1540</td>
<td>Cruise - 2 at 6075 lb. thrust each; 30-minute rating</td>
</tr>
<tr>
<td>Weight Empty</td>
<td>31562</td>
<td>Lift - 12 at 5812 lb. thrust ' each; 30-minute rating</td>
</tr>
<tr>
<td>Payload and Crew</td>
<td>16600</td>
<td></td>
</tr>
<tr>
<td>Trapped Oil</td>
<td>490</td>
<td></td>
</tr>
<tr>
<td>Fuel</td>
<td>5624</td>
<td></td>
</tr>
<tr>
<td>GROSS WEIGHT</td>
<td>54,276</td>
<td>Fuel Breakdown</td>
</tr>
<tr>
<td>(L/D)</td>
<td></td>
<td>Fuel (lbs) Range (mi) Time (hr)</td>
</tr>
<tr>
<td>Cruise</td>
<td></td>
<td>Hover &amp; Warm-up 608 30.3  .0133</td>
</tr>
<tr>
<td></td>
<td></td>
<td>Acceleration &amp; cl. 1135 30.3  .0731</td>
</tr>
<tr>
<td></td>
<td></td>
<td>Cruise 1348 130.9  .2909</td>
</tr>
<tr>
<td></td>
<td></td>
<td>Restart 437</td>
</tr>
<tr>
<td></td>
<td></td>
<td>Descent &amp; Decel. 566 38.8  .0976</td>
</tr>
<tr>
<td></td>
<td></td>
<td>Reserves 1529</td>
</tr>
<tr>
<td></td>
<td></td>
<td>TOTAL 5623 200.0  .4749</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>
Fig. F5a) JET LIFT

DOC Input Parameters

- Number of lift engines = 12
- Number of Cruise Engines = 12
- Max. Cruise Speed = 450 mph
- HP of One Engine =
- Thrust of One Lift Engine = 5812 lb.
- Thrust of One Cruise Engine = 5954 lb.
- Weight of One Lift Engine = 253 lb.
- Weight of One Cruise Engine = 674 lb.
- Number of Passengers = 80
- Design Range = 200 mi.
- Propeller (Rotor) Weight =
- Empty Weight (less engines) = 27,725 lb.
- Gross Weight = 54,276 lb.
- Annual Utilization = 3,000 hr.
- Depreciation Period = 12 yr.
- Production Run (Airframe) = 300
- $/lb Engines = $150/lb.
- Maintenance Factor = 1.0
- Engine TBO = 4000 hr.

DOC Output

<table>
<thead>
<tr>
<th>Flight Operations (¢/mi)</th>
<th>Stage length (mi.)</th>
<th>10</th>
<th>20</th>
<th>30</th>
<th>40</th>
<th>50</th>
<th>100</th>
<th>200</th>
</tr>
</thead>
<tbody>
<tr>
<td>Pilot</td>
<td>14.4</td>
<td>11.4</td>
<td>10.4</td>
<td>10.0</td>
<td>9.7</td>
<td>9.1</td>
<td>8.8</td>
<td></td>
</tr>
<tr>
<td>Copilot</td>
<td>9.2</td>
<td>7.1</td>
<td>6.5</td>
<td>6.1</td>
<td>5.9</td>
<td>5.5</td>
<td>5.3</td>
<td></td>
</tr>
<tr>
<td>Fuel</td>
<td>278.5</td>
<td>260.4</td>
<td>131.8</td>
<td>104.7</td>
<td>88.5</td>
<td>54.1</td>
<td>36.0</td>
<td></td>
</tr>
<tr>
<td>Insurance</td>
<td>17.9</td>
<td>12.8</td>
<td>11.1</td>
<td>10.2</td>
<td>9.7</td>
<td>8.6</td>
<td>8.1</td>
<td></td>
</tr>
<tr>
<td>TOTAL</td>
<td>320.1</td>
<td>291.8</td>
<td>159.8</td>
<td>131.0</td>
<td>113.7</td>
<td>77.3</td>
<td>58.2</td>
<td></td>
</tr>
</tbody>
</table>

Direct Maintenance (¢/mi)

| Aircraft       | 21.7               | 15.4| 13.3| 12.3| 11.6| 10.4| 9.8  |
| Materials Airframe | 9.5             | 6.7 | 5.8 | 5.4 | 5.1 | 4.5 | 4.3  |
| Materials Engines | 14.7          | 8.4 | 6.6 | 5.5 | 4.8 | 3.8 | 2.9  |
| Labor Engines   | 7.3                | 3.9 | 3.0 | 2.3 | 2.0 | 1.4 | 0.9  |
| TOTAL           | 39.4               | 24.7| 20.2|17.7|16.1|13.5 |11.7 |

Maintenance Burden (¢/mi)

| Aircraft       | 19.2               | 12.0| 9.9 | 8.6 | 7.8 | 6.6 | 5.7  |
| Props (Rotors) | -                  | -   | -   | -   | -   | -   | -    |
| Electronics    | 5.3                | 3.7 | 3.2 | 3.0 | 2.8 | 2.5 | 2.4  |
| Airframe Spares| 2.4                | 1.7 | 1.4 | 1.3 | 1.3 | 1.1 | 1.1  |
| Engine Spares  | 6.1                | 4.3 | 3.7 | 3.4 | 3.3 | 2.9 | 2.7  |
| TOTAL          | 43.5               | 30.9| 26.7|24.6|23.3|20.8 |19.6 |

Depreciation Items (¢/mi)

<table>
<thead>
<tr>
<th>(¢/mi)</th>
<th>Total DOC</th>
</tr>
</thead>
<tbody>
<tr>
<td>($/hr)</td>
<td>788.2</td>
</tr>
<tr>
<td>($/av. seat)</td>
<td>0.53</td>
</tr>
<tr>
<td>($/av. seat-mi)</td>
<td>5.28</td>
</tr>
</tbody>
</table>

Stage Length (mi) 10 20 30 40 50 100 200
Fuel Burn (lb) 1580 2971 2248 2381 2515 3069 4087
Block Speed (mph) 190 267 309 335 353 396 421
FIG. F6  CHARACTERISTICS OF THE STOL

80 passengers - 200 mi. design range

Cruise Altitude 20,000 ft.
Cruise Speed 350 mph

<table>
<thead>
<tr>
<th>Structure</th>
<th>Weight (lbs)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Wing</td>
<td>6578</td>
</tr>
<tr>
<td>Fuel System</td>
<td>185</td>
</tr>
<tr>
<td>Flight Controls</td>
<td>1079</td>
</tr>
<tr>
<td>Tail</td>
<td>1888</td>
</tr>
<tr>
<td>Fuselage</td>
<td>6560</td>
</tr>
<tr>
<td>Landing Gear</td>
<td>1618</td>
</tr>
<tr>
<td>Propulsion System</td>
<td></td>
</tr>
<tr>
<td>Engines</td>
<td>1865</td>
</tr>
<tr>
<td>Installation</td>
<td>933</td>
</tr>
<tr>
<td>Propellers</td>
<td>2042</td>
</tr>
<tr>
<td>Transmission</td>
<td>2982</td>
</tr>
<tr>
<td>Navigation Instruments</td>
<td>200</td>
</tr>
<tr>
<td>Hydraulics</td>
<td>570</td>
</tr>
<tr>
<td>Electrical Equipment</td>
<td>645</td>
</tr>
<tr>
<td>Electronics</td>
<td>642</td>
</tr>
<tr>
<td>Furnishings</td>
<td>3750</td>
</tr>
<tr>
<td>Air Conditioning &amp; De-Icing Eqt.</td>
<td>1540</td>
</tr>
<tr>
<td>Weight Empty</td>
<td>33077</td>
</tr>
<tr>
<td>Payload and Crew</td>
<td>16600</td>
</tr>
<tr>
<td>Trapped Oil</td>
<td>140</td>
</tr>
<tr>
<td>Fuel</td>
<td>4138</td>
</tr>
</tbody>
</table>

GROSS WEIGHT 53955

<table>
<thead>
<tr>
<th>Fuel Breakdown</th>
<th>Range (st.mi.)</th>
<th>Time (hr)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Start-up &amp; Taxi</td>
<td>191</td>
<td>--</td>
</tr>
<tr>
<td>T/O, Acc. &amp; Climb</td>
<td>642</td>
<td>24.8</td>
</tr>
<tr>
<td>Cruise</td>
<td>1220</td>
<td>132.4</td>
</tr>
<tr>
<td>Descend &amp; Decel.</td>
<td>131</td>
<td>42.8</td>
</tr>
<tr>
<td>Maneuver</td>
<td>341</td>
<td>--</td>
</tr>
<tr>
<td>Reserves</td>
<td>1613</td>
<td>--</td>
</tr>
<tr>
<td>Fuel Breakdown</td>
<td>4138</td>
<td>200.0</td>
</tr>
</tbody>
</table>

(L/D)_{cruise} = 11.3
**DOC Input Parameters**

- Number of Engines = 4
- Max. Cruise Speed = 350 mph
- HP of One Engine = 2,476 hp (NRP)
- Thrust of One Engine = -
- Weight of One Engine = 466 lb.
- Number of Passengers = 80
- Design Range = 200 mi.
- Propeller (Rotor) Weight = 2,042 lb.
- Empty Weight (less engines) = 31,352 lb.
- Gross Weight = 53,955 lb.
- Annual Utilization = 3000 hr.
- Depreciation Period = 12 yr.
- Production Run (Airframe) = 300
- $/lb engines = $300/lb
- Maintenance Factor = 1.1
- Engine TBO = 4000 hr.

