USNTPS-FTM-No. 106

U.S. NAVAL TEST PILOT SCHOOL
FLIGHT TEST MANUAL

ROTARY WING PERFORMANCE

NAVAL AIR WARFARE CENTER
PATUXENT RIVER, MARYLAND

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This Flight Test Manual, published under the authority of the Commanding Officer, U.S. Naval Test Pilot School, is intended primarily as a text for the pilots, engineers and flight officers attending the school. Additionally, it is intended to serve as a reference document for those engaged in flight testing. Corrections and update recommendations to this manual are welcome and may be submitted to:

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TABLE OF CONTENTS

ROTARY WING PERFORMANCE

CHAPTER ONE

INTRODUCTION

CHAPTER ONE

INTRODUCTION

1.1 OBJECTIVE

The Rotary Wing Performance Flight Test Manual (FTM) is intended to serve primarily as a text for the pilots, engineers, and Naval Flight Officers attending the U.S. Naval Test Pilot School (USNTPS). Additionally, it is intended to be a reference document for those engaged in rotary wing performance flight testing at USNTPS, the Naval Air Warfare Center Aircraft Division (NAWCAD), and other military and civilian organizations interested in helicopter performance flight testing. A manual with such broad aims may fall short of its intended mark for some future users, however, in the development of this manual, the primary purpose was a school text.

The manual is not intended to be a substitute for textbooks on helicopter performance. Rather, it is intended to summarize the applicable theory, show the relationship of the flight test variables to the theory, describe the flight test techniques, and aid in the data management required for each flight test. The Rotary Wing Performance FTM is not intended to replace the Flight Test and Engineering Group Report Writing Guide as the guide to correct reporting, but does contain a discussion on the mission suitability or effect on mission performance of the various performance parameters and a discussion of specification compliance where applicable.

This manual contains examples of performance parameters discussed in narrative and graphic format. The examples in this manual show trends, contain examples of current helicopters, and are in the format currently used in the USNTPS Helicopter Performance Testing, Data Reduction Programs.

This FTM, since it is a text for USNTPS, contains information relative to operations at USNTPS and NAWCAD; however, it does not contain information relative to the scope of a particular USNTPS syllabus exercise or to the reporting requirements for a particular exercise. Detailed information for each flight exercise will vary from time to time as resources and personnel change and will be briefed separately to each individual class.

1.2 ORGANIZATION

1.2.1 Manual Organization

This manual is organized into individual chapters. While an attempt is made to treat each subject chapter completely and independently, that is not always possible due to the very nature of the interdependence of performance parameters. For example, engine performance data (power available) influence hover performance. Likewise, power available data and level flight power required data influence climb performance. Where applicable, reference to a preceding chapter will be made to show the connections.

Chapter 1 is an overview of the manual.

Chapter 2 is concerned with pitot and static system performance. The source of system errors is discussed along with some methods of reducing those errors. Test methods of measured course, trailing bomb, trailing cone, tower fly-by, paced, and space positioning are described.

Chapter 3 covers the area of engine assessment. Discussion includes engine/rotor controls and displays; engine/rotor operating procedures; engine/rotor start, run-up, and shutdown characteristics; engine acceleration characteristics; engine trim response; engine/rotor stability; torque matching; engine limiting characteristics; and power contribution in descent.

Chapter 4 covers the area of engine performance characteristics. This chapter includes the relationship of corrected engine parameters: corrected engine shaft horsepower, corrected engine gas generator speed, corrected fuel flow, corrected power turbine inlet temperature, and specific fuel consumption. Inlet recovery is discussed along with specification compliance.

Chapter 5 covers hover performance and establishes the relationship between engine performance (power available) and airframe performance (power required). Both tethered hover and free-flight test methods are covered in this chapter. Test methods for the free-flight hover performance include ground reference using the hover height measuring device (HHMD) and air reference methods.

Chapter 6 covers vertical climb. This chapter deals with both test day vertical climb performance and generalized vertical climb performance. The relationship of power available and hover power required are brought together in the vertical climb. Ground referenced and air referenced methods are discussed.

Chapter 7 discusses level flight performance. This chapter deals with both test day data and generalized level flight performance. Again, this chapter draws together engine performance (power available) with airframe performance (power required) and develops important aircraft performance parameters (range, endurance, and maximum level flight speed). The test method using the constant weight to pressure ratio (W/δ) and rotor speed to temperature ratio ($N_R/\sqrt{\theta}$) is compared to the constant weight to density ratio (W/σ) method.

Chapter 8 discusses climb and descent performance. This chapter covers test day data as well as generalized climb and descent performance. The chapter brings together engine performance (power available) and airframe performance (power required in level flight) to predict aircraft performance parameters (rate of climb or descent). Test methods of power correction factor (K_P), weight correction factor (K_W), generalized climb performance, and test day performance are discussed.

1.2.2 Chapter Organization

Each chapter has the same internal organization where possible. Following the chapter introduction, the second section gives the purpose of the test. The third section is a review of the applicable theory. The next section discusses the test methods and techniques along with the data requirements and safety precautions applicable to those methods. The fifth section discusses data reduction, both manual and computer aided as presently performed in exercises at USNTPS. This section is not intended to replace the USNTPS Helicopter Performance Testing, Data Reduction Programs, but rather to complement it. The sixth section is the data analysis. The seventh section covers the relevant mission suitability aspects of the performance parameter. The eighth section discusses the relevant specification compliance issues. The ninth section is a glossary of terms introduced in the chapter. Finally, the tenth section is a listing of applicable references.

1.3 PERFORMANCE SYLLABUS

1.3.1 Overview

The performance syllabus at USNTPS consists of academic instruction, flight briefings, demonstration flights, practice flights, exercise flights, reports, and evaluation flights. The performance phase of instruction culminates in a group Technical Evaluation (TECHEVAL) and an individual performance evaluation flight. The culmination of training at USNTPS is a simulated Developmental Test IIA (DT IIA). This exercise incorporates all the performance, stability and control, and airborne systems instruction and practice.

The performance syllabus includes pitot static system testing, engine assessment testing, hover performance testing, level flight performance testing, and climb and descent testing. The syllabus is presented in a step-by-step, building block approach which allows concentration on specific objectives and fundamentals. This approach tends to isolate the individual areas and deemphasize the total evaluation. Progress through the syllabus must be made with the end objective in mind, the evaluation of the helicopter performance as a weapon system in the mission environment. The details of the current syllabus is contained in U.S. Naval Test Pilot School Notice 1542.

1.3.2 Flight Briefings

Each exercise in the performance syllabus is preceded by academic instruction to form the basis of the applicable theory and a printed and oral flight briefing presented by the principal instructor for the exercise. The flight briefing gives specific details of the exercise and the exercise requirements. The flight briefing covers the purpose, references, test conditions, method of test to be used, the scope of test, test planning, and report requirements. The briefing also covers the applicable safety requirements for the exercise as well as administrative and support requirements.

1.3.3 Demonstration Flights

One or more dual demonstration flights with an instructor are conducted during the performance phase. The purpose of the demonstration flight is to demonstrate and practice the correct test methods and techniques. The flight is not intended to cover, in detail, the theory on which the methods are based. In general, the demonstration flights will precede the academic instruction which will lay the foundation for the theory.

1.3.4 Practice Flights

Each student is afforded the opportunity to practice the performance test methods and techniques in flight after the demonstration flights and prior to the exercise or data flight. The purpose of these practice flights is to gain proficiency in the test techniques, data acquisition, and crew coordination necessary for safe and efficient flight testing.

1.3.5 Exercise Flights

Each student will be provided an opportunity to fly one or more flights as part of each exercise. The student is expected to plan the flight, have the plan approved, and fly the flight in accordance with the plan. The purpose of the flight is to gather both quantitative and qualitative data as part of an overall performance evaluation. The primary objective in the flight is the safe and efficient conduct of the flight. Under no conditions will flight safety be compromised.

1.3.6 Reports

A fundamental purpose of USNTPS is developing the test pilot/engineer's ability to report test results in clear, concise technical terms. After the completion of the exercise flight, the student reduces the data, that is, changes the format of the data from that gathered during the flight to that used for engineering analysis. The student analyzes the data for both mission suitability and specification compliance. The data are presented in the proper format and a report is prepared. The report process enhances the ability to combine factual data gathered from ground and flight tests with an analysis of the effect on mission suitability. Reporting further communicates the conclusions in a manner which answers the questions implicit in the purpose of the test.

1.4 REASONS FOR PERFORMANCE FLIGHT TESTING

1.4.1 Mission Suitability

Each aircraft or weapon system is developed to meet or fulfill a mission requirement. This mission is comprised of many sub-missions and tasks, and they must be performed in many areas of the world under different conditions. Mission suitability is the capability of the helicopter to fulfill its mission. In order to evaluate mission suitability, the student must have a thorough understanding of the entire mission and the environment in which it is to be performed. Reference material must be consulted, as well as discussions with class members and staff who are familiar with the mission.

Inherent to mission suitability determination is the conclusion whether the aircraft can perform the intended mission safely, whether capability can be substantially improved, or whether a particular characteristic should be avoided in the future. The test pilot/flight test engineer provides the information necessary to make decisions on a weapons system based on these conclusions. Additionally, during performance evaluations, specific characteristics may be identified that, if appropriately applied in flight technique, may enhance mission capability and should be recommended as standard operating procedure.

1.4.2 Specification Compliance

Each helicopter or weapon system is developed by a manufacturer to meet certain specifications. To insure contract compliance, the government must check the performance parameters of the system against the contractual requirements and/or specifications. This consideration demands accurate quantitative data derived as the result of flight testing.

1.4.3 Handbook Performance

To provide the operators in the field or fleet with the maximum performance possible from the helicopter, contractually supplied performance data must be verified and sometimes expanded for inclusion in the flight handbook. Additionally, operating techniques which affect the helicopter's performance must be verified or developed.

1.5 APPROACH TO PERFORMANCE FLIGHT TESTING

1.5.1 Safety/Risk Management

The paramount concern during any flight, and especially during flight testing, is the safety of those involved in the flight. The safety of the flight crew and ground support personnel is of utmost concern. However, risk is inescapable in any activity. But, although risk can not be eliminated, it can be minimized through proper risk management techniques. Thus, risk management is a tool whereby safety can be optimized. All flight operations and all flight testing must be conducted with the principle of safety in mind. Safety is promoted by detailed knowledge of the system, thorough planning, adherence to the plan, complete crew coordination (both flight crew and support crew), a rigorous look-out policy, and disciplined flight conduct. Safe flight testing is efficient and effective flight testing. The key to risk management is knowledge, planning, and execution. Remember, the first priority is to fly the aircraft safely, the second priority is to evaluate the aircraft.

1.5.2 Concept

The foundation of flight testing in general, and performance testing in particular, is dependent upon identifying the applicable variables for a given condition based on sound theory with appropriately applied assumptions and simplifications. The flight test is conducted holding as many variables as possible constant while incrementally changing one parameter and observing the result. Educated and efficient test planning and careful observation yield test results that address specific aspects of performance which can be analyzed in the context of the intended mission.

1.5.3 Incremental Build-Up

The concept of incremental build-up is one of the most important practical and philosophical flight testing lessons offered at USNTPS. Practically, build-up is the process of proceeding from the known to the unknown in an incremental, methodical pattern. Philosophically, build-up is the manner in which flight testing should be structured; beginning with the best documented, least hazardous data points and proceeding toward the desired end points, always conscious of the aircraft, pilot, and test limits. Build-up is applicable to both the overall structure of the evaluation and to each of the subtasks. There should be no surprises in flight testing. A major theme contained in the build-up concept is that in the event a data point yields an unexpected result, or a series of data points suggests an unexpected trend, the flight test ceases until the results are analyzed and explained.

1.5.4 Quantitative and Qualitative

While specification compliance and handbook performance requirements dictate that quantitative performance data are gathered, qualitative data are gathered also during performance flight testing. The educated observations of the flight test crew often provide more insight into the overall suitability of a weapons system for its mission than the quantitative data. During all flight tests the test crew continuously makes and records observations. These observations may be recorded manually or automatically. They may relate directly to the test being performed or they may relate to the general flight environment. Flight test time is expensive and all opportunities to evaluate the helicopter must be taken.

1.6 GENERALIZED PERFORMANCE DATA VERSUS TEST DAY DATA

The performance of the helicopter is evaluated under the conditions that exist at the time of evaluation. Data can be analyzed alone and conclusions can be made relative to the performance of the helicopter for that given set of test day conditions. Usually, data of this type are termed test day data. When the test day data are analyzed using a generalized method of data reduction, the performance of the helicopter can be determined at conditions of interest other than those at which the data was gathered. The distinction of test day data versus generalized data reduction methods will be made throughout this manual as appropriate.

1.7 INSTRUMENTATION

Performance flight testing is conducted in part to obtain accurate performance indications of the helicopter. To accomplish this objective, the source of the data must be as accurate as possible. In flight testing, a combination of calibrated production instruments and sensing systems, along with special flight test instrumentation and sensing systems, are used. Automatic data recording equipment usually is employed. These calibrated systems are compared to a reference system in the laboratory and a calibration curve or comparison is made. This calibration curve is used to increase the accuracy of the observed data by applying the calibration factor and obtaining corrected indicated data. The laboratory reference data are periodically compared to national standards which are maintained at agencies such as the National Bureau of Standards. The instrumentation systems currently in use consist of both cockpit observed and manually recorded data, onboard automatically recorded data, and data that are transmitted to the ground via telemetry. The telemetry data may be observed and recorded on the ground in real time. Thus, the flight test crew can be supported by a ground crew which monitors the progress of the flight.

1.8 GLOSSARY

1.8.1 Notations

DT IIA	Developmental Test II A	
FTM	Flight Test Manual	
HHMD	Hover height measuring device	
K_P	Power correction factor	
K_W	Weight correction factor	
NAWCAD	Naval Air Warfare Center Aircraft Division	
$N_R/\sqrt{\theta}$	Main rotor speed to the square root of the temperature ratio	%, rpm
TECHEVAL	Technical Evaluation	
W/δ	Weight to pressure ratio	lb
W/σ	Weight to density ratio	lb

1.9 REFERENCES

FTEG-TID-94-1-RWG, "Report Writing Guide for Flight Test and Engineering Group Reports", Naval Air Warfare Center Aircraft Division, Patuxent River, MD., 01 March 1994.

McCue, J.J., "U.S. Naval Test Pilot School, Helicopter Performance Testing, Data Reduction Programs", USNTPS, Patuxent River, MD., June 1982.