**DOC Output**

<table>
<thead>
<tr>
<th>Stage Length (mi.)</th>
<th>10</th>
<th>20</th>
<th>30</th>
<th>40</th>
<th>50</th>
<th>100</th>
<th>200</th>
</tr>
</thead>
<tbody>
<tr>
<td>Flight Operations ($/mi)</td>
<td>44.8</td>
<td>27.2</td>
<td>21.4</td>
<td>19.0</td>
<td>17.1</td>
<td>14.3</td>
<td>12.0</td>
</tr>
<tr>
<td>Pilot</td>
<td>29.5</td>
<td>17.7</td>
<td>13.8</td>
<td>12.2</td>
<td>10.9</td>
<td>9.1</td>
<td>7.5</td>
</tr>
<tr>
<td>Copilot</td>
<td>157.1</td>
<td>91.0</td>
<td>69.0</td>
<td>54.3</td>
<td>48.1</td>
<td>28.0</td>
<td>22.3</td>
</tr>
<tr>
<td>Fuel</td>
<td>76.8</td>
<td>43.6</td>
<td>32.5</td>
<td>27.9</td>
<td>24.4</td>
<td>19.2</td>
<td>14.8</td>
</tr>
<tr>
<td>Insurance</td>
<td>308.1</td>
<td>179.5</td>
<td>136.7</td>
<td>113.3</td>
<td>100.5</td>
<td>70.6</td>
<td>56.5</td>
</tr>
<tr>
<td>TOTAL</td>
<td>157.2</td>
<td>89.2</td>
<td>66.5</td>
<td>57.0</td>
<td>49.9</td>
<td>39.1</td>
<td>30.1</td>
</tr>
<tr>
<td>Maintenance Burden ($/mi)</td>
<td>66.1</td>
<td>37.5</td>
<td>28.0</td>
<td>24.0</td>
<td>21.0</td>
<td>16.4</td>
<td>12.7</td>
</tr>
<tr>
<td>Depreciation Items ($/mi)</td>
<td>93.0</td>
<td>52.8</td>
<td>39.4</td>
<td>33.8</td>
<td>29.5</td>
<td>23.2</td>
<td>17.8</td>
</tr>
<tr>
<td>Aircraft</td>
<td>28.2</td>
<td>16.0</td>
<td>11.9</td>
<td>10.2</td>
<td>8.9</td>
<td>7.0</td>
<td>5.4</td>
</tr>
<tr>
<td>Engines</td>
<td>7.0</td>
<td>4.0</td>
<td>3.0</td>
<td>2.5</td>
<td>2.2</td>
<td>1.7</td>
<td>1.3</td>
</tr>
<tr>
<td>Props (Rotors)</td>
<td>21.3</td>
<td>12.1</td>
<td>9.0</td>
<td>7.7</td>
<td>6.8</td>
<td>5.3</td>
<td>4.1</td>
</tr>
<tr>
<td>Electronics</td>
<td>10.8</td>
<td>6.1</td>
<td>4.6</td>
<td>3.9</td>
<td>3.4</td>
<td>2.7</td>
<td>2.1</td>
</tr>
<tr>
<td>Airframe Spares</td>
<td>21.1</td>
<td>12.0</td>
<td>8.9</td>
<td>7.7</td>
<td>6.7</td>
<td>5.3</td>
<td>3.4</td>
</tr>
<tr>
<td>Engine Spares</td>
<td>181.4</td>
<td>102.9</td>
<td>76.7</td>
<td>65.8</td>
<td>57.5</td>
<td>45.1</td>
<td>34.7</td>
</tr>
<tr>
<td>TOTAL DOC ($/mi)</td>
<td>712.8</td>
<td>409.1</td>
<td>307.9</td>
<td>260.1</td>
<td>228.8</td>
<td>171.3</td>
<td>134.1</td>
</tr>
<tr>
<td>($/hr)</td>
<td>331.2</td>
<td>332.8</td>
<td>334.1</td>
<td>327.8</td>
<td>328.6</td>
<td>310.4</td>
<td>312.2</td>
</tr>
<tr>
<td>($/av. seat)</td>
<td>0.89</td>
<td>1.02</td>
<td>1.16</td>
<td>1.30</td>
<td>1.43</td>
<td>2.14</td>
<td>3.52</td>
</tr>
<tr>
<td>($/av. seat-mi)</td>
<td>8.91</td>
<td>5.11</td>
<td>3.85</td>
<td>3.25</td>
<td>2.86</td>
<td>2.14</td>
<td>1.68</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Stage Length (mi.)</th>
<th>10</th>
<th>20</th>
<th>30</th>
<th>40</th>
<th>50</th>
<th>100</th>
<th>200</th>
</tr>
</thead>
<tbody>
<tr>
<td>Flight Operations ($/mi)</td>
<td>44.8</td>
<td>27.2</td>
<td>21.4</td>
<td>19.0</td>
<td>17.1</td>
<td>14.3</td>
<td>12.0</td>
</tr>
<tr>
<td>Pilot</td>
<td>29.5</td>
<td>17.7</td>
<td>13.8</td>
<td>12.2</td>
<td>10.9</td>
<td>9.1</td>
<td>7.5</td>
</tr>
<tr>
<td>Copilot</td>
<td>157.1</td>
<td>91.0</td>
<td>69.0</td>
<td>54.3</td>
<td>48.1</td>
<td>28.0</td>
<td>22.3</td>
</tr>
<tr>
<td>Fuel</td>
<td>76.8</td>
<td>43.6</td>
<td>32.5</td>
<td>27.9</td>
<td>24.4</td>
<td>19.2</td>
<td>14.8</td>
</tr>
<tr>
<td>Insurance</td>
<td>308.1</td>
<td>179.5</td>
<td>136.7</td>
<td>113.3</td>
<td>100.5</td>
<td>70.6</td>
<td>56.5</td>
</tr>
<tr>
<td>TOTAL</td>
<td>157.2</td>
<td>89.2</td>
<td>66.5</td>
<td>57.0</td>
<td>49.9</td>
<td>39.1</td>
<td>30.1</td>
</tr>
<tr>
<td>Maintenance Burden ($/mi)</td>
<td>66.1</td>
<td>37.5</td>
<td>28.0</td>
<td>24.0</td>
<td>21.0</td>
<td>16.4</td>
<td>12.7</td>
</tr>
<tr>
<td>Depreciation Items ($/mi)</td>
<td>93.0</td>
<td>52.8</td>
<td>39.4</td>
<td>33.8</td>
<td>29.5</td>
<td>23.2</td>
<td>17.8</td>
</tr>
<tr>
<td>Aircraft</td>
<td>28.2</td>
<td>16.0</td>
<td>11.9</td>
<td>10.2</td>
<td>8.9</td>
<td>7.0</td>
<td>5.4</td>
</tr>
<tr>
<td>Engines</td>
<td>7.0</td>
<td>4.0</td>
<td>3.0</td>
<td>2.5</td>
<td>2.2</td>
<td>1.7</td>
<td>1.3</td>
</tr>
<tr>
<td>Props (Rotors)</td>
<td>21.3</td>
<td>12.1</td>
<td>9.0</td>
<td>7.7</td>
<td>6.8</td>
<td>5.3</td>
<td>4.1</td>
</tr>
<tr>
<td>Electronics</td>
<td>10.8</td>
<td>6.1</td>
<td>4.6</td>
<td>3.9</td>
<td>3.4</td>
<td>2.7</td>
<td>2.1</td>
</tr>
<tr>
<td>Airframe Spares</td>
<td>21.1</td>
<td>12.0</td>
<td>8.9</td>
<td>7.7</td>
<td>6.7</td>
<td>5.3</td>
<td>3.4</td>
</tr>
<tr>
<td>Engine Spares</td>
<td>181.4</td>
<td>102.9</td>
<td>76.7</td>
<td>65.8</td>
<td>57.5</td>
<td>45.1</td>
<td>34.7</td>
</tr>
<tr>
<td>TOTAL DOC ($/mi)</td>
<td>712.8</td>
<td>409.1</td>
<td>307.9</td>
<td>260.1</td>
<td>228.8</td>
<td>171.3</td>
<td>134.1</td>
</tr>
<tr>
<td>($/hr)</td>
<td>331.2</td>
<td>332.8</td>
<td>334.1</td>
<td>327.8</td>
<td>328.6</td>
<td>310.4</td>
<td>312.2</td>
</tr>
<tr>
<td>($/av. seat)</td>
<td>0.89</td>
<td>1.02</td>
<td>1.16</td>
<td>1.30</td>
<td>1.43</td>
<td>2.14</td>
<td>3.52</td>
</tr>
<tr>
<td>($/av. seat-mi)</td>
<td>8.91</td>
<td>5.11</td>
<td>3.85</td>
<td>3.25</td>
<td>2.86</td>
<td>2.14</td>
<td>1.68</td>
</tr>
</tbody>
</table>
80 Passengers

COMPARISON OF AIRCRAFT

Design Range 200 mi
Design Altitude 20,000 ft

DOC (\$ / seat mi)

DISTANCE (MI)

FIGURE F7
COMPARISON OF AIRCRAFT Design

Range
- 400 mi

Altitude
- 20,000 ft

DISTANCE (MI)

FIGURE F8
FIGURE F9  STOL—OFF DESIGN OPERATION

Design Range — 200 miles
Design Altitude — 20,000 ft.
Design Velocity — 350 m.p.h.