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CHAPTER TWO

PITOT/STATIC SYSTEM PERFORMANCE

CHAPTER TWO

FIGURES

CHAPTER TWO

TABLES

CHAPTER TWO

EQUATIONS

$$\Delta H_{pos} = f(P_s - P_a)$$ *eq 2.1*

$$V_T^2 = \frac{2\gamma}{\gamma - 1}\frac{P_a}{\rho_a}\left[\left(\frac{P_T - P_a}{P_a} + 1\right)^{\frac{\gamma - 1}{\gamma}} - 1\right]$$ *eq 2.2*

$$V_c^2 = \frac{2\gamma}{\gamma - 1}\frac{P_{ssl}}{\rho_{ssl}}\left[\left(\frac{P_T - P_a}{P_{ssl}} + 1\right)^{\frac{\gamma - 1}{\gamma}} - 1\right]$$ *eq 2.3*

$$V_c = f(P_T - P_a) = f(q_c)$$ *eq 2.4*

$$V_i = f(P_T - P_s) = f(q_{ci})$$ *eq 2.5*

$$\Delta V_{pos} = f(P_s - P_a),\ q_c$$ *eq 2.6*

$$P_s + \frac{\rho V^2}{2} = \text{Constant}$$ *eq 2.7*

$$P_a + \frac{\rho V^2}{2} = P_T$$ *eq 2.8*

$$V = \sqrt{\frac{2(P_T - P_a)}{\rho}}$$ *eq 2.9*

$$V_e = \sqrt{\frac{2(P_T - P_a)}{0.0023769}} \qquad \textit{eq 2.10}$$

$$V_T = \frac{V_e}{\sqrt{\sigma}} \qquad \textit{eq 2.11}$$

$$H_{P_{ic}} = H_{P_o} + \Delta H_{P_{ic}} \qquad \textit{eq 2.12}$$

$$H_{P_{ic\,ref}} = H_{P_{o\,ref}} + \Delta H_{P_{ic\,ref}} \qquad \textit{eq 2.13}$$

$$H_{P_c} = H_{P_{ic\,ref}} - h_{ref} + h_{a/c} \qquad \textit{eq 2.14}$$

$$\Delta H_{pos} = H_{P_c} - H_{P_{ic}} \qquad \textit{eq 2.15}$$

$$V_{ic} = V_o + \Delta V_{ic} \qquad \textit{eq 2.16}$$

$$V_{G1} = 0.5917\left(\frac{D}{\Delta t_1}\right) \qquad \textit{eq 2.17}$$

$$V_{G2} = 0.5917\left(\frac{D}{\Delta t_2}\right) \qquad \textit{eq 2.18}$$

$$V_T = \frac{V_{G1} + V_{G2}}{2} \qquad \textit{eq 2.19}$$

$$T_a = T_o + \Delta T_{ic} \qquad \textit{eq 2.20}$$

$$V_c = \frac{V_T}{\sqrt{\sigma}} \qquad \textit{eq 2.21}$$

$$\Delta V_{pos} = V_c - V_{ic} \qquad \textit{eq 2.22}$$

$$\alpha_{ic} = \alpha_o + \Delta\alpha_{ic}$$ *eq 2.23*

$$\beta_{ic} = \beta_o + \Delta\beta_{ic}$$ *eq 2.24*

$$H_{P_c} = H_{P_{o\ ref}} + \Delta H_{P_{ic\ ref}} + \Delta H_{pos\ ref}$$ *eq 2.25*

$$H_{P_{ic\ ship}} = H_{P_{o\ ship}} + \Delta H_{P_{ic\ ship}}$$ *eq 2.26*

$$\Delta H_{pos\ ship} = H_{P_c} - H_{P_{ic\ ship}}$$ *eq 2.27*

$$V_c = V_{o\ ref} + \Delta V_{ic\ ref} + \Delta V_{pos\ ref}$$ *eq 2.28*

$$V_{ic\ ship} = V_{o\ ship} + \Delta V_{ic\ ship}$$ *eq 2.29*

$$\Delta V_{pos\ ship} = V_c - V_{ic\ ship}$$ *eq 2.30*

$$H_{P_{c\ twr}} = H_{P_{o\ twr}} + \Delta H_{P_{ic\ twr}}$$ *eq 2.31*

$$\Delta h = d \tan\theta$$ *eq 2.32*

$$H_{P_c} = H_{P_{ctwr}} + \Delta h$$ *eq 2.33*

$$\Delta P_{s\ ship} = P_{s\ ref} - P_{s_{ic\ ship}}$$ *eq 2.34*

$$T_a\ (°K) = T_a\ (°C) + 273.15$$ *eq 2.35*

$$a = 38.9678\ \sqrt{T_a\ (°K)}$$ *eq 2.36*

$$\Delta V_{pos} = \left[\frac{\Delta P_{s\ ship}\ \left(a^2\right)}{41.888\ \left(V_{ic}\right)}\right]\left[\frac{2.5}{1 + 0.2\left(\frac{V_{ic}}{a}\right)^2}\right] \qquad \textit{eq 2.37}$$

$$V_c = V_{ic} + \Delta V_{pos} \qquad \textit{eq 2.38}$$

CHAPTER TWO

PITOT/STATIC SYSTEM PERFORMANCE

2.1 INTRODUCTION

This chapter presents a discussion of pitot/static system performance testing. The theoretical aspects of these flight tests are included. Each test method and technique applicable to helicopter pitot/static testing is discussed in some detail. Data reduction techniques and some important factors in the analysis of the data are also included. Mission suitability factors are discussed and include real life examples of recent pitot/static problems. The chapter concludes with a glossary of terms used in these tests and the references which were used in constructing this portion of the manual.

2.2 PURPOSE OF TEST

The purpose of this flight test is to investigate thoroughly the flight characteristics of the aircraft pressure sensing systems to achieve the following objectives:

1. Determine the airspeed and altimeter correction data required for performance data processing.
2. Evaluate the requirements of pertinent Military Specifications.
3. Evaluate mission suitability problem areas.

2.3 THEORY

2.3.1 General

The altimeter, airspeed, and vertical rate of climb indicators are three universal flight instruments which require total and/or static pressure inputs to function. The indicated values of these instruments are often incorrect because of the effects of three general categories of errors: instrument errors, lag errors, and position errors.

2.3.2 Instrument Errors

Instrument errors are the result of manufacturing discrepancies, hysteresis, temperature changes, friction, and inertia of moving parts. A laboratory calibration of all flight instruments must be accomplished to determine the instrument corrections prior to an in-flight determination of position errors. Very sensitive instruments may require daily calibration. When the readings of two pressure altimeters are used to determine the error in a pressure sensing system, a precautionary check of calibration correlations is advisable. The problem arises from the fact that two calibrated instruments placed side by side with their readings corrected by use of calibration charts do not always provide the same resultant calibrated value. Tests such as the “Tower Fly-By” or the “Trailing Bomb” for altimeter calibration require an altimeter to provide a reference pressure altitude. These tests require that the reference altimeter be placed next to the ship system altimeter prior to and after each flight. Each altimeter reading should be recorded and, if after calibration corrections have been applied, a discrepancy still exists between the two readings, this discrepancy should be incorporated in the data reduction.

2.3.3 Lag Error

The presence of lag error in pressure measurements is generally associated with climbing/descending or accelerating/decelerating flight and is more evident in static systems. When changing ambient pressures are involved, as in climbing and descending flights, the speed of pressure propagation and the pressure drop associated with flow through a tube introduces lag between the indicated and actual pressure. The pressure lag error is basically a result of:

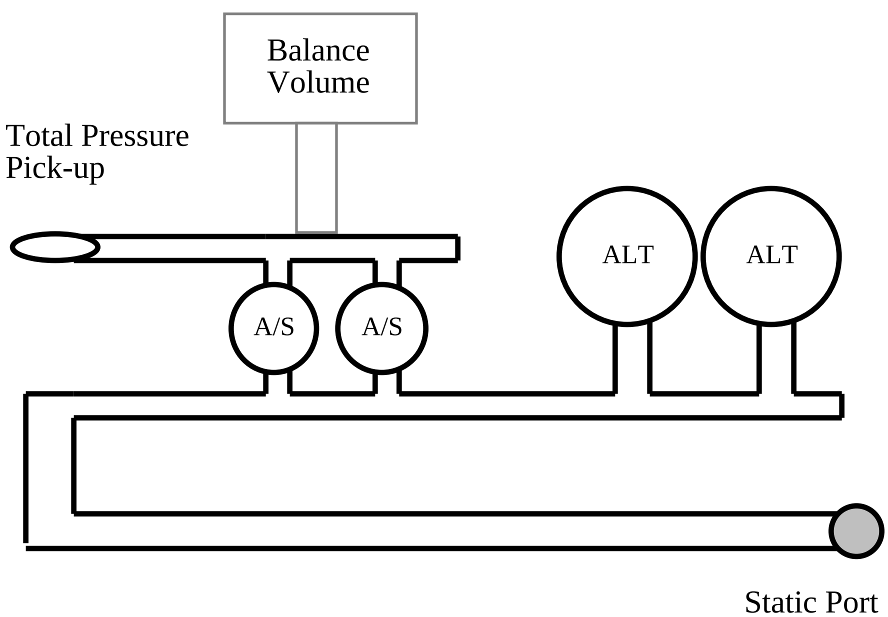
1. Pressure drop in the tubing due to viscous friction.
2. Inertia of the air mass in the tubing.
3. Volume of the system.
4. Instrument inertia and viscous and kinetic friction.
5. The finite speed of pressure propagation; i.e., acoustic lag.

Over a small pressure range the pressure lag is small and can be determined as a constant (λ). Once a lag error constant is determined, a correction can be applied through mathematical manipulations. Another approach, which is more suitable for testing at the

Naval Air Warfare Center Aircraft Division (NAWCAD), is to balance the pressure systems by equalizing their volumes. Balancing minimizes or removes lag error as a factor in airspeed data reduction for flight at a constant dynamic pressure.

2.3.3.1 LAG CONSTANT TEST

The pitot/static pressure systems of a given aircraft supply pressures to an unequal number of instruments and require different lengths of tubing for pressure transmission. The volume of the instrument cases plus the volume in the tubing, when added together for each pressure system, produce a volume mismatch between systems. Figure 2.1 illustrates a configuration where both the length of tubing and total instrument case volumes are unequal. If an increment of pressure were applied simultaneously across the total and static sources of Figure 2.1, the two systems would require different periods of time to stabilize at the new pressure level and a momentary error in indicated airspeed would result.



System	Length of 3/16 Inch Inside Diameter Tube	Total Volume of Instrument Cases
Static	18 ft	370×10^{-4} ft^3
Pitot	6 ft	20×10^{-4} ft^3

Figure 2.1
Analysis of Pitot and Static Systems Construction

The lag error constant (λ) represents the period of time (assuming a first order dynamic response) required for the pressure of each system to reach a value equal to 63.2 percent of the applied pressure increment as shown in Figure 2.2(a). This test can be accomplished on the ground by applying a suction sufficient to develop a ΔH_p equal to 500 feet or an indicated airspeed of 100 knots. Removal of the suction and timing the pressure drop to 184 feet or 37 knots results in the determination of λ_s, the static pressure lag error constant (Figures 2.2(b) and 2.2(c)). If a positive pressure is applied to the total pressure pickup (drain holes closed) to produce a 100 knot indication, the total pressure lag error constant (λ_T) can be determined by measuring the time required for the indicator to drop to 37 knots when the pressure is removed. Generally, the λ_T will be much smaller than the λ_s because of the smaller volume of the airspeed instrument case.

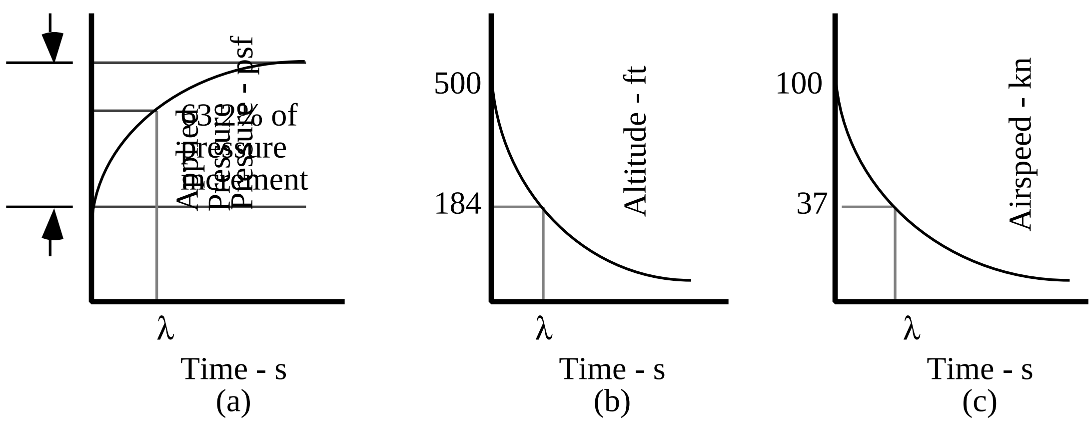


Figure 2.2
Determination of Pitot/Static System Lag Error Constant (λ)

2.3.3.2 SYSTEM BALANCING

The practical approach to lag error testing is to determine if a serious lag error exists, then eliminate it where possible. To test for airspeed system balance, a small increment of pressure (0.1 inch water) is applied simultaneously to both the pitot and static systems. If the airspeed indicator does not fluctuate, the combined systems are balanced and no lag error will exist in indicated airspeed data because the lag constants are matched. Movement of the airspeed pointer indicates that the addition of more volume is required in one of the systems. The addition of cans or tubing (Figure 2.1) generally will provide satisfactory airspeed indications. Balancing will not help the lag in the altimeter, as this difficulty is due to the length of the static system tubing. For instrumentation purposes, lag can be eliminated from the altimeter by locating a static pressure recorder remotely at the static port. The use of balanced airspeed systems and remote static pressure sensors is recommended for procurement testing.

2.3.4 Position Errors

Position errors, or "installation errors", are the result of other than free stream pressures at the pressure sensor, or errors in the local pressure at the source resulting from the shape, location, or orientation of the sensor. In a helicopter, these pressure errors are present at both the total pressure and the static pressure sensors.

Total pressure position errors result from the location of the pitot tube in the rotor downwash and the changing incidence of the probe. The static pressure position errors result from the complete flow around the airframe and may be influenced greatly by the rotor downwash velocity, angle of attack, and sideslip. The following conditions often produce variations in both total and static system position errors:

1. Gross weight/center of gravity locations.
2. Power and rotor speed.
3. Pressure patterns (in-ground effect/out-of-ground effect).
4. Sideslip angles.
5. In-flow angles (climb/descent/level flight).
6. Configuration (landing gear, wing stores, etc.).

2.3.4.1 ALTIMETER POSITION ERROR

The altimeter is an evacuated bellows which expands and contracts and is connected to a series of gears and levers which moves a pointer. The whole mechanism is vented to a static pressure source as shown in Figure 2.3. The static pressure may differ from the atmospheric pressure giving rise to altimeter position errors. The altimeter position error is solely a function of static pressure variations and is expressed as:

$$\Delta H_{pos} = f(P_s - P_a) \qquad \textit{eq 2.1}$$

Where:

ΔH_{pos} - Altimeter position error

P_s - Static pressure

P_a - Ambient pressure.

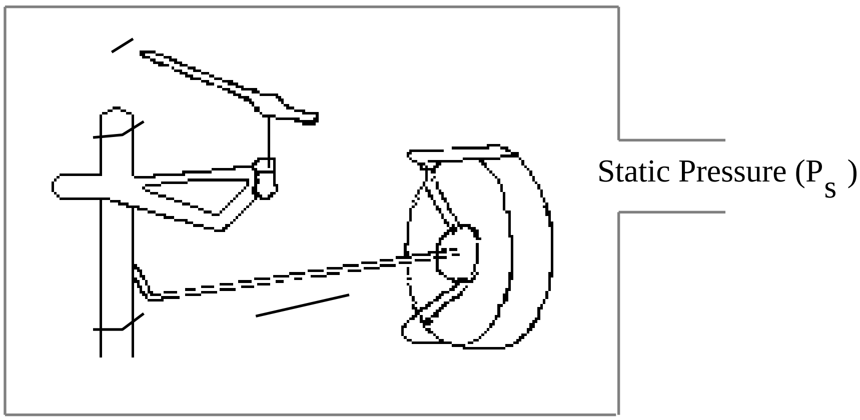


Figure 2.3
Schematic of the Altimeter

2.3.4.2 AIRSPEED POSITION ERROR

The airspeed indicator operates on the principle of Bernoulli's compressible equation for isentropic flow in which airspeed is a function of the difference between total and static pressure. At subsonic speeds Bernoulli's equation, expressed as follows, is applicable:

$$V_T^2 = \frac{2\gamma}{\gamma - 1}\frac{P_a}{\rho_a}\left[\left(\frac{P_T - P_a}{P_a} + 1\right)^{\frac{\gamma - 1}{\gamma}} - 1\right] \qquad \textit{eq 2.2}$$

Where:

V_T - True airspeed

P_a - Ambient pressure

ρ_a - Ambient air density

P_T - Free stream total pressure

γ - Ratio of specific heats (Air: γ = 1.4).