80 Passengers
FIGURE F10 TILT WING-OFF DESIGN OPERATION

- Design Range: 200 miles
- Design Altitude: 20,000 ft.
- Design Velocity: 400 m.p.h.
THRUST REQUIRED EXCEEDS 90% NORMAL RATED THRUST

FIGURE F11  JET LIFT-OFF DESIGN OPERATION

Design Range — 200 miles
Design Altitude — 20,000 ft.
Design Velocity — 450 m.p.h.

80 Passengers
STOL
DIRECT OPERATING COST VS SEATS ABREAST

Design Range - 200 mi
Cruise Velocity - 350 mph
Cruise Altitude - 20,000 ft

FIGURE F 12
TILT WING
DIRECT OPERATING COST VS SEATS ABREAST

Design Range - 200 mi
Cruise Velocity - 400 mph
Cruise Altitude - 20,000 ft

FIGURE F13

DOC (\$/seat mi)

NO. OF SEATS ABREAST

20 Passengers
40 Passengers
80 Passengers
120 Passengers
180 Passengers
JET LIFT
DIRECT OPERATING COST VS. SEATS ABREAST

Design Range - 200 mi
Cruise Velocity - 450 mph
Cruise Altitude - 20,000 ft

Figure F14
TILT WING AND PROPELLER

\[ V \cos i_n \]

\[ V \sin i_n \]

\[ i_n - i_w \]

FIGURE F15
FIGURE F16

Propeller disc

V

V cos \theta

V'
$V\sin\theta + V\cos\theta = V'$

$\alpha''$

$\beta''$

$i_n - i_w$

$2v$

$V''$

Prop Axis

Zero Lift Line

$V''$ IS RELATIVE VELOCITY VECTOR TO WING

FIGURE F17
EFFECT OF CORRECTION ON $C_L$

FIGURE F19
80 Passengers

TILT WING - COMPARISON OF TYPES -

Design Range - 200 mi
Design Velocity - 400 mph
Design Altitude - 20,000 ft

4 Engine - 4 Propeller

TILT WING - TILT NACELLE
WL = 109 psf
DL = 50 psf

4 Engine - 4 Propeller

WL = 83.4 psf
DL = 50 psf

FIGURE F20
TILT WING - TILT NACELLE HYBRID

4 Engine - 4 Propellers
80 Passengers
Aspect Ratio 9.5

Power Determined by Take-off Requirements
Wing Based on Optimum Design
Power Determined by Cruise Requirements
Wing Based on Propeller Size

Design Range -200 mi
Design Altitude -20,000 ft
Cruise Velocity -350 mph

DISC LOADING (psf)

FIGURE F21
TILT WING - TILT NACELLE

4 Engines - 4 Propellers

Wing Based on Cruise Optimum
NRP Based on Takeoff
NRP Based on Cruise
Wing Based on Rotor Diameter

HCR = 10,000
HCR = 20,000
HCR = 30,000

Design Range = 200 mi
Cruise Velocity = 350 mph
ETA = .8

FIGURE F22
TILT WING - TILT NACELLE

4 Engines - 4 Propellers

HCR = 20,000
HCR = 30,000
HCR = 10,000

NRP Based on Cruise

NRP Based on Takeoff

DISC LOADING (psf.)

FIGURE F23

Design Range - 200 mi
Cruise Velocity - 400 mph
ETA = .8
TILT WING - TILT NACELLE

Design Range - 200 mi
Cruise Velocity - 450 mph
ETA - .7

4 Engines - 4 Propellers

HCR = 20,000
HCR = 10,000
HCR = 30,000

NRP Based on Takeoff
NRP Based on Cruise

FIGURE F24

DISC LOADING (psf)

DOC ($/seat mi)
3.0

TILT WING - TILT NACELLE

4 Engines - 2 Propellers

Design Range - 200 mi
Cruise Velocity - 350 mph
ETA - .8

FIGURE F25
4 Engines - 2 Propellers  TILT WING - TILT NACELLE

Design Range - 200 mi
Cruise Velocity - 400 mph
ETA - .8

Wing Based on Rotor Diameter
NRP Based on Cruise
Wing Based on Cruise Optimum

HCR = 10,000
HCR = 20,000
HCR = 30,000

DOC ($/seat mi)
DISC LOADING (psf)

FIGURE F26
3.0 Engines - 2 Propellers  
TILT WING - TILT NACELLE

Design Range - 200 mi
Cruise Velocity - 450 mph
ETA = .7

Wing Based on Cruise Optimum

HCR = 10,000
HCR = 20,000
HCR = 30,000

NRP Based on Cruise

Wing Based on Rotor Diameter

NRP Based on Takeoff

FIGURE F27
80 Passengers

OPTIMUM HYBRID TILT WINGS

- COMPARISON -

Design Range - 200 mi
Cruise Velocity - 400 mph

DOC (\$/seat mi)

4 Engine - 4 Propeller Hybrid; \( h = 20,000 \) ft

4 Engine - 2 Propeller Hybrid; \( h = 10,000 \) ft

FIGURE F28
Fig. F30  CHARACTERISTICS OF THE TWO PROPELLER HYBRID TILT WING AIRCRAFT

80 Passenger - 200 mile design range
Disc loading = .15 psf
Cruise Altitude - 10,000 ft.
Cruise Speed - 400 MPH
6 Seats Abreast

<table>
<thead>
<tr>
<th>Structure</th>
<th>Weight (lbs)</th>
<th>Aircraft Characteristics</th>
</tr>
</thead>
<tbody>
<tr>
<td>Wing</td>
<td>4,128</td>
<td>Wing:</td>
</tr>
<tr>
<td>Fuel System</td>
<td>173</td>
<td>Span = 59.4</td>
</tr>
<tr>
<td>Flight Control</td>
<td>1,121</td>
<td>Aspect Ratio = 9.5</td>
</tr>
<tr>
<td>Tail</td>
<td>1,401</td>
<td>Area = 372</td>
</tr>
<tr>
<td>Fuselage</td>
<td>6,597</td>
<td>Wing Loading = 150.9</td>
</tr>
<tr>
<td>Landing Gear</td>
<td>1,682</td>
<td>Taper Ratio = .5</td>
</tr>
<tr>
<td>Propulsion System:</td>
<td></td>
<td>Sweepback angle = 0</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Engines</td>
<td>2,020</td>
<td>Fuselage:</td>
</tr>
<tr>
<td>Installation</td>
<td>1,010</td>
<td>Length = 77.2 ft.</td>
</tr>
<tr>
<td>Propellers</td>
<td>3,214</td>
<td>Diameter = 12.6 ft.</td>
</tr>
<tr>
<td>Transmission</td>
<td>6,722</td>
<td>Seats Abreast = 6</td>
</tr>
<tr>
<td>Navigation Instruments</td>
<td>200</td>
<td></td>
</tr>
<tr>
<td>Hydraulics</td>
<td>599</td>
<td></td>
</tr>
<tr>
<td>Electronic Equipment</td>
<td>659</td>
<td></td>
</tr>
<tr>
<td>Electronics</td>
<td>642</td>
<td></td>
</tr>
<tr>
<td>Furnishings</td>
<td>3,750</td>
<td>Engines:</td>
</tr>
<tr>
<td>Air Conditioning &amp; de-icing eqt.</td>
<td>1,540</td>
<td>Propellers:</td>
</tr>
<tr>
<td></td>
<td></td>
<td>Disc loading = 15.0 psf</td>
</tr>
<tr>
<td></td>
<td></td>
<td>Diameter = 48.8 ft.</td>
</tr>
<tr>
<td></td>
<td></td>
<td>Solidity = .0749</td>
</tr>
<tr>
<td></td>
<td></td>
<td>2 propellers</td>
</tr>
</tbody>
</table>

Weight Empty 35,458
Payload & Crew 16,600
Trapped Oil 140
Fuel 3,853
Gross Weight 56,051

Fuel Breakdown:

<table>
<thead>
<tr>
<th>Fuel Type</th>
<th>lbs</th>
<th>Distance (mi)</th>
<th>Time (hr)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Hover &amp; Warmup</td>
<td>76</td>
<td>14.1</td>
<td>.0033</td>
</tr>
<tr>
<td>Acceleration &amp; Climb</td>
<td>370</td>
<td>165.3</td>
<td>.0680</td>
</tr>
<tr>
<td>Cruise</td>
<td>1,829</td>
<td>20.6</td>
<td>.4133</td>
</tr>
<tr>
<td>Descent &amp; Decel.</td>
<td>118</td>
<td></td>
<td>.0599</td>
</tr>
<tr>
<td>Reserves</td>
<td>1,460</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

TOTAL 3,853 200.0 .5453
Fig. F31 CHARACTERISTICS OF THE 4 PROPELLER HYBRID TILT WING AIRCRAFT

80 Passenger - 200 mile design range
Disc Loading = 30 psf
Cruise altitude = 20,000 ft.
Cruise speed = 400 MPH
6 seats abreast