Airspeed indicators for low speed aircraft (including helicopters) only measure the difference between total pressure at the pitot tube and a static source. All other values in Equation 2.2 will be set to standard sea level values. This velocity is called calibrated airspeed, that is, the speed which is shown on a device which is calibrated with standard sea level pressure and density values.

$$V_c^2 = \frac{2\gamma}{\gamma - 1}\frac{P_{ssl}}{\rho_{ssl}}\left[\left(\frac{P_T - P_a}{P_{ssl}} + 1\right)^{\frac{\gamma - 1}{\gamma}} - 1\right] \qquad \textit{eq 2.3}$$

And:

$$V_c = f(P_T - P_a) = f(q_c) \qquad \textit{eq 2.4}$$

Where:

V_c	- Calibrated airspeed
γ	- Ratio of specific heats (Air: $\gamma = 1.4$)
P_a	- Ambient pressure
P_T	- Free stream total pressure
P_{ssl}	- Standard sea level pressure
ρ_{ssl}	- Standard sea level air density
q_c	- Dynamic pressure.

If the measurements of total pressure and static pressure in an airspeed indicator were exactly equal to the free stream total pressure and ambient pressure, the airspeed indicator would indicate calibrated airspeed (V_c). However, this relationship generally does not exist. The dynamic pressure sensed in flight is an indicated dynamic pressure (q_{ci}) and is equal to $P_T - P_s$ thus:

$$V_i = f(P_T - P_s) = f(q_{ci}) \qquad \textit{eq 2.5}$$

Where:

V_i	- Indicated airspeed
P_T	- Free stream total pressure
P_s	- Static pressure
q_{ci}	- Indicated dynamic pressure.

In operation, the airspeed indicator is similar to the altimeter, but instead of being evacuated, the inside of the capsule is connected to a total pressure source and the case is connected to a static pressure source. The instrument then senses P_T and P_s as shown in Figure 2.4. Any differential pressure ($P_T - P_s$) felt by the instrument will indicate the corresponding indicated airspeed (neglecting instrument error). Generally, in a subsonic fixed wing aircraft the airspeed error is only a function of the static pressure error:

$$\Delta V_{pos} = f(P_s - P_a), q_c \qquad \textit{eq 2.6}$$

Where:

ΔV_{pos} - Airspeed position error

P_s - Static pressure

P_a - Ambient pressure

q_c - Dynamic pressure.

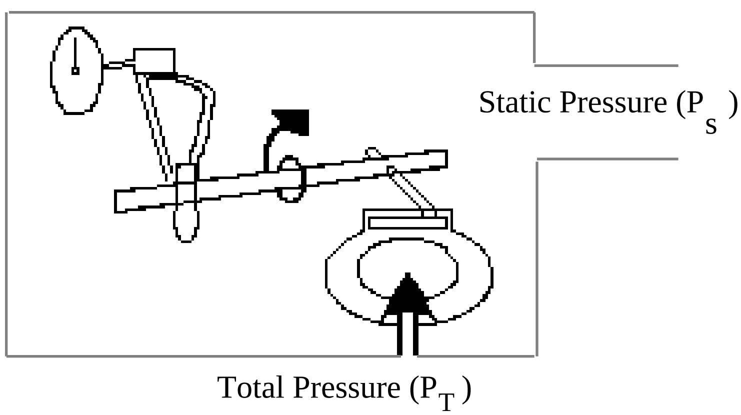


Figure 2.4
Schematic of the Airspeed Indicator

However, in a helicopter where the total pressure source is located in the downwash of the rotor and is subjected to significant change in angle of attack and sideslip, the total pressure error cannot be neglected. Both q_c and q_{ci} are influenced by the measurement of total pressure. Neglecting the total pressure error in a helicopter will result in an error in the determination of calibrated airspeed (V_c).

At speeds slow enough that compressibility effects can be ignored, Bernoulli's equation becomes even more simplified:

$$P_s + \frac{\rho V^2}{2} = \text{Constant} \qquad \textit{eq 2.7}$$

Where:

P_s	- Static pressure
ρ	- Density
V	- Velocity.

If this equation is written for two points in a flow field and one is the stagnation point (such as the pitot tube opening), then Bernoulli's equation would appear as follows:

$$P_a + \frac{\rho V^2}{2} = P_T \qquad \textit{eq 2.8}$$

Where:

P_a	- Ambient pressure
ρ	- Density
V	- Velocity (of the fluid)
P_T	- Free stream total pressure (at the stagnation point).

Some airspeed indicators use this relationship to provide indicated readings.

$$V = \sqrt{\frac{2(P_T - P_a)}{\rho}} \qquad \textit{eq 2.9}$$

Where:

V	- Velocity
P_T	- Free stream total pressure
P_a	- Ambient pressure
ρ	- Density.

If the density term is mechanized in the indicator to represent a standard sea level value, the resulting indication is related only to the difference between total and ambient pressure and is referred to as equivalent airspeed.

$$V_e = \sqrt{\frac{2(P_T - P_a)}{0.0023769}}$$ *eq 2.10*

Where:

V_e	- Equivalent airspeed
P_T	- Free stream total pressure
P_a	- Ambient pressure
0.0023769	- Standard sea level air density, slug / ft^3.

Using Equation 2.9, another important relationship is established between true airspeed and equivalent airspeed.

$$\frac{V_T}{V_e} = \frac{\sqrt{\frac{2(P_T - P_a)}{\rho}}}{\sqrt{\frac{2(P_T - P_a)}{\rho_{ssl}}}}$$

Or

$$V_T = \frac{V_e}{\sqrt{\sigma}}$$ *eq 2.11*

Where:

V_T	- True airspeed
V_e	- Equivalent airspeed
P_T	- Free stream total pressure
P_a	- Ambient pressure
ρ	- Density
ρ_{ssl}	- Standard sea level air density
σ	- Density ratio.

Since V_e and V_c were derived from slightly different assumptions, it is obvious that these two velocities are not the same. The difference between these two velocities is compressibility. A detailed discussion of these relationships is contained in the first reference listed at the end of this chapter. Careful evaluation of these relationships indicates that there is less than 2 knots difference between V_e and V_c, even at the relatively high helicopter operating conditions of 200 knots airspeed and 20,000 feet of altitude. Considerably less error is noted at more representative helicopter operating conditions (typically less than 0.5 knots). Therefore, under most circumstances, for helicopter applications, V_c is assumed equal to V_e.

2.3.5 Power Effects

Although it is difficult to ascertain an exact correlation, power/rotor speed combinations produce a significant effect on helicopter airspeed calibration. The significance of power effects is not difficult to comprehend, considering the proximity of the main rotor to the pitot/static source. The importance of power effects on a given helicopter may vary sharply with variations in calibrated airspeed, center of gravity, and flight path.

2.3.5.1 POSITION ERRORS IN HOVERING FLIGHT

Hovering flight provides an excellent point to start a consideration of power effects. When a helicopter's rotor is engaged, the altimeter normally will indicate a pressure altitude which is lower than indicated prior to engagement. As the helicopter rises vertically to a position only inches above the surface, the altimeter position error usually will reach its maximum negative value. Increasing gross weight during hover in-ground effect (IGE), generally will produce a more negative altimeter position error.

A hover out-of-ground effect (OGE) generally has a similar but reduced effect. There is no ambient pressure rise because of ground effect but there is still a higher than ambient pressure in the rotor wake and around the static source. This higher than actual ambient pressure signal causes the altimeter to indicate an altitude which is lower than actual. Therefore, a negative altimeter position error exists. Any increase in gross weight normally will produce a more negative altimeter position error.

2.3.5.2 PROBE ANGLE OF ATTACK

The angle of attack of the probe is affected during climbs and descents. The angle becomes smaller as the horizontal velocity increases and the vertical velocity decreases.

Figure 2.5 illustrates the angles involved in determining the angle of attack of the probe (α_p). Depending upon the helicopter flight attitudes, the pitot tube may be installed at an angle (up or down) with respect to the airframe centerline. This angle of incidence (α_i) is added to the airframe angle of attack (α_A) to obtain the probe angle of attack (α_p) in the vertical (X - Z) plane.

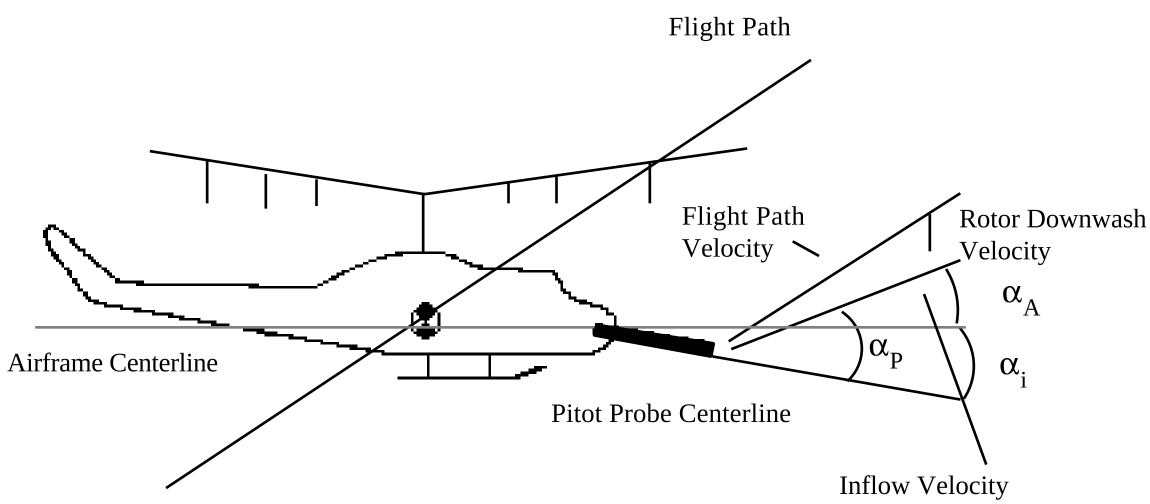


Figure 2.5
Probe Angle of Attack in the Vertical (X - Z) Plane

Most American built single rotor helicopters fly with an inherent right sideslip in balanced flight (zero bank angle and ball centered). The pitot tube may or may not be oriented off centerline, depending on the configuration, to compensate for the inherent sideslip. The combination of sideslip angle of incidence (β_i) and airframe sideslip angle (β_A) results in a probe sideslip angle (β_p) during balanced flight. Figure 2.6 illustrates these sideslip angles. Nominal sideslip angles should produce little total source error.

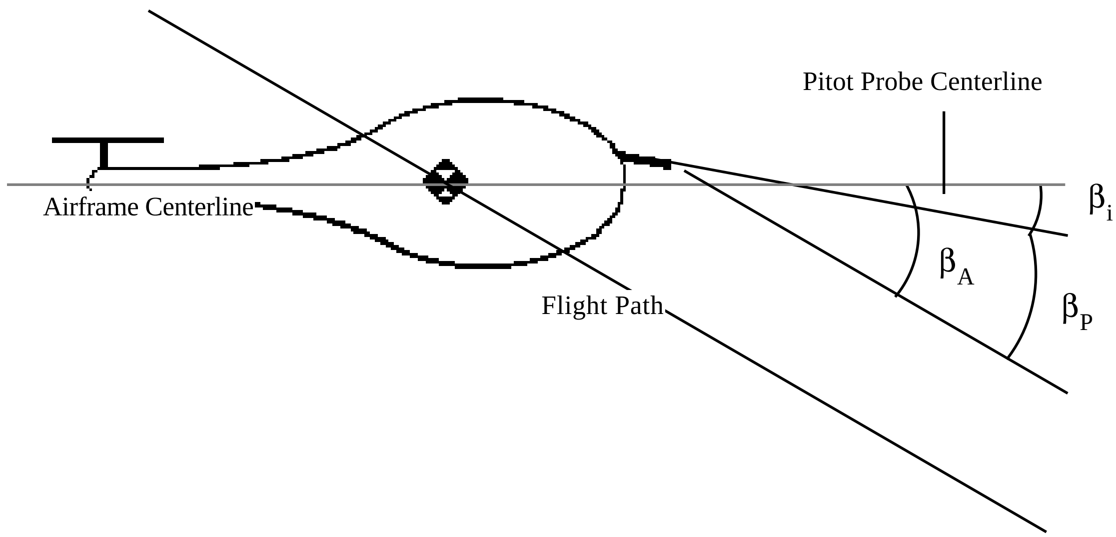


Figure 2.6
Probe Angle of Attack in the Horizontal (X - Y) Plane

The resultant probe inflow angle generally can be assumed an important characteristic if the angle is greater than 15 degrees. If a large angle of attack exists as the result of climbing flight or slow, high power level flight, a small change in probe sideslip angle, β_p, may have a significant effect on the airspeed position error. Slight yaw oscillations may cause the airspeed indicator to fluctuate at the same frequency as the yaw oscillations. Figure 2.7 illustrates the summation of probe angle of attack and sideslip angle to obtain the resultant probe inflow.

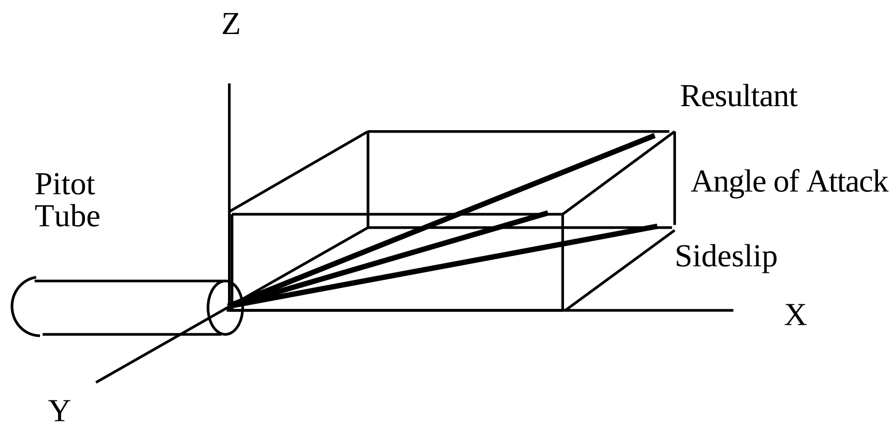


Figure 2.7
Resultant Probe Inflow

2.3.6 Pitot Tube Design

That part of the total pressure not sensed through the pitot tube is referred to as pressure defect, and is a function of angle of attack. However, it is also a function of Mach number and orifice diameter. As explained in NACA Report 919, the total pressure defect increases as the angle of attack or sideslip angle increases from zero, decreases as Mach number increases subsonically, and decreases as the ratio of orifice diameter to tube outside diameter increases. In general, if the ratio of orifice diameter to tube diameter is equal to one, the total pressure defect is zero up to angles of attack of 25 degrees. As the diameter ratio decreases to 0.74, the defect is still insignificant. But as the ratio of diameters decreases to 0.3, there is approximately a 5 percent total pressure defect at 15 degrees angle of attack, 12 percent at 20 degrees, and 22 percent at 25 degrees. For given values of orifice diameter and tube diameter, with an elongated nose shape (for example, semielliptical or ogival), the elongation is equivalent to an effective increase in the ratio of diameters and the magnitude of the total pressure defect will be less than is indicated above for a hemispherical head. These pitot tube design guidelines are general rules for accurate sensing of total pressure. All systems must be evaluated in flight test, but departure from these proven design parameters should prompt particular evaluator interest.

2.4 TEST METHODS AND TECHNIQUES

2.4.1 General

The calibration of pitot and static pressure systems can be accomplished by determining the errors in each system independently or simultaneously. The test methods for calibrating pitot and static systems are numerous and often a test is known by several different titles within the aviation industry. Often, more than one system requires calibration, such as separate pilot, copilot systems, and flight test (boom) pitot/static systems. It is especially important in these tests to have detailed knowledge of the particular system plumbing. Care must be taken to calibrate the "standard ship system". All too frequently a non-representative "flight test expedient" system is actually calibrated. The most common calibration techniques are listed below and will be discussed briefly. Individual instrument calibration should never be overlooked in these tests. Pitot/static systems should be leak checked prior to calibration test programs.

1. Measured Course.
2. Trailing Source.
3. Tower Fly-By.
4. Space Positioning.
5. Radar Altimeter.
6. Paced.

2.4.2 Measured Course

The measured course method is an airspeed (pitot/static system) calibration which requires that the helicopter be flown over a course of known length to determine true airspeed (V_T) from time and distance data. Calibrated airspeed calculated from true airspeed is compared to the indicated airspeed of the helicopter to obtain the airspeed position error. The conversion of true airspeed requires both the pressure altitude and free air temperature be recorded at each data point. Often the measured course method is used in conjunction with the tower fly-by method to obtain a low altitude static pressure error (altimeter position error) calibration for the entire range of calibrated airspeeds. In other cases, the entire airspeed position error is assumed to be due to static pressure error and thus a calculated altimeter error can be obtained. The validity of this test method is predicated on four equally important parameters:

1. Accuracy of elapsed time determination.
2. Accuracy of course measurement and course length.
3. A constant indicated airspeed over the course.
4. Wind conditions.

Measurement of elapsed time is of utmost importance and should be one of the first considerations when preparing for a test. Elapsed time can be measured with extremely accurate electronic devices. On the other end of the spectrum is the human observer with a stopwatch.

Flying a measured course requires a considerable amount of pilot effort to maintain a stabilized airspeed for a prolonged period of time in close proximity to the ground. The problems involved in this test are a function of the overall helicopter flying qualities and vary sharply with different aircraft. Averaging or integrating fluctuations in airspeed is not conducive to obtaining accurate results. It must be possible for the pilot to maintain flight with small airspeed variations for some finite period of time at any given airspeed. This period of time is generally short on the backside of the level flight power polar. An estimate of the maximum time periods that stable airspeeds can be maintained for the particular aircraft should be made to establish the optimum course length for the different airspeeds to be evaluated.

The course length must be measured accurately and clearly marked. The minimum measured course length recommended for use when a stopwatch is used to determine elapsed time is listed in Table 2.I. The course lengths are minimum and the pilot is required to fly over the course for a minimum of 10 seconds. A timing error, for example, of 0.5 second over a 10 second course represents a 5% error. Longer lengths will produce higher degrees of accuracy and should be used if a constant speed can be maintained.

Table 2.I
Minimum Recommended Measured Course Length

Test Airspeeds	Course Lengths
0 to 29 KTAS	500 feet
30 to 59 KTAS	1000 feet
60 to 89 KTAS	1500 feet
90 to 119 KTAS	2000 feet
120 to 149 KTAS	2500 feet
150 to 179 KTAS	3000 feet
180 to 210 KTAS	3500 feet

Ideally, winds should be calm when using the measured course. Data taken with winds can be corrected provided wind direction and velocity are constant. Wind (direction and velocity) data should be collected for each data point using calibrated sensitive equipment (usually portable) located on or very close to the ground speed course. In order to determine the no wind curve, runs should be made in both directions of the course (reciprocal headings). All runs must be flown on the course heading allowing the helicopter to drift with the wind as shown in Figure 2.8.

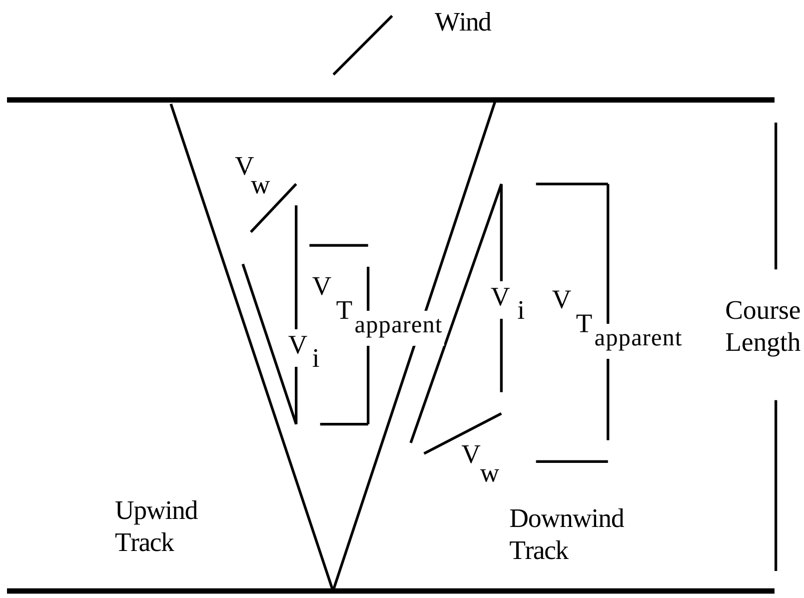


Figure 2.8
Wind Effect

True airspeed is determined by averaging the ground speeds. Calibrated airspeed is calculated using standard atmosphere relationships. This method of airspeed and altimeter system calibration assumes that the pitot errors are insignificant compared to static errors and is limited to level flight data point calibrations.

2.4.2.1 DATA REQUIRED

D, Δt, V_o, H_{P_o}, T_o, N_R, Q, GW, CG, α_o, β_o, $h_{a/c}$, h_{ref}, $H_{P_{o\,ref}}$.

Configuration

Wind information (azimuth and velocity).

Note: Velocity and altitude data must be recorded for each system to be calibrated.

Where:

D	- Course length
V_o	- Observed airspeed
$h_{a/c}$	- Aircraft height above ground
h_{ref}	- Reference altimeter height AGL
H_{P_o}	- Observed pressure altitude
$H_{P_{o\,ref}}$	- Reference observed pressure altitude
T_o	- Observed temperature
N_R	- Main rotor speed
Q	- Engine torque
GW	- Gross weight
CG	- Center of gravity
Δt	- Elapsed time
α_o	- Observed angle of attack
β_o	- Observed sideslip angle.

2.4.2.2 TEST CRITERIA

1. Balanced, wings level flight.
2. Constant aircraft heading.
3. Constant airspeed.
4. Constant altitude.
5. Constant wind speed and direction.

2.4.2.3 DATA REQUIREMENTS

1. Stabilize 10 s prior to course start.
2. Record data during course length.
3. V_o ± 0.5 kn.
4. H_{P_o} ± 20 ft over course length.
5. N_R ± 0.5%.

2.4.2.4 SAFETY CONSIDERATIONS/RISK MANAGEMENT

These tests are normally conducted in close proximity to the ground, and it is imperative that the flight crew maintain frequent visual contact with the ground. The extreme concentration required to fly accurate data points sometimes distracts the pilot from proper attention to outside references. Often these tests are conducted over highly uniform surfaces (water or dry lake bed courses) and thus produce a significant depth perception hazard.

2.4.3 Trailing Source

Total and static pressures can be measured by suspending a pitot/static source on a cable and comparing the results directly with the pitot and static systems installed in the test aircraft. These pressure sensings are transmitted through tubes to the helicopter where they are converted to accurate pressure altitudes and calibrated airspeeds by sensitive, calibrated instruments. Since the pressures from the source are transmitted through tubes to the helicopter for conversion to airspeeds and altitudes, no error is introduced by trailing the source below the helicopter. The airspeed and altimeter position error corrections for a given flight condition are determined by subtracting the source airspeed and altitude from the airspeed and altitude indicated by the ship's service system. The cable should be a minimum of 1-1/2 to 2 rotor diameters in length, have as small an outside diameter as practical, and have a rough exterior finish. The maximum speed of this test method may be limited to the speed at which the trailing source becomes unstable. Depending upon the frequency of the cable oscillation and the resultant maximum displacement of the towed source, large errors may be introduced into the towed source measurements. These errors are reflected as scatter and prevent the data from one flight being correlated with a second flight. In addition to the errors induced by the tube oscillations, a fin stabilized source, once disturbed, tends to fly by itself and may move up into the rotor wash. This situation becomes critical with respect to tail rotor clearance.

There are many configurations of sources such as a high drag cone afterbody or a fin stabilized cone, and some sources transmit electrical signals from transducers to recorders in the test aircraft. USNTPS is presently using a high drag cone afterbody configuration which has produced relatively high degrees of accuracy.

The test helicopter will be stabilized prior to recording a data point; however, the trailing source may or may not be stable when the data is recorded. Therefore, a means must be provided to monitor the trailing source. Generally, the stability can be determined in the cockpit if an auxiliary airspeed readout is provided. Thus, when using a trailing source, data should be recorded only when the aircraft and source are both stabilized in smooth air.

Since there is no known method of predicting the point of instability of these calibration systems, the trailing source should be monitored continuously and the flight should be planned to accomplish moderate speed level flight data points first and progress in a build-up fashion toward the higher speed points. Trailing source systems have been known to exhibit instabilities at both very low speeds and high speeds. Tail rotor to cable clearance is another important factor in calibrating low speed descending data points.

This method offers additional calibration opportunities over others in that pitot/static systems may be calibrated in climbs and descents as well as in level flight. An added advantage is pitot installation errors can be separated from static source installation errors. Trailing source systems should be leak checked each time that they are installed on the test aircraft.

2.4.3.1 DATA REQUIRED

V_o, $V_{o\ ref}$, H_{P_o}, $H_{P_{o\ ref}}$, T_o, N_R, Q, GW, CG, α_o, β_o, Configuration.

Note: Velocity and altitude data must be recorded for each system to be calibrated as well as the trailing source system (reference data).

2.4.3.2 TEST CRITERIA

1. Balanced, wings level flight.
2. Constant aircraft heading.
3. Constant airspeed.
4. Constant altitude (or unaccelerated climb/descent).
5. Steady indications on airspeed and altimeter systems.

2.4.3.3 DATA REQUIREMENTS

1. Stabilize 30 s prior to data record.
2. Record data for 15 s.
3. V_o ± 0.5 kn.
4. H_{P_o} ± 10 ft.
5. N_R ± 0.5%.

2.4.3.4 SAFETY CONSIDERATONS/RISK MANAGEMENT

Considerable flight and ground crew coordination is required to deploy and recover a trailing source system safely. Thorough planning and detailed pre-flight briefing is essential to ensure that each individual knows the proper procedure. Sometimes a chase aircraft helps by providing verbal instructions over the radio.

A hazard common to most helicopter tests is static electricity buildup. The trailing source should be allowed to touch the ground before ground crewmen touch the device, unless other static grounding provisions are available.

Pitot/static source instability stories are too numerous to recount. When these devices exhibit unstable tendencies is difficult to predict. Such factors as probe design, cable length, airspeed, rotor wake impingement, aircraft vibration levels and atmospheric turbulence influence the onset of these instabilities. Trailing source devices should be monitored (preferably visually) at all times. Chase aircraft or on-board crewmen normally accomplish this function. Under most circumstances, the onset of the instability is of sufficiently low frequency and amplitude that corrective action can be taken. In the event that the probe starts to exhibit unstable behavior, always return to a flight condition which was previously satisfactory (such as slow down, roll out or level off). Should the instability grow to hazardous proportions (nearing contact with rotor systems or fuselage), provisions for device jettison must be provided. Jettison devices vary in complexity but should be ground checked by the flight crew to ensure complete familiarity with procedures and proper mechanical operation.

Local unit operating procedures may require the use of parachutes for flight crew members on these missions. Overflight of populated areas should be avoided.

2.4.4 Tower Fly-By

This method is a simple and excellent way to determine accurately static system error. A tall tower of known height is required as an observation point. The free stream static pressure can be established in any number of ways (such as a sensitive calibrated altimeter in the tower) and should be recorded for each pass of the test aircraft. The test aircraft is flown down a predetermined path passing at a known distance (d) from the tower (Figure 2.9). Any deviation in the height of the aircraft (Δh) can be determined by visual observation or by photography and simple geometry. The simplicity of this method allows a large number of accurate data points to be recorded quickly and inexpensively. The major limitations of this method are that it assumes that all of the position error is attributed to the static source error (i.e., no pitot source error) and only level flight calibrations can be performed.

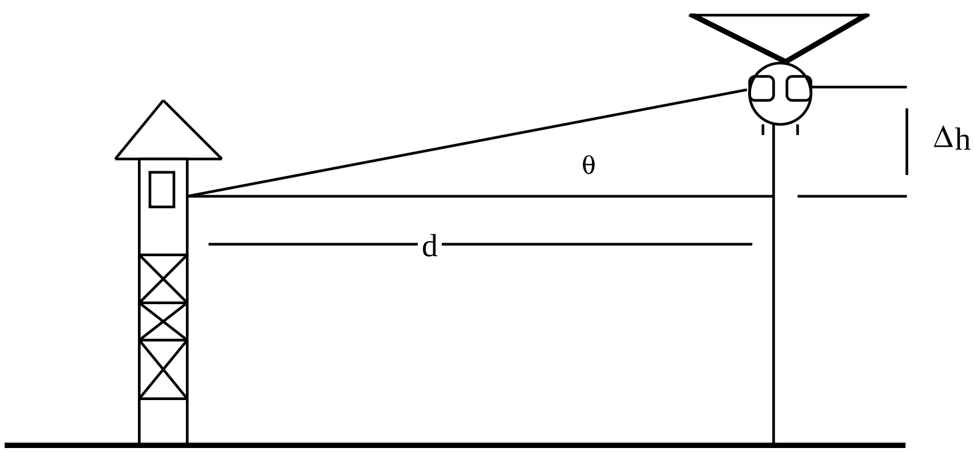


Figure 2.9
Tower Fly-By

It is important to ensure that there is no false position error introduced during this test as a result of instrument calibration errors. Prior to the flight, with the test aircraft in a static condition, the instrument to be used to establish the free air static pressure in the tower is placed next to the aircraft test instrumentation. With both of these instruments in the same environment (and with their respective instrument corrections applied), the indications should be the same. If there is a discrepancy, the difference in readings should be included in the data reduction.

2.4.4.1 DATA REQUIRED

V_o, $V_{o\,ref}$, $H_{P_{o\,twr}}$, T_o, N_R, Q, GW, CG, α_o, β_o, Configuration.

Note: Velocity and altitude data must be recorded for each system to be calibrated as well as tower reference pressure altitude, temperature and aircraft geometric height data (d, θ).

2.4.4.2 TEST CRITERIA

1. Balanced, wings level flight.
2. Constant heading, track over predetermined path.
3. Constant airspeed.
4. Constant altitude.

2.4.4.3 DATA REQUIREMENTS

1. Stabilize 15 s prior to abeam tower.
2. V_o ± 0.5 kn.
3. H_{P_o} ± 10 ft.
4. N_R ± 0.5%.
5. Ground track ± 1% of stand-off distance.