<table>
<thead>
<tr>
<th>Structure</th>
<th>Weight (lbs)</th>
<th>Aircraft Characteristics</th>
</tr>
</thead>
<tbody>
<tr>
<td>Wing</td>
<td>5,665</td>
<td>Wing:</td>
</tr>
<tr>
<td>Fuel System</td>
<td>171</td>
<td>Span = 76.6 ft.</td>
</tr>
<tr>
<td>Flight Control</td>
<td>1,144</td>
<td>Aspect Ratio = 9.5</td>
</tr>
<tr>
<td>Tail</td>
<td>1,430</td>
<td>Area = 618. sq. ft.</td>
</tr>
<tr>
<td>Fuselage</td>
<td>6,618</td>
<td>Wing Loading = 92.6 psf</td>
</tr>
<tr>
<td>Landing Gear</td>
<td>1,716</td>
<td>Taper Ratio = .5</td>
</tr>
<tr>
<td>Propulsion System</td>
<td></td>
<td>Sweepback Angle = 0</td>
</tr>
<tr>
<td>Engines</td>
<td>2,801</td>
<td>Fuselage:</td>
</tr>
<tr>
<td>Installation</td>
<td>1,401</td>
<td>Length = 77.2 ft.</td>
</tr>
<tr>
<td>Propellers</td>
<td>3,231</td>
<td>Diameter = 12.6 ft.</td>
</tr>
<tr>
<td>Transmissions</td>
<td>5,055</td>
<td>Seats Abreast = 6</td>
</tr>
<tr>
<td>Navigation Instruments</td>
<td>200</td>
<td>Engine:</td>
</tr>
<tr>
<td>Hydraulics</td>
<td>615</td>
<td>4 at 4,460 HP each - 30 min. rating</td>
</tr>
<tr>
<td>Electronic Equipment</td>
<td>666</td>
<td>Propeller:</td>
</tr>
<tr>
<td>Electronics</td>
<td>642</td>
<td>Disc loading = 30 psf</td>
</tr>
<tr>
<td>Furnishings</td>
<td>3,750</td>
<td>Diameter = 24.6 ft.</td>
</tr>
<tr>
<td>Air Conditioning &amp; de-icing eqt.</td>
<td>1,540</td>
<td>Solidity = .150</td>
</tr>
<tr>
<td></td>
<td>3,819</td>
<td>4 propellers</td>
</tr>
<tr>
<td>Weight Empty</td>
<td>36,646</td>
<td></td>
</tr>
<tr>
<td>Payload &amp; Crew</td>
<td>16,600</td>
<td></td>
</tr>
<tr>
<td>Trapped Oil</td>
<td>140</td>
<td></td>
</tr>
<tr>
<td>Fuel</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Gross Weight</td>
<td>57,205</td>
<td>Fuel Breakdown</td>
</tr>
<tr>
<td></td>
<td></td>
<td>Hover &amp; Warmup</td>
</tr>
<tr>
<td></td>
<td></td>
<td>105</td>
</tr>
<tr>
<td></td>
<td></td>
<td>Acceleration &amp; Climb</td>
</tr>
<tr>
<td></td>
<td></td>
<td>645</td>
</tr>
<tr>
<td></td>
<td></td>
<td>Cruise</td>
</tr>
<tr>
<td></td>
<td></td>
<td>1453</td>
</tr>
<tr>
<td></td>
<td></td>
<td>Descent &amp; Deceleration</td>
</tr>
<tr>
<td></td>
<td></td>
<td>209</td>
</tr>
<tr>
<td></td>
<td></td>
<td>Reserves</td>
</tr>
<tr>
<td></td>
<td></td>
<td>1407</td>
</tr>
<tr>
<td></td>
<td></td>
<td>TOTAL</td>
</tr>
<tr>
<td></td>
<td></td>
<td>3819</td>
</tr>
<tr>
<td></td>
<td></td>
<td>200.0</td>
</tr>
<tr>
<td></td>
<td></td>
<td>.5618</td>
</tr>
</tbody>
</table>
80 Passengers

OPTIMUM TILT WINGS

-COMPARISON-

Cruise Velocity - 400 mph
Design Range - 200 miles

FIGURE F29
HYBRID TILT WING - TILT NACELLE

Cruise Velocity - 400 mph
Cruise Altitude - 20,000 ft

4 Engines - 4 Propellers
80 Passengers

DISC LOADING (psf)

DOC (\$/seat mi)

\[ A_R = 11.5 \]
\[ A_R = 9.5 \]
\[ A_R = 6.0 \]

FIGURE F32
HYBRID TILT
WING - TILT NACELLE

4 Engines - 2 Props
80 Passengers

Cruise Velocity - 400 mph
Cruise Altitude - 20,000 ft

DOC ($/seat mi)

DISC LOADING (psf)

FIGURE F33
80 Passengers  CONVENTIONAL TILT WING

Design Range = 200 mi
Cruise Velocity = 400 mph
Cruise Altitude = 20,000 ft

FIGURE F 34
Fig. F35  FOUR ENGINES – FOUR PROPELLER CONVENTIONAL TILT WING

\[ V_{cr} = 400 \text{ mph} \quad DL = 50 \text{ psf} \quad \text{Utilization} = 3000 \text{ hrs/yr} \]

Design range = 200 miles, 80 passengers

<table>
<thead>
<tr>
<th>No.</th>
<th>Description</th>
<th>Value</th>
<th>Value</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>Cruise Altitude (ft)</td>
<td>10,000</td>
<td>20,000</td>
<td>30,000</td>
</tr>
<tr>
<td>2</td>
<td>DOC (C/s-m)</td>
<td>1.748</td>
<td>1.689</td>
<td>1.679</td>
</tr>
<tr>
<td>3</td>
<td>Gross weight (lb)</td>
<td>63,360</td>
<td>61,126</td>
<td>59,974</td>
</tr>
<tr>
<td>4</td>
<td>Fuel burned (lb)</td>
<td>3367</td>
<td>2713</td>
<td>2364</td>
</tr>
<tr>
<td>5</td>
<td>Wing area (ft(^2))</td>
<td>760</td>
<td>733</td>
<td>781</td>
</tr>
<tr>
<td>6</td>
<td>Wing weight (lbs)</td>
<td>6782.</td>
<td>6507.</td>
<td>6692.</td>
</tr>
<tr>
<td>7</td>
<td>Wing loading (psf)</td>
<td>83.4</td>
<td>83.4</td>
<td>76.8</td>
</tr>
<tr>
<td>8</td>
<td>Total NRP (hp)</td>
<td>20,824</td>
<td>20,089</td>
<td>18,973</td>
</tr>
<tr>
<td>9</td>
<td>Total Flight Cost ($)</td>
<td>279.6</td>
<td>270.2</td>
<td>268.6</td>
</tr>
<tr>
<td>10</td>
<td>Total Fuel Cost ($)</td>
<td>58.5</td>
<td>47.2</td>
<td>41.1</td>
</tr>
<tr>
<td>11</td>
<td>Total Flight Cost less Fuel ($)</td>
<td>221.1</td>
<td>223.0</td>
<td>227.5</td>
</tr>
<tr>
<td>12</td>
<td>Total cost ($/hr)</td>
<td>526.</td>
<td>491.</td>
<td>465.</td>
</tr>
<tr>
<td>13</td>
<td>Total cost less Fuel ($/hr)</td>
<td>416.</td>
<td>405.5</td>
<td>394.</td>
</tr>
<tr>
<td>14</td>
<td>Total cost less Fuel (C/s-m)</td>
<td>1.382</td>
<td>1.394</td>
<td>1.422</td>
</tr>
<tr>
<td>15</td>
<td>Fuel cost (C/s-m)</td>
<td>.366</td>
<td>.295</td>
<td>.257</td>
</tr>
<tr>
<td>16</td>
<td>Block speed (mph)</td>
<td>377.</td>
<td>364.</td>
<td>347</td>
</tr>
<tr>
<td>17</td>
<td>Block time (hrs)</td>
<td>.531</td>
<td>.550</td>
<td>.577</td>
</tr>
<tr>
<td>18</td>
<td>Time to climb (hrs)</td>
<td>.0239</td>
<td>.0604</td>
<td>.1163</td>
</tr>
<tr>
<td>19</td>
<td>Time to climb/block time</td>
<td>.0450</td>
<td>.110</td>
<td>.202</td>
</tr>
<tr>
<td>20</td>
<td>Average forward velocity during climb (mph)</td>
<td>172</td>
<td>186</td>
<td>194</td>
</tr>
<tr>
<td>21</td>
<td>Average rate of climb (fpm)</td>
<td>5939</td>
<td>5108</td>
<td>4084</td>
</tr>
<tr>
<td>22</td>
<td>Wing design based on: disc loading to wing loading ratio for all altitudes</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>23</td>
<td>Cruise ( C_L )</td>
<td>.276</td>
<td>.383</td>
<td>.500</td>
</tr>
<tr>
<td>No.</td>
<td>Description</td>
<td>Value 1</td>
<td>Value 2</td>
<td>Value 3</td>
</tr>
<tr>
<td>-----</td>
<td>------------------------------------</td>
<td>---------</td>
<td>---------</td>
<td>---------</td>
</tr>
<tr>
<td>1.</td>
<td>Cruise altitude (ft)</td>
<td>10,000</td>
<td>20,000</td>
<td>30,000</td>
</tr>
<tr>
<td>2.</td>
<td>Doc ($/s-m)</td>
<td>1.527</td>
<td>1.502</td>
<td>1.558</td>
</tr>
<tr>
<td>3.</td>
<td>Gross weight (lb)</td>
<td>58,922</td>
<td>57,205</td>
<td>57,492</td>
</tr>
<tr>
<td>4.</td>
<td>Fuel burned (lb)</td>
<td>2945.8</td>
<td>2426.0</td>
<td>2148.2</td>
</tr>
<tr>
<td>5.</td>
<td>Wing area (ft²)</td>
<td>633</td>
<td>618</td>
<td>752</td>
</tr>
<tr>
<td>6.</td>
<td>Wing weight (lb)</td>
<td>5843.</td>
<td>5665.</td>
<td>6390.</td>
</tr>
<tr>
<td>7.</td>
<td>Wing loading (psf)</td>
<td>93.0</td>
<td>92.6</td>
<td>76.5</td>
</tr>
<tr>
<td>8.</td>
<td>Total NRP (hp)</td>
<td>15323</td>
<td>14880</td>
<td>14997</td>
</tr>
<tr>
<td>9.</td>
<td>Total flight cost ($)</td>
<td>244.2</td>
<td>240.2</td>
<td>249.2</td>
</tr>
<tr>
<td>10.</td>
<td>Total fuel cost ($)</td>
<td>51.2</td>
<td>42.2</td>
<td>37.4</td>
</tr>
<tr>
<td>11.</td>
<td>Total flight cost less fuel ($)</td>
<td>193.0</td>
<td>198.0</td>
<td>211.8</td>
</tr>
<tr>
<td>12.</td>
<td>Total cost ($/hr)</td>
<td>455.5</td>
<td>427.</td>
<td>418.</td>
</tr>
<tr>
<td>13.</td>
<td>Total cost less fuel ($/hr)</td>
<td>360</td>
<td>352</td>
<td>355</td>
</tr>
<tr>
<td>14.</td>
<td>Total cost less fuel ($/s-m)</td>
<td>1.206</td>
<td>1.237</td>
<td>1.323</td>
</tr>
<tr>
<td>15.</td>
<td>Fuel cost ($/s-m)</td>
<td>.320</td>
<td>.264</td>
<td>.234</td>
</tr>
<tr>
<td>16.</td>
<td>Block speed (mph)</td>
<td>373</td>
<td>356</td>
<td>335</td>
</tr>
<tr>
<td>17.</td>
<td>Block time (hrs)</td>
<td>.536</td>
<td>.562</td>
<td>.596</td>
</tr>
<tr>
<td>18.</td>
<td>Time to climb (hrs)</td>
<td>.0324</td>
<td>.0836</td>
<td>.1512</td>
</tr>
<tr>
<td>19.</td>
<td>Time to climb/block time</td>
<td>.0604</td>
<td>.1488</td>
<td>.254</td>
</tr>
<tr>
<td>20.</td>
<td>Average forward velocity during climb (mph)</td>
<td>175</td>
<td>188</td>
<td>193</td>
</tr>
<tr>
<td>21.</td>
<td>Average rate of climb (fpm)</td>
<td>4376</td>
<td>3688</td>
<td>3141</td>
</tr>
<tr>
<td>23.</td>
<td>Cruise $C_L$</td>
<td>.308</td>
<td>.425</td>
<td>.500</td>
</tr>
</tbody>
</table>