2.4.4.4 SAFETY CONSIDERATONS/RISK MANAGEMENT

This test procedure requires considerable pilot concentration. Attention can be focused easily on aircraft instruments without proper regard to terrain clearance. Close proximity to the ground and divided attention is not a good combination, but is an unfortunate necessity to accomplish this procedure. Complete familiarity with aircraft normal and emergency operating procedures can prevent excessive pilot distractions. Team work and well briefed/rehearsed data collection procedures can minimize these dangers.

Most courses of this type are over highly uniform terrain such as water or dry lake beds. This situation contributes to poor depth perception. A second crew member or ground safety observer can share backup altitude monitoring duties.

2.4.5 Space Positioning

Space positioning systems vary widely with respect to principle of operation. Automatic or manual optical tracking systems, radar tracking, and radio ranging systems fall into this category. A space positioning arrangement generally employs at least three tracking stations. These tracking stations track the test aircraft by radar lock on or by a manual/visual sight arrangement. Depending upon the configuration, angular and linear displacements are recorded and through a system of triangulation, a computer solution of tapeline altitude and ground speed is obtained. The accuracy of the data can be increased by compensating for tracking errors developed as a result of the random drift (away from a prearranged target point on the aircraft). Accuracy can be improved by using an on-board transponder to enhance the tracking process. Regardless of the tracking method, raw data is reduced using high speed computer programs to provide position, velocity, and acceleration information. Normally the test aircraft is flown over a prearranged course to provide the station with a good target and the optimum tracking angles.

Depending on the accuracy desired and the existing wind conditions, balloons can be released and tracked to determine wind velocity and direction. Wind information can then be fed into the computer for each data point and true airspeed determined. The true airspeed is used to determine calibrated airspeed and airspeed position error.

Many types of systems exist at various facilities, both contractor and government controlled. Most of these facilities have conducted extensive test programs to verify the accuracy of the output data that can be made available. These data packages should be examined closely to ensure that program requirements can be met with this type equipment.

The use of space positioning systems requires detailed planning and coordination. Exact correlation between onboard and ground recording systems is essential. These systems are generally in high demand by programs competing for resource availability and priority. Due to the inherent complexities of hardware and software, expensive price tags are always associated with this technique. The great value of this method is that large amounts of data can be obtained in a short amount of time. Another aspect of these systems that makes them attractive is the wide variety of flight conditions that can be calibrated, such as climbs and descents. Accelerating and decelerating maneuvers can be time correlated if the data systems are synchronized.

2.4.5.1 DATA REQUIRED

V_o, H_{P_o}, T_o, N_R, Q, GW, CG, α_o, β_o, Configuration.

Wind direction and velocity.

Note: Velocity and altitude data must be recorded for each system to be calibrated.

2.4.5.2 TEST CRITERIA

1. Balanced, wings level flight (or stabilized turn).
2. Constant aircraft heading (or stabilized turn).
3. Constant airspeed (or stable rate of change during accel/decel maneuvers).
4. Constant altitude (or stable rate of change during climbs and descents).
5. Constant wind speed and direction.

2.4.5.3 DATA REQUIREMENTS

1. Stabilized 30 s prior to data record.
2. Record data for 15 s.
3. V_o ± 0.5 kn.
4. H_{P_o} ± 10 ft.
5. N_R ± 0.5%.

2.4.5.4 SAFETY CONSIDERATONS/RISK MANAGEMENT

Knowledge of and adherence to normal and emergency operating procedures and limitations.

2.4.6 Radar Altimeter

A calibrated radar altimeter can be used to determine altimeter position errors present in level flight at low altitude. A level runway is required to establish a reference altitude for the test. A sensitive pressure altimeter should be placed at the runway elevation during test runs. The radar height is added to the runway pressure altitude and compared with the aircraft altimeter system indication to determine position errors. The aircraft altimeter may be used as the reference altimeter if the helicopter is landed on the runway reference point and the rotor disengaged. With the rotor not engaged, the service system altimeter reading before and after the test runs can be used to establish a base pressure altitude to which the radar height can be added. This method is recommended only for rough approximation of gross altimeter position errors. Flight above the level surface should be conducted low enough to provide accurate height information with the radar altimeter, but not so low that the pressure field around the helicopter will be affected by ground proximity. A good guideline is generally two main rotor diameters of altitude.

2.4.6.1 DATA REQUIRED

Surface pressure altitude, V_o, H_{P_o}, T_o, N_R, Q, GW, CG, α_o, β_o, radar altitude. Configuration.

2.4.6.2 TEST CRITERIA

1. Balanced, wings level flight.
2. Constant aircraft heading.
3. Constant airspeed.
4. Constant radar altimeter altitude.

2.4.6.3 DATA REQUIREMENTS

1. Stabilized flight for 15 s.
2. $V_o \pm 0.5$ kn.
3. $H_{P_o} \pm 10$ ft.
4. Radar altimeter altitude ± 3 ft.

2.4.6.4 SAFETY CONSIDERATONS/RISK MANAGEMENT

The safety considerations relevant to this test are similar to those discussed for the measured course method.

2.4.7 Paced

The use of a pace calibration system offers many advantages and is used frequently in obtaining pitot/static position error calibrations. Very low speed calibrations can be obtained using ground vehicles for establishing reference speed. More commonly, specially outfitted and calibrated pace aircraft are used for this purpose. These pace aircraft generally have expensive, specially designed, separate pitot/static systems which have been extensively calibrated with the most precise methods available. These calibrations are kept current and periodically cross-checked on low speed courses or with space positioning equipment. These pace calibration aircraft systems, when properly maintained and documented, offer the tester a method of obtaining very accurate position error information (pitot and static) for a wide variety of flight conditions (level flight, climbs, descents) in a short period of time.

This test method requires precise formation flying. The pace aircraft can be lead or trail, depending on individual preference. The lead aircraft establishes the test point by stabilizing on the desired data point. The aircraft required to maintain formation position must stay close enough to the lead aircraft to detect minor relative speed differences but far enough away to prevent pitot/static interference between aircraft systems and/or compromising flight safety. Experience has shown that good results can be obtained by maintaining a left or right echelon (approximately 45 degrees) formation at a distance of approximately 1-1/2 to 2 rotor discs. It is also important to fly in formation such that the airspeed and altimeter instruments of both aircraft are at the same elevation. A common method used to accomplish this is to fly formation such that the lead aircraft pilot's helmet is on the horizon as observed by the second aircraft pilot.

Leak checks of all pitot/static systems, including the pace aircraft systems, is required prior to and following calibration flights. Many pace systems provide two separate calibration systems to further guarantee data accuracy.

2.4.7.1 DATA REQUIRED

V_o, H_{P_o}, T_o, N_R, Q, GW, CG, α_o, β_o, Configuration.

Note: Velocity and altitude data must be recorded for each system to be calibrated as well as pace aircraft pitot/static data.

2.4.7.2 TEST CRITERIA

1. Balanced, wings level flight.
2. Constant heading.
3. Constant airspeed.
4. Constant altitude (or stable rate of change during climbs and descents).
5. No relative motion between the test and pace aircraft.

2.4.7.3 DATA REQUIREMENTS

1. Stabilize 30 s prior to data collection.
2. Record data for 15 s.
3. V_o ± 0.5 kn.
4. H_{P_o} ± 10 ft.
5. N_R ± 0.5%.

2.4.7.4 SAFETY CONSIDERATONS/RISK MANAGEMENT

All the hazards of close formation flying are present with the pace calibration method. To reduce this risk, frequent practice and teamwork are required. In addition to the workload present with formation flying, considerable attention must be directed toward accurately reading altimeter, airspeed, and other data instruments in order to document each data point. These increased risk factors may be mitigated by providing for some automatic data collection and/or additional crew members to collect the necessary information while the pilot directs his attention to conducting safe formation flight. Again, careful planning, a good mission briefing and professional execution of the plan significantly decrease the inherent risks associated with these tests. All crew members should be required to wear parachutes. Minimum safe calibration altitudes should provide sufficient allowance for bailout. All members of the flight crews should be alert to the potential for mid-air collision and a flight break away procedure should be established. Minimum visual flight conditions should be established in the planning phase (such as 3 miles visibility and 1000 foot cloud clearance), and inadvertent instrument meteorological condition break-up procedures should be established. Pilots of both aircraft must be aware of the minimum control airspeeds for their particular aircraft in each of the various configurations required during the test and some buffer should be provided for formation maneuvering. As in all flight tests, normal and emergency operating procedures and limitations must be known intimately. Upon completion of tests, a method of flight break-up should be planned. At no time during these tests should there be a question in anyone's mind who is responsible for providing safe distance between aircraft. When this responsibility changes (i.e., flight lead changes), it should be so stated and acknowledged. The aircraft charged with the responsibility for safe stand-off distance must never lose sight of the lead aircraft. In the event that visual contact is lost, confess the error so that both crews can watch and avoid.

2.5 DATA REDUCTION

2.5.1 General

In general, reduction of pitot/static system calibration data falls into two categories: 1) Direct comparison between observed quantities and reference (truth) quantities, and 2) Altitude depression method, where all the pitot/static error is assumed to be at the static source and altimeter as well as airspeed errors are computed based on this assumption. A data reduction format will be given for several representative and most often used test methods which were discussed in this chapter.

2.5.2 Measured Course

The following equations are used in the measured course data reduction.

$$H_{P_{ic}} = H_{P_o} + \Delta H_{P_{ic}} \qquad \textit{eq 2.12}$$

Where:

$H_{P_{ic}}$	- Instrument corrected pressure altitude
H_{P_o}	- Observed pressure altitude
$\Delta H_{P_{ic}}$	- Altimeter instrument correction.

$$H_{P_{ic\,ref}} = H_{P_{o\,ref}} + \Delta H_{P_{ic\,ref}} \qquad \textit{eq 2.13}$$

Where:

$H_{P_{ic\,ref}}$	- Reference instrument corrected pressure altitude
$H_{P_{o\,ref}}$	- Reference observed pressure altitude
$\Delta H_{P_{ic\,ref}}$	- Reference altimeter instrument correction.

$$H_{P_c} = H_{P_{ic\,ref}} - h_{ref} + h_{a/c} \qquad \textit{eq 2.14}$$

Where:

H_{P_c}	- Calibrated pressure altitude
$H_{P_{ic\,ref}}$	- Reference instrument corrected pressure altitude
h_{ref}	- Reference altimeter height AGL
$h_{a/c}$	- Aircraft height above ground.

$$\Delta H_{pos} = H_{P_c} - H_{P_{ic}} \qquad \textit{eq 2.15}$$

Where:

ΔH_{pos}	- Altimeter position error
H_{P_c}	- Calibrated pressure altitude
$H_{P_{ic}}$	- Instrument corrected pressure altitude.

$$V_{ic} = V_o + \Delta V_{ic} \qquad \textit{eq 2.16}$$

Where:

V_{ic}	- Instrument corrected airspeed
V_o	- Observed airspeed
ΔV_{ic}	- Airspeed instrument correction.

$$V_{G1} = 0.5917\left(\frac{D}{\Delta t_1}\right) \qquad \textit{eq 2.17}$$

$$V_{G2} = 0.5917\left(\frac{D}{\Delta t_2}\right) \qquad \textit{eq 2.18}$$

Where:

V_G	- Ground speed
D	- Course length
Δt	- Elapsed time.

$$V_T = \frac{V_{G1} + V_{G2}}{2} \qquad \textit{eq 2.19}$$

Where:

V_T	- True airspeed
V_G	- Ground speed.

$$T_a = T_o + \Delta T_{ic} \qquad \textit{eq 2.20}$$

Where:

T_a	- Ambient temperature
T_o	- Observed temperature
ΔT_{ic}	- Temperature instrument correction.

$$V_c \cong \frac{V_T}{\sqrt{\sigma}} \qquad \textit{eq 2.21}$$

Where:

V_c	- Calibrated airspeed
V_T	- True airspeed
σ	- Density ratio.

$$\Delta V_{pos} = V_c - V_{ic} \qquad \textit{eq 2.22}$$

Where:

ΔV_{pos} - Airspeed position error

V_c - Calibrated airspeed

V_{ic} - Instrument corrected airspeed.

$$\alpha_{ic} = \alpha_o + \Delta\alpha_{ic} \qquad \textit{eq 2.23}$$

Where:

α_{ic} - Instrument corrected angle of attack

α_o - Observed angle of attack

$\Delta\alpha_{ic}$ - Angle of attack instrument correction.

$$\beta_{ic} = \beta_o + \Delta\beta_{ic} \qquad \textit{eq 2.24}$$

Where:

β_{ic} - Instrument corrected sideslip angle

β_o - Observed sideslip angle

$\Delta\beta_{ic}$ - Sideslip angle instrument correction.

From the observed airspeed and altitude data, compute V_c, V_{ic}, ΔH_{pos}, ΔV_{pos}, α_{ic}, β_{ic} as follows:

Parameter	Notation	Formula	Units	Remarks
(1) Observed pressure altitude	H_{P_o}		ft	Set altimeter 29.92 in Hg
(2) Altimeter instrument correction	$\Delta H_{P_{ic}}$		ft	Lab calibration
(3) Instrument corrected pressure altitude	$H_{P_{ic}}$	eq 2.12	ft	

(4) Aircraft height above ground	$h_{a/c}$		ft	From photograph or other measurement technique
(5) Reference observed pressure altitude	$H_{P_{o\ ref}}$		ft	Near center of course (29.92 in Hg)
(6) Reference altimeter instrument correction	$\Delta H_{P_{ic\ ref}}$		ft	Lab calibration
(7) Reference instrument corrected pressure altitude	$H_{P_{ic\ ref}}$	eq 2.13	ft	
(8) Reference altimeter height above ground	h_{ref}		ft	Known height
(9) Calibrated pressure altitude	H_{P_c}	eq 2.14	ft	
(10) Altimeter position error	ΔH_{pos}	eq 2.15	ft	Correction to be added
(11) Observed airspeed	V_o		KOAS	
(12) Airspeed instrument correction	ΔV_{ic}		kn	Lab calibration
(13) Instrument corrected airspeed	V_{ic}	eq 2.16	KIAS	
(14) Elapsed time$_1$	Δt_1		s	One direction
(15) Elapsed time$_2$	Δt_2		s	Reciprocal direction
(16) Course length	D		ft	Known distance
(17) Ground speed$_1$	V_{G1}	eq 2.17	kn	One direction
(18) Ground speed$_2$	V_{G2}	eq 2.18	kn	Reciprocal direction
(19) True airspeed	V_T	eq 2.19	KTAS	

(20) Observed temperature	T_o		°C	Ground reading
(21) Temperature instrument correction	ΔT_{ic}		°C	Lab calibration
(22) Ambient temperature	T_a	eq 2.20	°C	No recovery or Mach correction required
(23) Density altitude	H_D		ft	From tables or calculated using(22) and (9)
(24) Square root of the density ratio	$\sqrt{\sigma}$			From tables or calculated using (23)
(25) Calibrated airspeed	V_c	eq 2.21	KCAS	Assume V_e=V_c
(26) Airspeed position error	ΔV_{pos}	eq 2.22	kn	Correction to be added
(27) Observed angle of attack	α_o		deg	
(28) Angle of attack instrument correction	$\Delta\alpha_{ic}$		deg	Lab calibration
(29) Instrument corrected angle of attack	α_{ic}	eq 2.23	deg	
(30) Observed sideslip angle	β_o		deg	
(31) Sideslip angle instrument correction	$\Delta\beta_{ic}$		deg	Lab calibration
(32) Instrument corrected sideslip angle	β_{ic}	eq 2.24	deg	

Plot instrument corrected airspeed versus calibrated airspeed and airspeed position error as shown in Figure 2.10.

Plot instrument corrected airspeed versus altimeter position error as shown in Figure 2.11.

Plot instrument corrected angle of attack versus airspeed position error and altimeter position error as shown in Figure 2.12.

Plot instrument corrected sideslip angle versus airspeed position error and altimeter position error as shown in Figure 2.13.

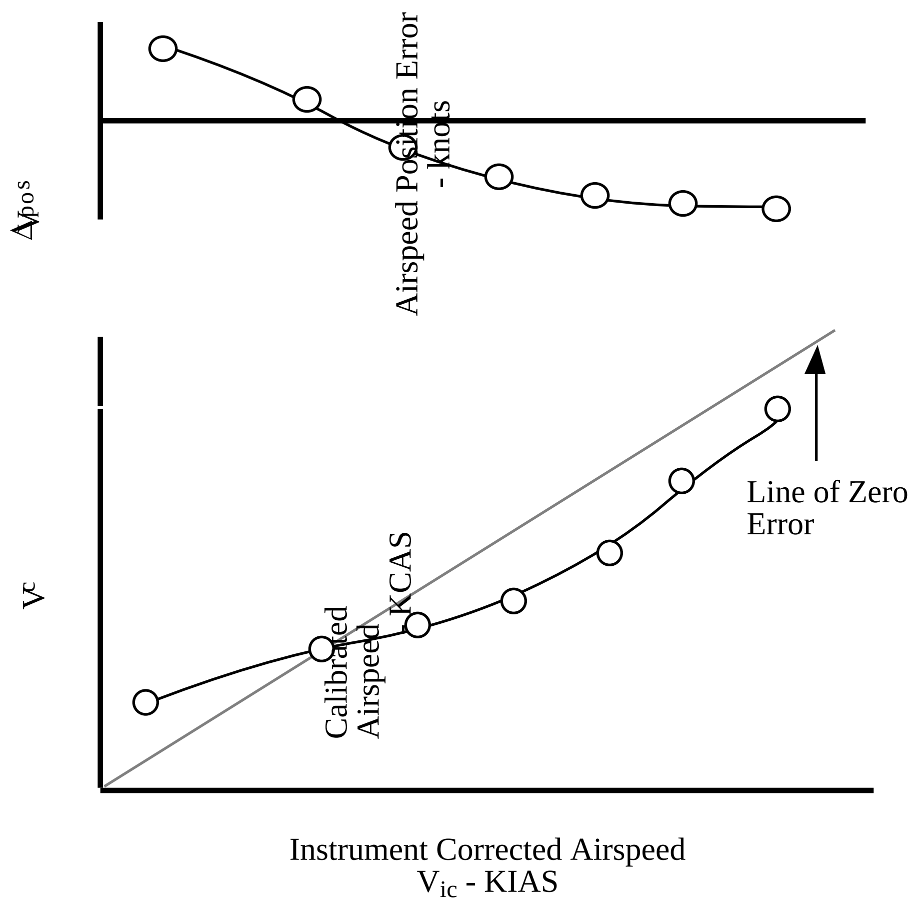


Figure 2.10
Airspeed System Calibration

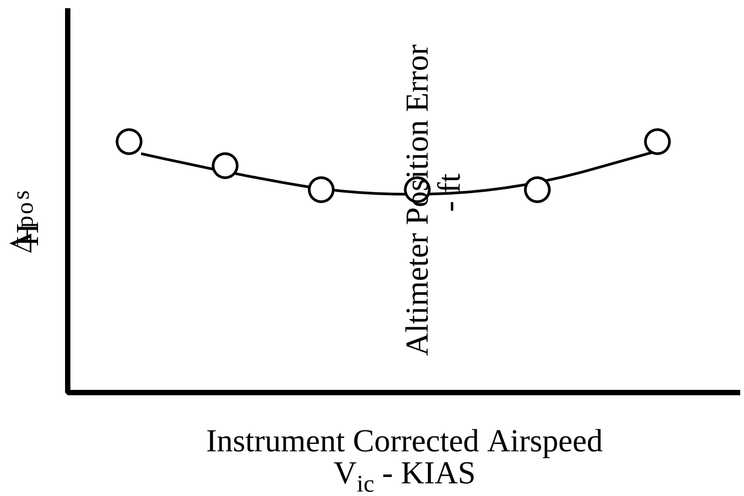


Figure 2.11
Altimeter System Calbiration

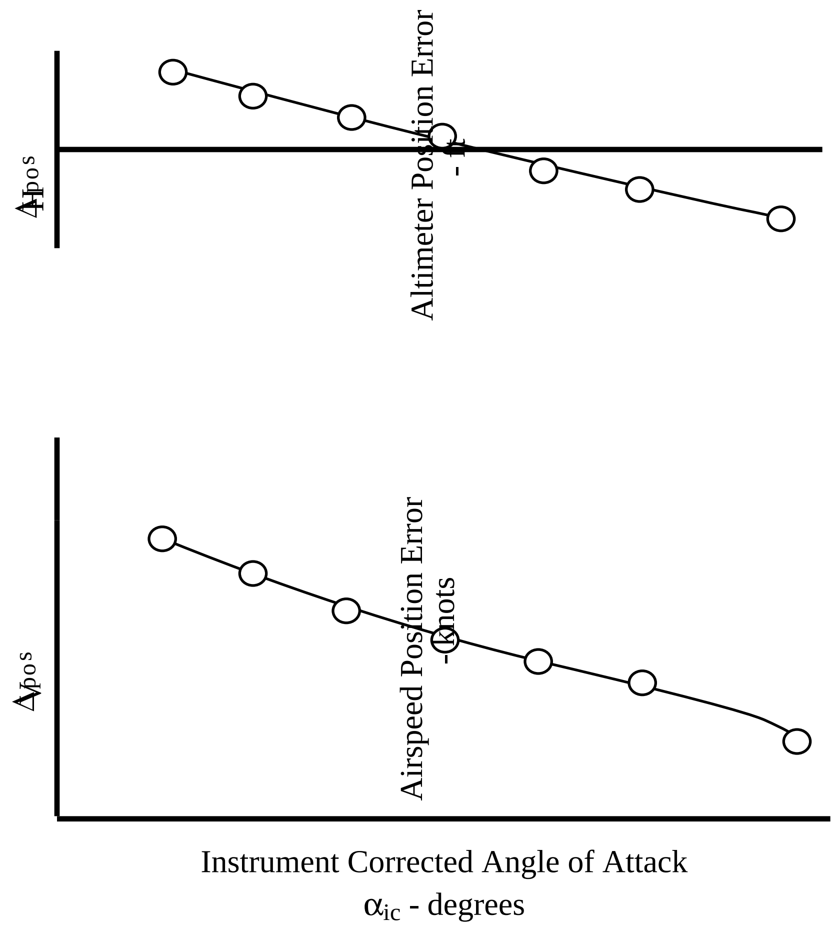


Figure 2.12
Altimeter and Airspeed Position Errors Due to Angle of Attack

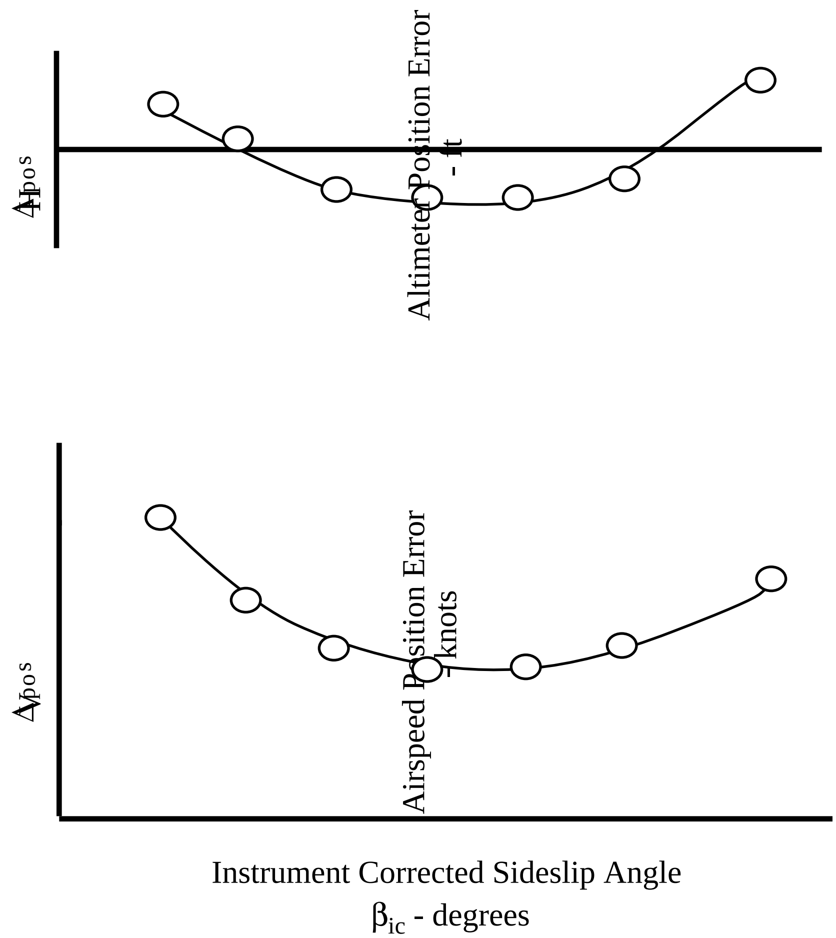


Figure 2.13
Altimeter and Airspeed Position Errors Due to Angle of Sideslip

2.5.3 Trailing Source/Paced

The following equations are used in the trailing source/paced data reduction.

$$H_{P_c} = H_{P_{o\,ref}} + \Delta H_{P_{ic\,ref}} + \Delta H_{pos\,ref} \qquad \textit{eq 2.25}$$

Where:

H_{P_c}	- Calibrated pressure altitude
$H_{P_{o\,ref}}$	- Reference observed pressure altitude
$\Delta H_{P_{ic\,ref}}$	- Reference altimeter instrument correction
$\Delta H_{pos\,ref}$	- Reference altimeter position error.

$$H_{P_{ic\ ship}} = H_{P_{o\ ship}} + \Delta H_{P_{ic\ ship}}$$ *eq 2.26*

Where:

$H_{P_{ic\ ship}}$	- Ship instrument corrected pressure altitude
$H_{P_{o\ ship}}$	- Ship observed pressure altitude
$\Delta H_{P_{ic\ ship}}$	- Ship altimeter instrument correction.

$$\Delta H_{pos\ ship} = H_{P_c} - H_{P_{ic\ ship}}$$ *eq 2.27*

Where:

$\Delta H_{pos\ ship}$	- Ship altimeter position error
H_{P_c}	- Calibrated pressure altitude
$H_{P_{ic\ ship}}$	- Ship instrument corrected pressure altitude.

$$V_c = V_{o\ ref} + \Delta V_{ic\ ref} + \Delta V_{pos\ ref}$$ *eq 2.28*

Where:

V_c	- Calibrated airspeed
$V_{o\ ref}$	- Reference observed airspeed
$\Delta V_{ic\ ref}$	- Reference airspeed instrument correction
$\Delta V_{pos\ ref}$	- Reference airspeed position error.

$$V_{ic\ ship} = V_{o\ ship} + \Delta V_{ic\ ship}$$ *eq 2.29*

Where:

$V_{ic\ ship}$	- Ship instrument corrected airspeed
$V_{o\ ship}$	- Ship observed airspeed
$\Delta V_{ic\ ship}$	- Ship airspeed instrument correction.

$$\Delta V_{pos\ ship} = V_c - V_{ic\ ship}$$ *eq 2.30*

Where:

$\Delta V_{pos\ ship}$	- Ship airspeed position error
V_c	- Calibrated airspeed
$V_{ic\ ship}$	- Ship instrument corrected airspeed.

$$T_a = T_o + \Delta T_{ic} \qquad \textit{eq 2.20}$$

$$\alpha_{ic} = \alpha_o + \Delta\alpha_{ic} \qquad \textit{eq 2.23}$$

$$\beta_{ic} = \beta_o + \Delta\beta_{ic} \qquad \textit{eq 2.24}$$

From the observed airspeed and pressure altitude data, compute V_c, V_{ic}, ΔH_{pos}, ΔV_{pos}, α_{ic}, β_{ic} as follows:

Parameter	Notation	Formula	Units	Remarks
(1) Reference observed pressure altitude	$H_{P_{o\ ref}}$		ft	Set altimeter 29.92 in Hg
(2) Reference altimeter instrument correction	$\Delta H_{P_{ic\ ref}}$		ft	Lab calibration
(3) Reference altimeter position error	$\Delta H_{pos\ ref}$		ft	From reference system calibration
(4) Calibrated pressure altitude	H_{P_c}	eq 2.25	ft	
(5) Ship observed pressure altitude	$H_{P_{o\ ship}}$		ft	Set altimeter 29.92 in Hg
(6) Ship altimeter instrument correction	$\Delta H_{P_{ic\ ship}}$		ft	Lab calibration
(7) Ship instrument corrected pressure altitude	$H_{P_{ic\ ship}}$	eq 2.26	ft	
(8) Ship altimeter position error	$\Delta H_{pos\ ship}$	eq 2.27	ft	Correction to be added

(9) Reference observed airspeed	$V_{o\ ref}$		KOAS	
(10) Reference airspeed instrument correction	$\Delta V_{ic\ ref}$		kn	Lab calibration
(11) Reference airspeed position error	$\Delta V_{pos\ ref}$		kn	From reference system calibration
(12) Calibrated airspeed	V_c	eq 2.28	KCAS	
(13) Ship observed airspeed	$V_{o\ ship}$		KOAS	
(14) Ship airspeed instrument correction	$\Delta V_{ic\ ship}$		kn	Lab calibration
(15) Ship instrument corrrected airspeed	$V_{ic\ ship}$	eq 2.29	KIAS	
(16) Ship airspeed position error	$\Delta V_{pos\ ship}$	eq 2.30	kn	Correction to be added
(17) Observed temperature	T_o		°C	(Reference)
(18) Temperature instrument correction	ΔT_{ic}		°C	(Reference) Lab calibration
(19) Ambient temperature	T_a	eq 2.20	°C	Instrument correction, ignore recovery and Mach correction
(20) Density altitude	H_D		ft	From tables or calculated using (19) and (4)
(21) Observed angle of attack	α_o		deg	
(22) Angle of attack instrument correction	$\Delta\alpha_{ic}$		deg	Lab calibration
(23) Instrument corrected angle of attack	α_{ic}	eq 2.23	deg	

(24) Observed sideslip angle	β_o		deg	
(25) Sideslip angle instrument correction	$\Delta\beta_{ic}$		deg	Lab calibration
(26) Instrument corrected sideslip angle	β_{ic}	eq 2.24	deg	

Plot instrument corrected airspeed versus calibrated airspeed and airspeed position error as shown in Figure 2.10.

Plot instrument corrected airspeed versus altimeter position error as shown in Figure 2.11.

Plot instrument corrected angle of attack versus airspeed position error and altimeter position error as shown in Figure 2.12.

Plot instrument corrected sideslip angle versus airspeed position error and altimeter position error as shown in Figure 2.13.