Design range = 200 miles; 80 passengers
Fig. F37  FOUR ENGINE - TWO PROPELLER HYBRID TILT WING

\[ V_{cr} = 400 \text{ mph} \]
\[ DL = 15 \text{ psf} \]
\[ \text{Utilization} = 3000 \text{ hrs/yr} \]
Design range = 200; 80 passengers. Cruise \( C_L = .50 \) at all cruise altitudes.

<table>
<thead>
<tr>
<th>Cruise altitude (ft)</th>
<th>10,000</th>
<th>20,000</th>
<th>30,000</th>
</tr>
</thead>
<tbody>
<tr>
<td>DOC ($/s-m)</td>
<td>1.347</td>
<td>1.466</td>
<td>1.741</td>
</tr>
<tr>
<td>Gross weight (lb)</td>
<td>56,051</td>
<td>58,423</td>
<td>65,612</td>
</tr>
<tr>
<td>Fuel burned (lb)</td>
<td>2402</td>
<td>2247</td>
<td>2332</td>
</tr>
<tr>
<td>Wing area (ft(^2))</td>
<td>372.</td>
<td>537.</td>
<td>858.</td>
</tr>
<tr>
<td>Wing weight (lbs)</td>
<td>4,128</td>
<td>5,265</td>
<td>7,437</td>
</tr>
<tr>
<td>Wing loading (psf)</td>
<td>151.</td>
<td>109.</td>
<td>76.5</td>
</tr>
<tr>
<td>Total NRP (hp)</td>
<td>10,732</td>
<td>12,064</td>
<td>16,324</td>
</tr>
<tr>
<td>Total flight cost ($)</td>
<td>215.4</td>
<td>234.4</td>
<td>278.6</td>
</tr>
<tr>
<td>Total fuel cost ($)</td>
<td>41.8</td>
<td>39.1</td>
<td>40.5</td>
</tr>
<tr>
<td>Total flight cost less fuel ($)</td>
<td>173.6</td>
<td>195.3</td>
<td>238.1</td>
</tr>
<tr>
<td>Total cost ($/hr)</td>
<td>395.</td>
<td>406.</td>
<td>465.</td>
</tr>
<tr>
<td>Total cost less fuel ($/hr)</td>
<td>318.</td>
<td>338.</td>
<td>398.</td>
</tr>
<tr>
<td>Total cost less fuel ($/s-m)</td>
<td>1.085</td>
<td>1.222</td>
<td>1.488</td>
</tr>
<tr>
<td>Fuel cost ($/s-m)</td>
<td>.261</td>
<td>.244</td>
<td>.253</td>
</tr>
<tr>
<td>Block speed (mph)</td>
<td>367</td>
<td>346</td>
<td>334</td>
</tr>
<tr>
<td>Block time (hrs)</td>
<td>.545</td>
<td>.578</td>
<td>.599</td>
</tr>
<tr>
<td>Time to climb (hrs)</td>
<td>.0571</td>
<td>.1218</td>
<td>.1603</td>
</tr>
<tr>
<td>Time to climb/block time</td>
<td>.105</td>
<td>.211</td>
<td>.268</td>
</tr>
<tr>
<td>Average forward velocity during climb (mph)</td>
<td>205.</td>
<td>200.</td>
<td>196.</td>
</tr>
<tr>
<td>Average rate of climb (fpm)</td>
<td>2482.</td>
<td>2531.</td>
<td>2963.</td>
</tr>
<tr>
<td>Wing design based on:</td>
<td>cruise</td>
<td>cruise</td>
<td>cruise</td>
</tr>
<tr>
<td>Cruise ( C_L )</td>
<td>.500</td>
<td>.500</td>
<td>.500</td>
</tr>
</tbody>
</table>
FIGURE F38 COMPARISON OF PROPELLER AIRCRAFT TYPES

Design Range — 200 miles
Cruise Altitude — 20,000 ft.

80 Passengers
FIGURE F39 MULTIPLE HOP CAPABILITIES

Design Range — 200 miles

80 Passengers

HOP DISTANCE — miles
FIGURE F40  MULTIPLE HOPS CAPABILITIES

Design Range — 400 miles

Passengers

TILT WING
STOL
JET LIFT
STOWED ROTOR TRANSITION SPEED ($V_{TR}$) VARIATION

$V_{TR} = 125$ mph

$V_{TR} = 160$ mph

$V_{TR} = 185$ mph

FIGURE RI
FIGURE R3

COMPOUND ROTOR SOLIDITY (σ) VARIATION

DOC (q/seat mi)

STAGE LENGTH (MI)

σ = 0.2
σ = 0.125
σ = 0.15
STOWED ROTOR SOLIDITY ($\sigma$) VARIATION

STAGE LENGTH (MI)

FIGURE R4
FIGURE R5

HELIPECCTOR ROTOR SOLIDITY (σ) VARIATION

DOC ($/seat mi$)

0 25 50 75 100 150 200
STAGE LENGTH (MI)

σ = .10 & .135

σ = .115
STOWED ROTOR TRANSITION TIME ($t_{TR}$) VARIATION

FIGURE R6

- $t_{TR} = 2$ min
- $t_{TR} = 1$ min

DOC ($\$/seat mi$)

STAGE LENGTH (MI)
Figure R7 shows the rotor weight penalty for stowing ($K_{ST}$) as a function of stage length (MI) in dollars per seat mile ($DOC$). The graph includes curves for $K_{ST} = 1.2$, $K_{ST} = 1.5$, and $K_{ST} = 2.0$. As stage length increases, the $DOC$ decreases for all $K_{ST}$ values.
HELIQUOTER SEAT PITCH (SP) & SEAT WIDTH (SW) VARIATION

STAGE LENGTH (MI)

FIGURE R8

DOC ($ / seat mi)

SP = 36"  SW = 24"
SP = 34"  SW = 19"
COMPOUND CRUISE SPEED VARIATION

\[ \mu = 0.55 \text{ (263 mph)} \]

\[ \mu = 0.65 \text{ (292 mph)} \]

STAGE LENGTH (MI)

FIGURE R9
STOWED ROTOR CRUISE SPEED VARIATION

DOC ($/seat mi)

400 mph
500 mph
450 mph

STAGE LENGTH (MI)

FIGURE R10
HELCOPTER CRUISE SPEED VARIATION

\[ \mu = 0.40 \text{ (212 mph)} \]

\[ \mu = 0.45 \text{ (230 mph)} \]

FIGURE RII

STAGE LENGTH (MI)

DOC ($/\text{seat mi}$)
COMPOUND CRUISE ALTITUDE VARIATION

| 1500 ft | 3.0 |
| 5000 ft | 2.5 |

Figure R12
HELICOPTER CRUISE ALTITUDE VARIATION

STAGE LENGTH (MI)

DOC ($/seat mi.)