2.5.4 Tower Fly-By

The following equations are used in the tower fly-by data reduction.

$$H_{P_{ic\ ship}} = H_{P_{o\ ship}} + \Delta H_{P_{ic\ ship}} \qquad \textit{eq 2.26}$$

$$V_{ic} = V_o + \Delta V_{ic} \qquad \textit{eq 2.16}$$

$$H_{P_{c\ twr}} = H_{P_{o\ twr}} + \Delta H_{P_{ic\ twr}} \qquad \textit{eq 2.31}$$

Where:

$H_{P_{c\ twr}}$	- Tower calibrated pressure altitude
$H_{P_{o\ twr}}$	- Tower observed pressure altitude
$\Delta H_{P_{ic\ twr}}$	- Tower altimeter instrument correction.

$$\Delta h = d \tan\theta \qquad \textit{eq 2.32}$$

Where:

Δh	- Aircraft height above tower
d	- Horizontal distance
θ	- Angle.

$$H_{P_c} = H_{P_{ctwr}} + \Delta h \qquad \textit{eq 2.33}$$

Where:

H_{P_c}	- Calibrated pressure altitude
$H_{P_{c\,twr}}$	- Tower calibrated pressure altitude
Δh	- Aircraft height above tower.

$$\Delta H_{pos\ ship} = H_{P_c} - H_{P_{ic\ ship}} \qquad \textit{eq 2.27}$$

$$\Delta P_{s\ ship} = P_{s\ ref} - P_{s_{ic\ ship}} \qquad \textit{eq 2.34}$$

Where:

$\Delta P_{s\ ship}$	- Ship static pressure error
$P_{s\ ref}$	- Reference static pressure
$P_{s_{ic\ ship}}$	- Ship instrument corrected static pressure.

$$T_a = T_o + \Delta T_{ic} \qquad \textit{eq 2.20}$$

$$T_a\ (°K) = T_a\ (°C) + 273.15 \qquad \textit{eq 2.35}$$

Where:

T_a	- Ambient temperature.

$$a = 38.9678 \sqrt{T_a\ (°K)} \qquad \textit{eq 2.36}$$

Where:

a	- Speed of sound
T_a	- Ambient temperature.

$$\Delta V_{pos} = \left[\frac{\Delta P_{s\ ship}\left(a^2\right)}{41.888\ \left(V_{ic}\right)}\right]\left[\frac{2.5}{1 + 0.2\left(\frac{V_{ic}}{a}\right)^2}\right] \qquad \textit{eq 2.37}$$

Where:

ΔV_{pos}	- Airspeed position error
$\Delta P_{s\ ship}$	- Ship static pressure error
a	- Speed of sound
V_{ic}	- Instrument corrected airspeed.

$$V_c = V_{ic} + \Delta V_{pos} \qquad \textit{eq 2.38}$$

Where:

V_c	- Calibrated airspeed
V_{ic}	- Instrument corrected airspeed
ΔV_{pos}	- Airspeed position error.

$$\alpha_{ic} = \alpha_o + \Delta\alpha_{ic} \qquad \textit{eq 2.23}$$

$$\beta_{ic} = \beta_o + \Delta\beta_{ic} \qquad \textit{eq 2.24}$$

From the observed airspeed, pressure altitude and tower data, compute V_c, V_{ic}, ΔH_{pos}, ΔV_{pos}, α_{ic}, β_{ic} as follows:

Parameter	Notation	Formula	Units	Remarks
(1) Ship observed pressure altitude	$H_{P_{o\ ship}}$		ft	Set altimeter 29.92 in Hg
(2) Ship altimeter instrument correction	$\Delta H_{P_{ic\ ship}}$		ft	Lab calibration
(3) Ship instrument corrected pressure altitude	$H_{P_{ic\ ship}}$	eq 2.26	ft	
(4) Observed airspeed	V_o		KOAS	
(5) Airspeed instrument correction	ΔV_{ic}		kn	Lab calibration
(6) Instrument corrected airspeed	V_{ic}	eq 2.16	KIAS	
(7) Tower observed pressure altitude	$H_{P_{o\ twr}}$		ft	29.92 in Hg
(8) Tower altimeter instrument correction	$\Delta H_{P_{ic\ twr}}$		ft	Lab calibration
(9) Tower calibrated pressure altitude	$H_{P_{c\ twr}}$	eq 2.31	ft	
(10) Horizontal distance from tower to flight path	d		ft	Known distance
(11) Theodolite angle	θ		deg	Photographically measured
(12) Aircraft height above tower	Δh	eq 2.32	ft	

(13) Calibrated pressure altitude	H_{P_c}	eq 2.33	ft	
(14) Ship altimeter position error	$\Delta H_{pos\ ship}$	eq 2.27	ft	
(15) Reference static pressure	$P_{s\ ref}$		in Hg	From tables or calculated using (13)
16) Ship instrument corrected static pressure	$P_{s_{ic\ ship}}$		in Hg	From tables or calculated using (3)
(17) Ship static pressure error	$\Delta P_{s\ ship}$	eq 2.34	in Hg	
(18) Observed temperature	T_o		°C	(Tower)
(19) Temperature instrument correction	ΔT_{ic}		°C	Lab calibration
(20) Ambient temperature	T_a	eq 2.20	°C	Instrument correction, ignore recovery and Mach correction
(21) Ambient temperature	T_a	eq 2.35	°K	Conversion
(22) Speed of sound	a	eq 2.36	kn	
(23) Airspeed position error	ΔV_{pos}	eq 2.37	kn	
(24) Calibrated airspeed	V_c	eq 2.38	KCAS	
(25) Observed angle of attack	α_o		deg	
(26) Angle of attack instrument correction	$\Delta\alpha_{ic}$		deg	Lab calibration
(27) Instrument corrected angle of attack	α_{ic}	eq 2.23	deg	

(28) Observed sideslip angle	β_o		deg	
(29) Sideslip angle instrument correction	$\Delta\beta_{ic}$		deg	Lab calibration
(30) Instrument corrected sideslip angle	β_{ic}	eq 2.24	deg	

Plot instrument corrected airspeed versus calibrated airspeed and airspeed position error as shown in Figure 2.10.

Plot instrument corrected airspeed versus altimeter position error as shown in Figure 2.11.

Plot instrument corrected angle of attack versus airspeed position error and altimeter position error as shown in Figure 2.12.

Plot instrument corrected sideslip angle versus airspeed position error and altimeter position error as shown in Figure 2.13.

2.6 DATA ANALYSIS

Analysis of pitot/static system performance data is done primarily graphically. Superposition of the specification limits (Paragraph 2.8) on the data plots discussed in paragraph 2.5 quickly identifies problems with system performance. Care must be taken to group data for presentation. All level flight data should be plotted separately from other flight condition data. Alternate configurations must also be evaluated when applicable. Within each grouping of airspeed and altimeter calibration curves, one should be alert for discontinuities, large errors and trends. Error tolerances will be discussed relative to mission suitability and specification compliance in succeeding paragraphs. Data trends may sometimes provide clues to problem areas and/or sources of system errors. Discontinuities, as in any primary flight instrument presentation, are not desirable, and again can provide information which may aid in the solution of unacceptable system characteristics. In all cases, the analysis of the data must be tempered with a detailed understanding of the system being evaluated and the conditions under which it was tested.

Collection of quantitative data provides the opportunity to analyze qualitatively the system performance. Unfavorable qualitative comments or unexplained difficulty in performing the airspeed calibration tasks can be an indication of pitot/static system indication problems. Unbalanced pitot and static systems, large lag errors, indicator oscillations and poor system performance at extreme flight conditions frequently are identified first with pilot qualitative comments.

2.7 MISSION SUITABILITY

2.7.1 General

The flight calibration of aircraft pitot and static systems is a very necessary and important test. A large measure of the accuracy of all performance and flying qualities evaluations depends on the validity of these calibration tests. Errors in the pitot and static pressure systems may have serious implications when considered in conjunction with extreme speeds (high/low), maneuvering flight, and the instrument flight rules (IFR) low altitude missions required of current helicopters. The seriousness of the problem is compounded if pressure errors are transmitted to automatic flight control systems such as altitude retention systems, stability augmentation systems (SAS), or stabilator components.

2.7.2 Case Examples

Early models of the AH-1 helicopter used a nose mounted pitot tube and dual flush mounted fuselage static sources. The incorporation of a telescopic sight unit at the nose of the aircraft forced designers to seek an alternate location for the pitot source. Starting with the AH-1Q, the pitot source for subsequent Cobra helicopters was moved from the nose of the aircraft to an offset position left of the main rotor transmission cowling. The new location offers several undesirable characteristics. This position is primarily responsible for several observed airspeed problems.

The close proximity of the pitot source to the main rotor disc makes it highly susceptible to power effects. Large power changes result in excessive airspeed variations, even at cruise airspeed. The differences in airspeed position error with power setting is responsible for this phenomenon. This characteristic is particularly disconcerting during flight in instrument meteorological conditions. Each time a power charge is made, the change in airspeed position error results in an off desired speed condition (even if the actual aircraft speed does not change), forcing the pilot to make necessary control movements to get back to the desired indicated airspeed. This pitot system design deficiency results in increased pilot workload and often is evaluated as unacceptable for instrument flight purposes.

Another characteristic caused by the pitot tube location is observed when evaluating airspeed position error is sideslip. Typical values of airspeed position error (at 40 KCAS) with sideslip are shown in Figure 2.14. The excessive position errors and asymmetric nature of the data is directly attributable to pitot tube location. At the airspeed in the example, the AH-1S has an inherent right sideslip of approximately 14 degrees and the airspeed indicator is essentially unusable with sideslip angles greater than 20 degrees to the right. Nap-of-the-earth flight at 40 knots indicated airspeed would be very difficult due to this design characteristic. The addition of a low airspeed system to later models of the AH-1S was required to improve this problem area.

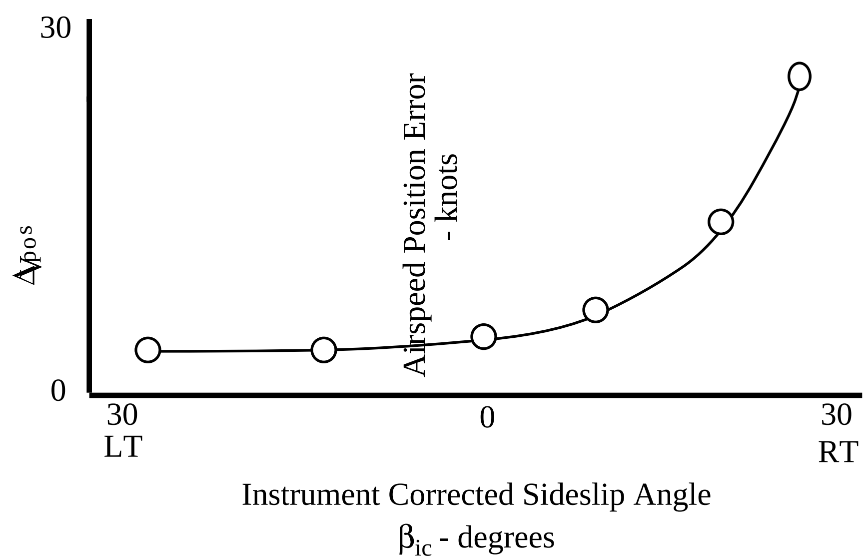


Figure 2.14
Typical AH-1S Airspeed System Correction Versus Sideslip

Another pitot/static system which has received much attention is installed on the UH-60A aircraft. The difficulties associated with this system are complicated greatly due to the interface between the pitot/static system and an automatic stabilator. Any irregularities sensed by the pitot/static system are programmed automatically into the stabilator and effect overall aircraft handling qualities.

The pitot/static system sources on the UH-60A helicopter are installed above the pilot and copilot cockpits. This high mounting position (relative to the main rotor system) allows rotor system wake impingement at a relatively high airspeed. Typical takeoff profiles allow this wake interference to take place at approximately 50 knots. A time history of both airspeed systems shows a short airspeed reversal; that is, an indicated decrease in airspeed when in fact the aircraft has not slowed at all. When this reversal is fed into the automatic stabilator system, it produces a stabilator response that causes the nose of the aircraft to pitch down requiring pilot compensation to maintain desired pitch attitude. This maneuver is particularly disconcerting when attempting takeoffs with sling loads or when flying the aircraft using night vision goggles. Several hardware modifications have been applied in an attempt to minimize this problem, but the perceptive pilot can still observe this phenomenon.

The pitot/static probes are also impinged by the main rotor wake in low speed high power climbs. While attempting to conduct an intermediate rated power climb at an indicated airspeed of 60 knots, the pilot frequently observes conditions as shown in Figure 2.15. The sinusoidal oscillation of both airspeed indicators, the stabilator, and longitudinal control movements required to minimize pitch attitude response is readily evident in this figure. Although this is not a frequently performed maneuver, it is still extremely disconcerting to the pilot, especially during conditions of reduced outside visual cues.

A test program of a twin pitot tube installation in another helicopter was completed without recording any data for the copilot airspeed system. The pilot and copilot airspeed systems were fed by separate pitot tubes which had very different position errors. Low speed and very high speed indications did not agree between systems and the result was disconcerting to the crew.

An important consideration is that the pilot may use observed airspeed as calibrated airspeed and hence include errors in navigation calculations.

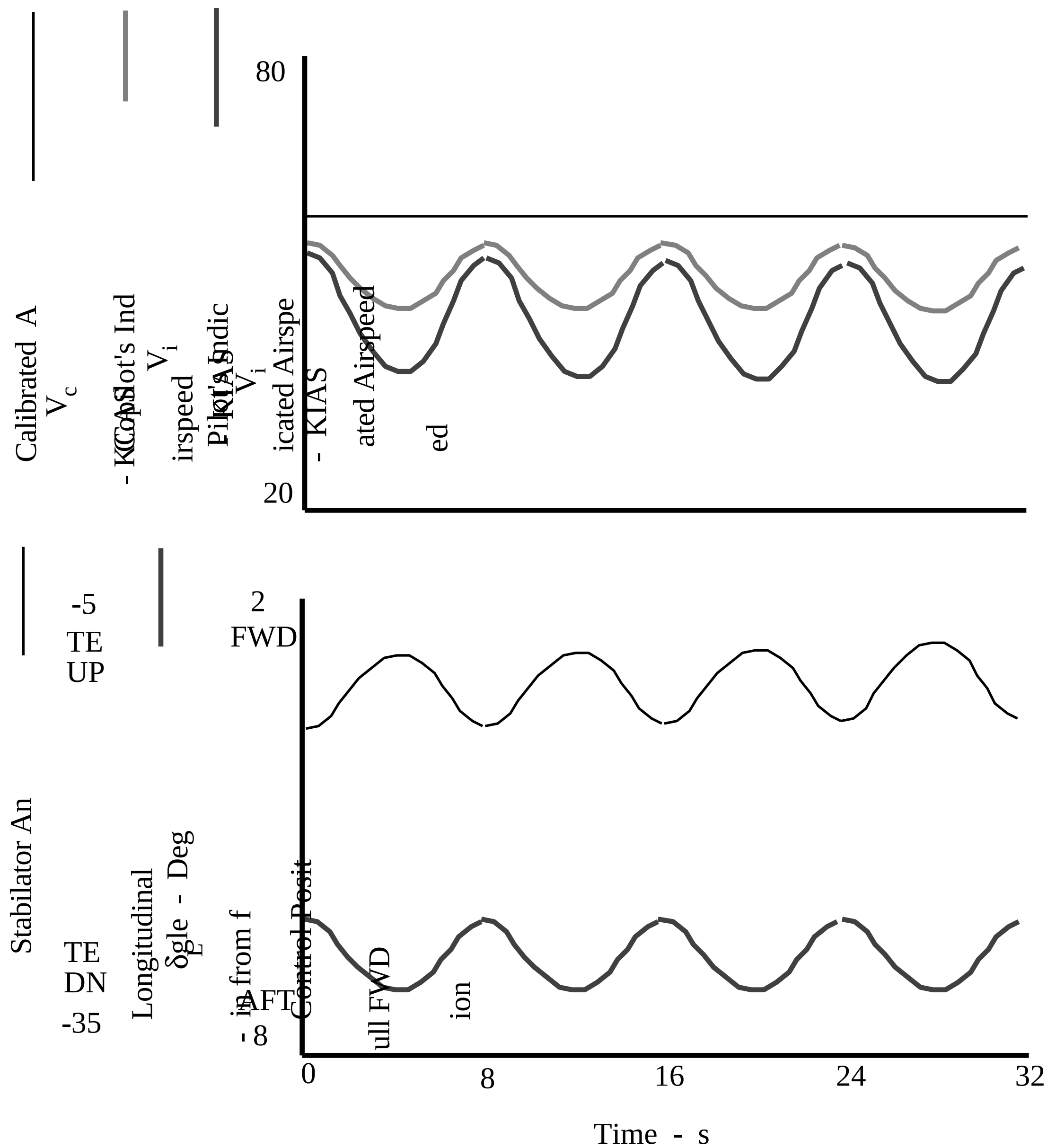


Figure 2.15
Typical UH-60A Pitot/Static System Time History

2.7.3 Scope of Test

The requirements of military specifications and the intended mission of the aircraft initially define the scope of the pitot and static system evaluation. The scope of the performance and flying qualities investigation may dictate an increase in the scope beyond that required above. This increase in scope may require flights at various rotor speeds, gross weights, or external configurations, and some testing may be required for calibration of the instrumentation boom system alone. Generally, a flight check of the requirements of the specifications for maneuvering flight may be accomplished concurrently with the standard test.