FIGURE R14
COMPARISON OF SELECTED AIRCRAFT

STOL
STOWED ROTOR
HELICOPTER
HYBRID TILT WING
JET LIFT

FIGURE R16
Fig. R17 CHARACTERISTICS OF A TYPICAL COMPOUND

80 Passengers - 200 mi. design range
Cruise Altitude 1500 ft., Speed 292 mph

<table>
<thead>
<tr>
<th>Structure</th>
<th>Weight (lbs)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Wing</td>
<td>1255</td>
</tr>
<tr>
<td>Fuel System</td>
<td>471</td>
</tr>
<tr>
<td>Flight Control</td>
<td>1508</td>
</tr>
<tr>
<td>Tail System (including tail</td>
<td>2946</td>
</tr>
<tr>
<td>rotor)</td>
<td></td>
</tr>
<tr>
<td>Fuselage</td>
<td>7498</td>
</tr>
<tr>
<td>Landing Gear</td>
<td>1784</td>
</tr>
<tr>
<td>Propulsion System</td>
<td></td>
</tr>
<tr>
<td>Engine</td>
<td>2204</td>
</tr>
<tr>
<td>Installation</td>
<td>1102</td>
</tr>
<tr>
<td>Rotor System</td>
<td>4796</td>
</tr>
<tr>
<td>Transmission</td>
<td>5415</td>
</tr>
<tr>
<td>Instrument</td>
<td>200</td>
</tr>
<tr>
<td>Electrical &amp; Hydraulic Equipment</td>
<td>1326</td>
</tr>
<tr>
<td>Electronics</td>
<td>642</td>
</tr>
<tr>
<td>Furnishings</td>
<td>3750</td>
</tr>
<tr>
<td>Aircraft Equivalent Flat Plate</td>
<td>1540</td>
</tr>
<tr>
<td>Area (sq. ft.)</td>
<td></td>
</tr>
<tr>
<td>Weight Empty</td>
<td>35336</td>
</tr>
<tr>
<td>Payload &amp; Crew</td>
<td>16600</td>
</tr>
<tr>
<td>Fuel and Oil</td>
<td>6422</td>
</tr>
<tr>
<td><strong>GROSS WEIGHT</strong></td>
<td>59460</td>
</tr>
</tbody>
</table>

Aircraft Equivalent Flat Plate Area $F = 27.5$ sq. ft.  
Aircraft Equivalent L/D in Cruise = 5.3

### Aircraft Characteristics

**Wing:**
- Aspect Ratio = 8
- Area = 154 sq. ft.
- Wing Loading = 197 psf.

**Fuselage:**
- Length = 79.0 ft.
- Diameter = 11.0 ft.
- Seats Abreast = 5

**Engines:**
- 4 at 4140 hp each
- 30 minute rating

**Rotors (1):**
- Disc loading = 14.6 psf
- Diameter = 72.2 ft.
- Solidity = 0.15
- Tip Speed (ft/sec) = 690 (hover), 659 (cruise)

**Fuel Breakdown:**

<table>
<thead>
<tr>
<th>Fuel (lbs.)</th>
<th>Range (mi)</th>
<th>Time (hr)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Hover</td>
<td>20</td>
<td>-0.003</td>
</tr>
<tr>
<td>Transition</td>
<td>-</td>
<td>-</td>
</tr>
<tr>
<td>Acceleration</td>
<td>311</td>
<td>4.0</td>
</tr>
<tr>
<td>Climb</td>
<td>46</td>
<td>1.2</td>
</tr>
<tr>
<td>Cruise</td>
<td>3827</td>
<td>190.6</td>
</tr>
<tr>
<td>Descent &amp; Deceler.</td>
<td>45</td>
<td>4.2</td>
</tr>
<tr>
<td>Reserves</td>
<td>2033</td>
<td>-</td>
</tr>
</tbody>
</table>

**TOTALS** | 6282 | 200.0 | 0.715
**Fig. R18 CHARACTERISTICS OF A TYPICAL STOWED ROTOR A/C**

80 Passengers - 200 mi. Design Range
Cruise Altitude 5000 ft., Speed 450 MPH

<table>
<thead>
<tr>
<th>Structure</th>
<th>Weight (lbs)</th>
<th>Aircraft Characteristics</th>
</tr>
</thead>
<tbody>
<tr>
<td>Wing</td>
<td>6010</td>
<td></td>
</tr>
<tr>
<td>Fuel System</td>
<td>479</td>
<td></td>
</tr>
<tr>
<td>Flight Control</td>
<td>1786</td>
<td></td>
</tr>
<tr>
<td>Tail System (including tail rotor)</td>
<td>2257</td>
<td></td>
</tr>
<tr>
<td>Fuselage</td>
<td>10512</td>
<td></td>
</tr>
<tr>
<td>Landing Gear</td>
<td>2221</td>
<td></td>
</tr>
<tr>
<td>Propulsion System</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Engines</td>
<td>2601</td>
<td></td>
</tr>
<tr>
<td>Installation</td>
<td>1300</td>
<td></td>
</tr>
<tr>
<td>Rotor System</td>
<td>8823</td>
<td></td>
</tr>
<tr>
<td>Transmission</td>
<td>7179</td>
<td></td>
</tr>
<tr>
<td>Instruments</td>
<td>200</td>
<td></td>
</tr>
<tr>
<td>Electrical &amp; Hydraulic Eqpt</td>
<td>1622</td>
<td></td>
</tr>
<tr>
<td>Electronics</td>
<td>642</td>
<td></td>
</tr>
<tr>
<td>Furnishings</td>
<td>3750</td>
<td></td>
</tr>
<tr>
<td>Air Conditioning &amp; De-icing Equipment</td>
<td>1540</td>
<td></td>
</tr>
<tr>
<td>Weight Empty</td>
<td>49624</td>
<td></td>
</tr>
<tr>
<td>Payload &amp; Crew</td>
<td>16600</td>
<td></td>
</tr>
<tr>
<td>Fuel and Oil</td>
<td>6531</td>
<td></td>
</tr>
<tr>
<td><strong>GROSS WEIGHT</strong></td>
<td><strong>74056</strong></td>
<td></td>
</tr>
</tbody>
</table>

Aircraft Equivalent Flat Plate Area Cruise F = 20.9 sq. ft.

Aircraft Equivalent L/D in Cruise = 7.8

<table>
<thead>
<tr>
<th>Fuel Breakdown:</th>
<th>Fuel (lbs)</th>
<th>Range (mi)</th>
<th>Time (hr)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Hover</td>
<td>15</td>
<td>-</td>
<td>.003</td>
</tr>
<tr>
<td>Transition</td>
<td>90</td>
<td>2.7</td>
<td>.020</td>
</tr>
<tr>
<td>Acceleration</td>
<td>446</td>
<td>15.8</td>
<td>.050</td>
</tr>
<tr>
<td>Climb</td>
<td>196</td>
<td>6.0</td>
<td>.021</td>
</tr>
<tr>
<td>Cruise</td>
<td>2869</td>
<td>163.5</td>
<td>.363</td>
</tr>
<tr>
<td>Descent &amp; Deceleration</td>
<td>71</td>
<td>12.1</td>
<td>.041</td>
</tr>
<tr>
<td>Reserves</td>
<td>2074</td>
<td>-</td>
<td>-</td>
</tr>
<tr>
<td><strong>TOTALS</strong></td>
<td><strong>6391</strong></td>
<td><strong>200.0</strong></td>
<td><strong>.495</strong></td>
</tr>
</tbody>
</table>
**Fig. R19** CHARACTERISTICS OF A TYPICAL HELICOPTER

80 Passengers = 200 mi. Design Range
Cruise Altitude 1500 ft., Speed 230 mph

<table>
<thead>
<tr>
<th>Structure</th>
<th>Weight (lbs)</th>
<th>Aircraft Characteristics</th>
</tr>
</thead>
<tbody>
<tr>
<td>Wing</td>
<td>-</td>
<td>Wing:</td>
</tr>
<tr>
<td>Fuel System</td>
<td>283</td>
<td>Aspect Ratio = -</td>
</tr>
<tr>
<td>Flight Control</td>
<td>1329</td>
<td>Area = - sq. ft.</td>
</tr>
<tr>
<td>Tail System (including tail rotor)</td>
<td>4795</td>
<td>Wing Loading = psf</td>
</tr>
<tr>
<td>Fuselage</td>
<td>8778</td>
<td></td>
</tr>
<tr>
<td>Landing Gear</td>
<td>1514</td>
<td></td>
</tr>
<tr>
<td>Propulsion System</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Engines</td>
<td>1135</td>
<td>Fuselage:</td>
</tr>
<tr>
<td>Installation</td>
<td>568</td>
<td>Length = 77.2 ft.</td>
</tr>
<tr>
<td>Rotor System</td>
<td>4254</td>
<td>Diameter = 11.0 ft.</td>
</tr>
<tr>
<td>Transmission</td>
<td>200</td>
<td>Seats Abreast = 5</td>
</tr>
<tr>
<td>Instruments</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Electrical &amp; Hydraulic Equipment</td>
<td>1144</td>
<td>Engines = 4 at 2130 hp each</td>
</tr>
<tr>
<td></td>
<td></td>
<td>30 minute rating</td>
</tr>
<tr>
<td>Weight Empty</td>
<td>29365</td>
<td></td>
</tr>
<tr>
<td>Payload &amp; Crew</td>
<td>16600</td>
<td>Rotors(2):</td>
</tr>
<tr>
<td>Fuel and Oil</td>
<td>3910</td>
<td>Disc Loading = 8.1 psf</td>
</tr>
<tr>
<td></td>
<td></td>
<td>Diameter = 63.2 ft.</td>
</tr>
<tr>
<td></td>
<td></td>
<td>Solidity = 0.115</td>
</tr>
<tr>
<td></td>
<td></td>
<td>Tip Speed (Ft/sec) = 586 (hover), 750 (cruise)</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Aircraft Equivalent Flat Plate Area F= 21.6 sq.ft.</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Aircraft Equivalent L/D in Cruise = 6.3</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Fuel Breakdown:</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Fuel (lbs)</td>
<td>10</td>
<td>Range (mi)</td>
</tr>
<tr>
<td>Time (hr)</td>
<td>.003</td>
<td>200.0</td>
</tr>
<tr>
<td></td>
<td>.021</td>
<td>.894</td>
</tr>
<tr>
<td></td>
<td>.009</td>
<td></td>
</tr>
<tr>
<td></td>
<td>.839</td>
<td></td>
</tr>
<tr>
<td></td>
<td>.023</td>
<td></td>
</tr>
<tr>
<td>TOTALS</td>
<td>3770</td>
<td></td>
</tr>
</tbody>
</table>

---

*Source: U.S. Air Force, 1960*
**Fig. R20  TYPICAL COMPOUND**

### DOC Input Parameters

- Number of Engines = 4
- Max. Cruise Speed = 292 mph
- HP of One Engine = 3443 hp (NRP)
- Weight of One Engine = 551 lb.
- Number of Passengers = 80
- Design Range = 200 mi.
- Propeller (Rotor) Weight = 4796 lb.
- Empty Weight (less engines) = 34234 lb
- Gross Weight = 59460 lb.
- Annual Utilization = 3000 hr.
- Depreciation Period = 12 yr.
- Production Run (Airframe) = 300
- $/lb engines = $300/lb
- Maintenance Factor = 1.3
- Engine TBO = 4000 hr.

### DOC Output

<table>
<thead>
<tr>
<th>Stage Length (mi)</th>
<th>10</th>
<th>20</th>
<th>30</th>
<th>40</th>
<th>50</th>
<th>100</th>
<th>200</th>
</tr>
</thead>
<tbody>
<tr>
<td>Flight Operations (¢/mi)</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Pilot</td>
<td>16.0</td>
<td>13.4</td>
<td>12.5</td>
<td>12.1</td>
<td>11.8</td>
<td>11.3</td>
<td>11.1</td>
</tr>
<tr>
<td>Copilot</td>
<td>10.1</td>
<td>8.4</td>
<td>7.8</td>
<td>7.5</td>
<td>7.4</td>
<td>7.0</td>
<td>6.9</td>
</tr>
<tr>
<td>Fuel</td>
<td>76.9</td>
<td>56.3</td>
<td>49.4</td>
<td>46.0</td>
<td>43.9</td>
<td>39.8</td>
<td>37.7</td>
</tr>
<tr>
<td>Insurance</td>
<td>25.8</td>
<td>19.8</td>
<td>17.8</td>
<td>16.8</td>
<td>16.2</td>
<td>15.0</td>
<td>14.4</td>
</tr>
<tr>
<td>TOTAL</td>
<td>128.8</td>
<td>97.8</td>
<td>87.5</td>
<td>82.4</td>
<td>79.3</td>
<td>73.1</td>
<td>70.0</td>
</tr>
</tbody>
</table>

| Direct Maintenance (¢/mi) |
| Labor Airframe | 13.7 | 10.5 | 9.4 | 8.9 | 8.6 | 7.9 | 7.6 |
| Materials Airframe | 18.0 | 13.8 | 12.4 | 11.7 | 11.3 | 10.4 | 10.0 |
| Materials Engines | 27.8 | 21.3 | 19.1 | 18.1 | 17.4 | 16.1 | 15.5 |
| Labor Engines | 3.0 | 2.3 | 2.0 | 1.9 | 1.9 | 1.7 | 1.6 |
| TOTAL | 62.5 | 47.9 | 43.0 | 40.6 | 39.1 | 36.2 | 34.7 |

| Maintenance Burden (¢/mi) |
| Aircraft | 28.2 | 21.6 | 19.4 | 18.3 | 17.7 | 16.3 | 15.7 |
| Engines | 10.0 | 7.7 | 6.9 | 6.5 | 6.3 | 5.8 | 5.6 |
| Props (Rotors) | 5.0 | 3.8 | 3.4 | 3.2 | 3.1 | 2.9 | 2.8 |
| Electronics | 6.4 | 4.9 | 4.4 | 4.2 | 4.0 | 3.7 | 3.6 |
| Airframe Spares | 3.5 | 2.7 | 2.4 | 2.3 | 2.2 | 2.1 | 2.0 |
| Engine Spares | 7.5 | 5.8 | 5.2 | 4.9 | 4.7 | 4.4 | 4.2 |
| TOTAL | 60.8 | 46.5 | 41.8 | 39.4 | 38.0 | 35.2 | 33.8 |

| TOTAL DOC |
| (¢/mi) | 277.5 | 211.8 | 189.9 | 178.9 | 172.4 | 159.2 | 152.6 |
| ($) | 419.1 | 413.8 | 411.2 | 409.7 | 408.7 | 406.4 | 405.2 |
| ($/av. seat) | 0.35 | 0.53 | 0.71 | 0.90 | 1.08 | 1.19 | 1.32 |
| ($/av. seat-mi) | 3.47 | 2.65 | 2.37 | 2.24 | 2.15 | 1.99 | 1.91 |

<table>
<thead>
<tr>
<th>Stage Length (mi)</th>
<th>10</th>
<th>20</th>
<th>30</th>
<th>40</th>
<th>50</th>
<th>100</th>
<th>200</th>
</tr>
</thead>
<tbody>
<tr>
<td>Fuel Burn (lb)</td>
<td>434</td>
<td>635</td>
<td>836</td>
<td>1036</td>
<td>1237</td>
<td>2241</td>
<td>4249</td>
</tr>
<tr>
<td>Block Speed (mph)</td>
<td>155</td>
<td>203</td>
<td>226</td>
<td>239</td>
<td>248</td>
<td>268</td>
<td>280</td>
</tr>
</tbody>
</table>
**Fig. R21  TYPICAL STOWED ROTOR**

**DOC Input Parameters**

- Number of Engines = 4
- Max. Cruise Speed = 450 mph
- HP of One Engine = 3252 hp (NRP)
- Weight of One Engine = 650 lb.
- Number of Passengers = 80
- Design Range = 200 mi.
- Propeller (Rotor) Weight = 8823 lb.
- Empty Weight (less engines) = 48323 lb
- Production Run (Airframe) = 300
- $/lb engines = $300/lb
- Maintenance Factor = 1.3
- Engine TBO = 4000 hr.

**DOC Output**

<table>
<thead>
<tr>
<th>Stage Length (mi.)</th>
<th>40</th>
<th>50</th>
<th>60</th>
<th>70</th>
<th>80</th>
<th>100</th>
<th>200</th>
</tr>
</thead>
<tbody>
<tr>
<td>Flight Operations (¢/mi)</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Pilot</td>
<td>11.1</td>
<td>10.6</td>
<td>10.3</td>
<td>10.0</td>
<td>9.9</td>
<td>9.6</td>
<td>9.1</td>
</tr>
<tr>
<td>Copilot</td>
<td>6.9</td>
<td>6.5</td>
<td>6.3</td>
<td>6.2</td>
<td>6.0</td>
<td>5.9</td>
<td>5.5</td>
</tr>
<tr>
<td>Fuel</td>
<td>38.9</td>
<td>37.4</td>
<td>36.3</td>
<td>35.5</td>
<td>35.0</td>
<td>34.2</td>
<td>32.6</td>
</tr>
<tr>
<td>Insurance</td>
<td>19.0</td>
<td>17.6</td>
<td>16.7</td>
<td>16.1</td>
<td>15.6</td>
<td>14.9</td>
<td>13.5</td>
</tr>
<tr>
<td>TOTAL</td>
<td>76.0</td>
<td>72.2</td>
<td>69.6</td>
<td>67.8</td>
<td>66.4</td>
<td>64.5</td>
<td>60.7</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Direct Maintenance (¢/mi)</th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Labor Airframe</td>
<td>8.7</td>
<td>8.1</td>
<td>7.7</td>
<td>7.4</td>
<td>7.1</td>
<td>6.8</td>
<td>6.2</td>
</tr>
<tr>
<td>Materials Airframe</td>
<td>13.3</td>
<td>12.3</td>
<td>11.7</td>
<td>11.2</td>
<td>10.9</td>
<td>10.4</td>
<td>9.4</td>
</tr>
<tr>
<td>Materials Engines</td>
<td>17.8</td>
<td>16.6</td>
<td>15.7</td>
<td>15.1</td>
<td>14.6</td>
<td>14.0</td>
<td>12.7</td>
</tr>
<tr>
<td>Labor Engines</td>
<td>1.6</td>
<td>1.5</td>
<td>1.4</td>
<td>1.3</td>
<td>1.3</td>
<td>1.2</td>
<td>1.1</td>
</tr>
<tr>
<td>TOTAL</td>
<td>41.4</td>
<td>38.4</td>
<td>36.5</td>
<td>35.0</td>
<td>34.0</td>
<td>32.5</td>
<td>29.5</td>
</tr>
</tbody>
</table>

**Maintenance Burden (¢/mi)**

<table>
<thead>
<tr>
<th>Depreciation Items (¢/mi)</th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Aircraft</td>
<td>20.9</td>
<td>19.4</td>
<td>18.4</td>
<td>17.7</td>
<td>17.1</td>
<td>16.4</td>
<td>14.9</td>
</tr>
<tr>
<td>Engines</td>
<td>6.4</td>
<td>5.9</td>
<td>5.6</td>
<td>5.4</td>
<td>5.3</td>
<td>5.0</td>
<td>4.6</td>
</tr>
<tr>
<td>Props (Rotors)</td>
<td>4.9</td>
<td>4.6</td>
<td>4.3</td>
<td>4.2</td>
<td>4.0</td>
<td>3.9</td>
<td>3.5</td>
</tr>
<tr>
<td>Electronics</td>
<td>3.5</td>
<td>3.2</td>
<td>3.1</td>
<td>2.9</td>
<td>2.9</td>
<td>2.7</td>
<td>2.5</td>
</tr>
<tr>
<td>Airframe Spares</td>
<td>2.7</td>
<td>2.5</td>
<td>2.4</td>
<td>2.3</td>
<td>2.2</td>
<td>2.1</td>
<td>1.9</td>
</tr>
<tr>
<td>Engine Spares</td>
<td>4.8</td>
<td>4.5</td>
<td>4.2</td>
<td>4.1</td>
<td>3.9</td>
<td>3.8</td>
<td>3.4</td>
</tr>
<tr>
<td>TOTAL</td>
<td>43.2</td>
<td>40.1</td>
<td>38.0</td>
<td>36.5</td>
<td>35.4</td>
<td>33.9</td>
<td>30.7</td>
</tr>
</tbody>
</table>

**TOTAL DOC**

<table>
<thead>
<tr>
<th>¢/mi</th>
<th>177.1</th>
<th>165.9</th>
<th>158.5</th>
<th>153.3</th>
<th>149.3</th>
<th>143.7</th>
<th>132.6</th>
</tr>
</thead>
<tbody>
<tr>
<td>$/hr</td>
<td>490.6</td>
<td>494.4</td>
<td>497.2</td>
<td>499.5</td>
<td>501.3</td>
<td>504.0</td>
<td>510.4</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>($/av. seat)</th>
<th>1.04</th>
<th>1.19</th>
<th>1.34</th>
<th>1.49</th>
<th>1.80</th>
<th>3.32</th>
</tr>
</thead>
<tbody>
<tr>
<td>($/av. seat-mi)</td>
<td>2.21</td>
<td>2.07</td>
<td>1.98</td>
<td>1.87</td>
<td>1.80</td>
<td>1.66</td>
</tr>
</tbody>
</table>

**Stage Length (mi.)**

- 40  50  60  70  80  100  200

**Fuel Burn (lb.)**

- 878 1054 1229 1405 1580 1931 3686

**Block Speed (mph)**

- 287 310 327 340 351 367 404
**Fig. R22 TYPICAL HELICOPTER**

**DOC Input Parameters**

<table>
<thead>
<tr>
<th>Item</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Number of Engines</td>
<td>4</td>
</tr>
<tr>
<td>Max. Cruise Speed</td>
<td>230 mph</td>
</tr>
<tr>
<td>HP of One Engine</td>
<td>1773 hp (NRP)</td>
</tr>
<tr>
<td>Weight of One Engine</td>
<td>284 lb.</td>
</tr>
<tr>
<td>Number of Passengers</td>
<td>80</td>
</tr>
<tr>
<td>Design Range</td>
<td>200 mi.</td>
</tr>
<tr>
<td>Propeller (Rotor) Weight</td>
<td>4254 lb.</td>
</tr>
<tr>
<td>Empty Weight (less engines)</td>
<td>28798 lb.</td>
</tr>
<tr>
<td>Gross Weight</td>
<td>50442 lb.</td>
</tr>
<tr>
<td>Annual Utilization</td>
<td>3000 hr.</td>
</tr>
<tr>
<td>Depreciation Period</td>
<td>12 yr.</td>
</tr>
<tr>
<td>Production Run (Airframe)</td>
<td>300</td>
</tr>
<tr>
<td>$/lb engines</td>
<td>$300./lb.</td>
</tr>
<tr>
<td>Maintenance Factor</td>
<td>1.3</td>
</tr>
<tr>
<td>Engine TBO</td>
<td>4000 hr.</td>
</tr>
</tbody>
</table>

**DOC Output**

| Stage Length (mi.) | 10   20   30   40   50   100  200 |
|--------------------|------|------|------|------|------|------|------|
| Flight Operations  |      |      |      |      |      |      |      |
| Pilot              | 16.5 | 14.4 | 13.7 | 13.3 | 13.1 | 12.7 | 12.5 |
| Copilot            | 10.5 | 9.1  | 8.6  | 8.4  | 8.2  | 7.9  | 7.8  |
| Fuel               | 41.5 | 32.6 | 29.6 | 28.2 | 27.3 | 25.5 | 24.6 |
| Insurance          | 21.1 | 17.3 | 16.0 | 15.4 | 15.0 | 14.2 | 13.8 |
| TOTAL              | 89.6 | 73.3 | 67.9 | 65.2 | 63.6 | 60.4 | 58.7 |

| Stage Length (mi) | 10   20   30   40   50   100  200 |
|-------------------|------|------|------|------|------|------|------|
| Block Speed (mph) | 146  | 179  | 193  | 201  | 206  | 218  | 224  |

<table>
<thead>
<tr>
<th>Direct Maintenance ($/mi)</th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Labor Airframe</td>
<td>13.5</td>
<td>11.1</td>
<td>10.3</td>
<td>9.8</td>
<td>9.6</td>
<td>9.1</td>
<td>8.9</td>
</tr>
<tr>
<td>Materials Airframe</td>
<td>16.5</td>
<td>13.5</td>
<td>12.5</td>
<td>12.0</td>
<td>11.7</td>
<td>11.1</td>
<td>10.8</td>
</tr>
<tr>
<td>Materials Engines</td>
<td>14.7</td>
<td>12.0</td>
<td>11.1</td>
<td>10.7</td>
<td>10.4</td>
<td>9.9</td>
<td>9.6</td>
</tr>
<tr>
<td>Labor Engines</td>
<td>2.9</td>
<td>2.4</td>
<td>2.2</td>
<td>2.1</td>
<td>2.0</td>
<td>1.9</td>
<td>1.9</td>
</tr>
<tr>
<td>TOTAL</td>
<td>47.6</td>
<td>39.0</td>
<td>36.1</td>
<td>34.6</td>
<td>33.7</td>
<td>32.0</td>
<td>31.1</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Maintenance Burden ($/mi)</th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Aircraft</td>
<td>24.6</td>
<td>20.1</td>
<td>18.6</td>
<td>17.9</td>
<td>17.4</td>
<td>16.5</td>
<td>16.1</td>
</tr>
<tr>
<td>Engines</td>
<td>5.5</td>
<td>4.5</td>
<td>4.2</td>
<td>4.0</td>
<td>3.9</td>
<td>3.7</td>
<td>3.6</td>
</tr>
<tr>
<td>Props (Rotors)</td>
<td>4.7</td>
<td>3.8</td>
<td>3.5</td>
<td>3.4</td>
<td>3.3</td>
<td>3.1</td>
<td>3.1</td>
</tr>
<tr>
<td>Electronics</td>
<td>6.8</td>
<td>5.6</td>
<td>5.2</td>
<td>5.0</td>
<td>4.8</td>
<td>4.6</td>
<td>4.5</td>
</tr>
<tr>
<td>Airframe Spares</td>
<td>3.2</td>
<td>2.6</td>
<td>2.4</td>
<td>2.3</td>
<td>2.2</td>
<td>2.1</td>
<td>2.1</td>
</tr>
<tr>
<td>Engine Spares</td>
<td>4.1</td>
<td>3.4</td>
<td>3.1</td>
<td>3.0</td>
<td>2.9</td>
<td>2.8</td>
<td>2.7</td>
</tr>
<tr>
<td>TOTAL</td>
<td>48.9</td>
<td>40.0</td>
<td>37.0</td>
<td>35.5</td>
<td>34.7</td>
<td>32.9</td>
<td>32.0</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>TOTAL DOC</th>
<th>208.0</th>
<th>170.2</th>
<th>157.6</th>
<th>151.3</th>
<th>147.5</th>
<th>139.9</th>
<th>136.2</th>
</tr>
</thead>
<tbody>
<tr>
<td>($/hr)</td>
<td>292.7</td>
<td>290.3</td>
<td>289.3</td>
<td>288.7</td>
<td>288.3</td>
<td>287.5</td>
<td>287.1</td>
</tr>
<tr>
<td>($/av. seat)</td>
<td>0.26</td>
<td>0.43</td>
<td>0.59</td>
<td>0.76</td>
<td>0.92</td>
<td>1.75</td>
<td>3.40</td>
</tr>
<tr>
<td>($/av. seat-mi)</td>
<td>2.60</td>
<td>2.13</td>
<td>1.97</td>
<td>1.89</td>
<td>1.84</td>
<td>1.75</td>
<td>1.70</td>
</tr>
</tbody>
</table>