The flight test technique itself is not excessively demanding, but the pilot must be alert for characteristics which may not be obvious in the normal data reduction process.

It is the responsibility of the test pilot to observe and report undesirable characteristics in the pitot/static systems before they produce catastrophic results or degrade mission performance. When evaluating a pitot/static system in a helicopter, it is very important to consider the mission of the helicopter. Results of a pitot/static system calibration attained at sea level may indicate a satisfactory system. However, if the helicopter is to be operated at altitudes considerably higher than sea level, testing at operational altitude is required. The determination of mission suitability of a pitot/static system in a helicopter requires a thorough examination of the test results and the application of the test results to the intended mission of the helicopter.

2.8 SPECIFICATION COMPLIANCE

2.8.1 General

There are two military specifications which cover the requirements for pressure sensing systems in military aircraft. MIL-I-5072-1 covers all types of pitot/static tube operated instrument systems (Figure 2.16), while MIL-I-6115A deals with instrument systems operated by a pitot tube and a flush static port (Figure 2.17). Both specifications describe in detail the requirements for construction and testing of these systems and state that one of the following methods of test will be used to determine “installation error” (position error):

1. Speed course method.
2. Trailing tube.
3. Altimeter depression method (good for static error only).
4. Pacer airplane method.

Although Federal Aviation Regulations (FAR) are not directly applicable, pitot/static system requirements are generally evaluated to ensure safe operation on the federal airways. These requirements can be found in FAR Parts 27.1323 and 27.1335.

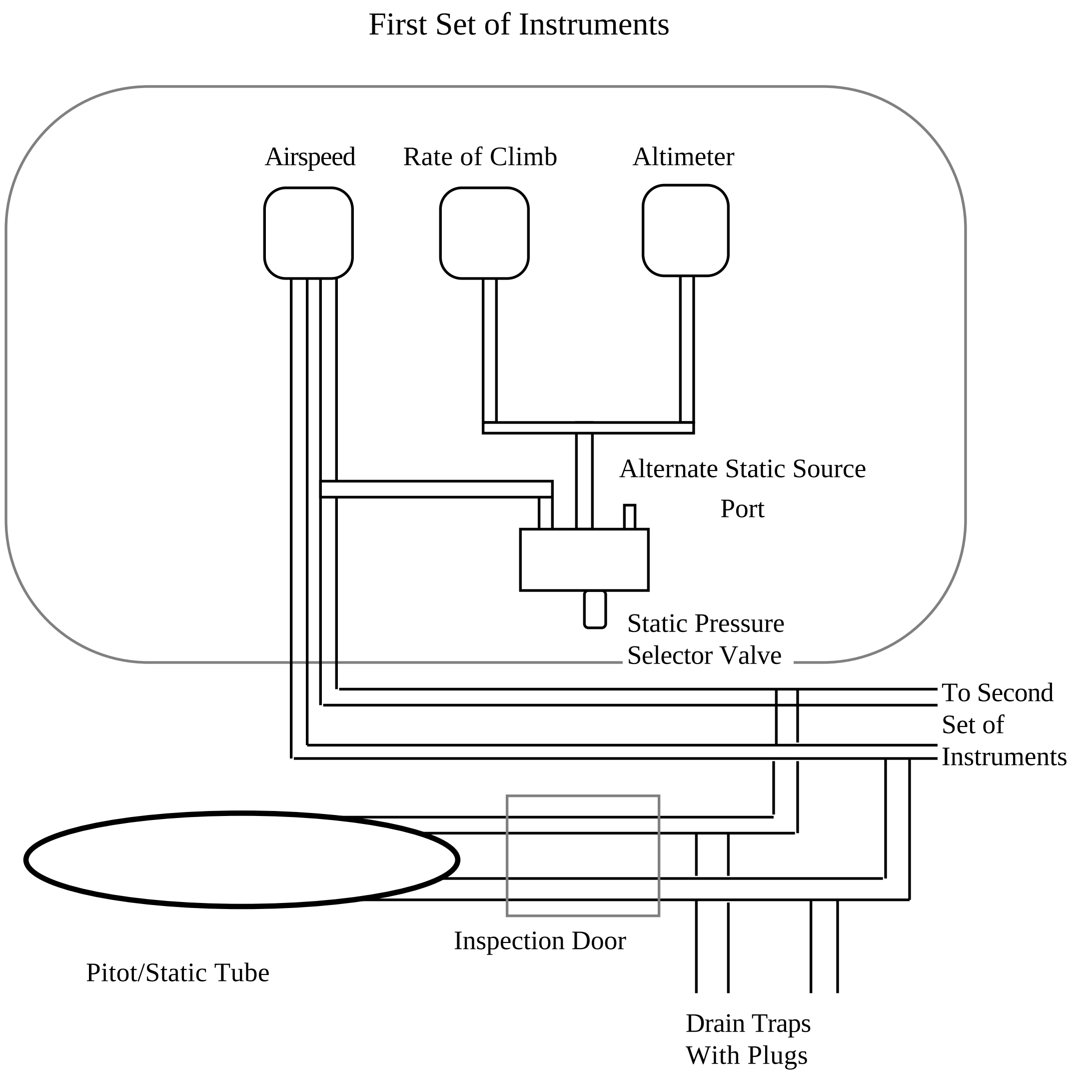


Figure 2.16
Pitot/Static System as Referred to in MIL-I-5072-1

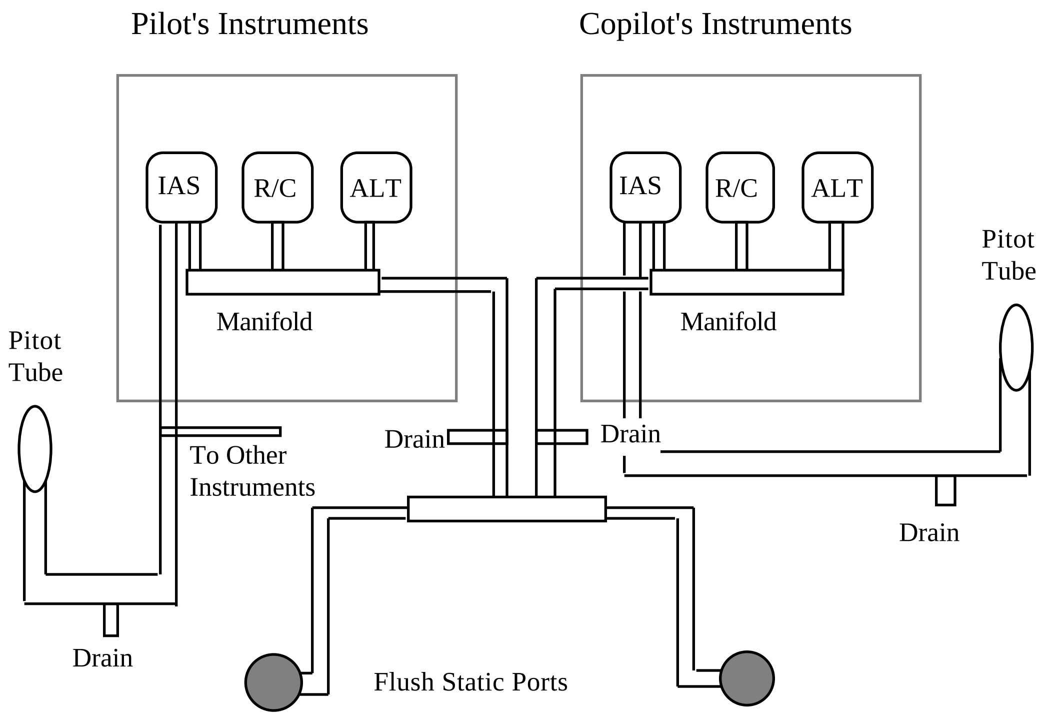


Figure 2.17
Pitot/Static System as Referred to in MIL-I-6115A

2.8.2 Tolerances

The requirements of both military specifications are written for fixed wing aircraft and must be interpreted for the helicopter or compound helicopter. Table I of MIL-I-5072-1 and MIL-I-6115A states the tolerance allowed on airspeed indicator and altimeter readings for five aircraft flight configurations. These configurations are sometimes difficult to equate to helicopter operations. The tolerances must be interpreted to cover level, climbing, and descending flight in a helicopter.

2.8.3 Maneuvers

2.8.3.1 GENERAL

There are four additional tests required by both military specifications which deal with the effect of maneuvers on pressure system operations. These tests are quite important and the descriptions of the test requirements are reprinted below as they appear in MIL-I-5072-1 and MIL-I-6115A.

2.8.3.2 PULLUP

"A rate of climb indicator shall be connected to the static pressure system of each pitot/static tube (the pilot's and copilot's instruments may be used). The variation of static pressure during pullups from straight and level flight shall be determined at a safe altitude above the ground and at least three widely separated indicated airspeeds. During an abrupt "pullup" from level flight, the rate of climb indicator shall indicate "Up" without excessive hesitation and shall not indicate "Down" before it indicates "Up"."

2.8.3.3 PUSHOVER

"A rate of climb indicator shall be connected to the static pressure system of each pitot/static tube (the pilot's and copilot's instruments may be used). The variation of static pressure during pushover from straight and level flight shall be determined at a safe altitude above the ground and at least three widely separated indicated airspeeds. During an abrupt "pushover" from level flight, the rate of climb indicator shall indicate "Down" without excessive hesitation and shall not indicate "UP" before it indicates "Down"."

2.8.3.4 YAWING

"Sufficient maneuvering shall be done in flight to determine that the installation of the pitot/static tube shall provide accurate static pressure to the flight instruments during yawing maneuvers of the airplane."

2.8.3.5 ROUGH AIR

"Sufficient maneuvering shall be done in flight to determine that the installation of the pitot/static tube shall produce no objectionable instrument pointer oscillation in rough air. Pointer oscillation of the airspeed indicator shall not exceed 3 knots (4 mph)."

2.9 GLOSSARY

2.9.1 Notations

a	Speed of sound	kn, ft/s
CG	Center of gravity	in
d	Horizontal distance	ft
D	Course length	ft
FAR	Federal Aviation Regulations	
GW	Gross weight	lb
$h_{a/c}$	Aircraft height above ground	ft
h_{ref}	Reference altimeter height AGL	ft
H_D	Density altitude	ft
H_P	Pressure altitude	ft
H_{P_c}	Calibrated pressure altitude	ft
$H_{P_{c\ twr}}$	Tower calibrated pressure altitude	ft
$H_{P_{ic}}$	Instrument corrected pressure altitude	ft
$H_{P_{ic\ ref}}$	Reference instrument corrected pressure altitude	ft
$H_{P_{ic\ ship}}$	Ship instrument corrected pressure altitude	ft
H_{P_o}	Observed pressure altitude	ft
$H_{P_{o\ ref}}$	Reference observed pressure altitude	ft
$H_{P_{o\ ship}}$	Ship observed pressure altitude	ft
$H_{P_{o\ twr}}$	Tower observed pressure altitude	ft
Δh	Aircraft height above tower	ft
ΔH_P	Change in pressure altitude	ft
$\Delta H_{P_{ic}}$	Altimeter instrument correction	ft
$\Delta H_{P_{ic\ ref}}$	Reference altimeter instrument correction	ft
$\Delta H_{P_{ic\ ship}}$	Ship altimeter instrument correction	ft
$\Delta H_{P_{ic\ twr}}$	Tower altimeter instrument correction	ft
ΔH_{pos}	Altimeter position error	ft

$\Delta H_{pos\ ref}$	Reference altimeter position error	ft
$\Delta H_{pos\ ship}$	Ship altimeter position error	ft
IFR	Instrument flight rules	
IGE	In-ground effect	
NAWCAD	Naval Air Warfare Center Aircraft Division	
N_R	Main rotor speed	%, rpm
OGE	Out-of-ground effect	
P	Pressure	psi, psf
P_a	Ambient pressure	psi, psf
P_s	Static pressure	psi, psf
$P_{s\ ref}$	Reference static pressure	in Hg
$P_{s_{ic\ ship}}$	Ship instrument corrected static pressure	in Hg
P_{ssl}	Standard sea level pressure	14.7 psi, 2116.2 psf
P_T	Free stream total pressure	psi, psf
$\Delta P_{s\ ship}$	Ship static pressure error	in Hg
q_c	Dynamic pressure	psi, psf
q_{ci}	Indicated dynamic pressure	psi, psf
Q	Engine torque	%, psi
SAS	Stability augmentation system	
T_a	Ambient temperature	°C
T_o	Observed temperature	°C
Δt	Elapsed time	s
ΔT_{ic}	Temperature instrument correction	°C
V	Velocity	kn
V_c	Calibrated airspeed	KCAS
V_e	Equivalent airspeed	KEAS
V_G	Ground speed	kn
V_i	Indicated airspeed	KIAS
V_{ic}	Instrument corrected airspeed	KIAS
V_{icship}	Ship instrument corrected airspeed	KIAS

V_o	Observed airspeed	KOAS
$V_{o\ ref}$	Reference observed airspeed	KOAS
$V_{o\ ship}$	Ship observed airspeed	KOAS
V_T	True airspeed	KTAS
ΔV_{ic}	Airspeed instrument correction	kn
$\Delta V_{ic\ ref}$	Reference airspeed instrument correction	kn
$\Delta V_{ic\ ship}$	Ship airspeed instrument correction	kn
ΔV_{pos}	Airspeed position error	kn
$\Delta V_{pos\ ref}$	Reference airspeed position error	kn

2.9.2 Greek Symbols

α (Alpha)	Angle of attack	
α_A	Airframe angle of attack	deg
α_i	Angle of incidence	deg
α_{ic}	Instrument corrected angle of attack	deg
α_o	Observed angle of attack	deg
α_p	Probe angle of attack	deg
$\Delta\alpha_{ic}$	Angle of attack instrument correction	deg
β (Beta)	Sideslip angle	deg
β_A	Airframe sideslip angle	deg
β_i	Sideslip angle of incidence	deg
β_{ic}	Instrument corrected sideslip angle	deg
β_o	Observed sideslip angle	deg
β_p	Probe sideslip angle	deg
$\Delta\beta_{ic}$	Sideslip angle instrument correction	deg
γ (Gamma)	Ratio of specific heats (Air: $\gamma = 1.4$)	
λ (Lambda)	Lag error constant	
λ_s	Static pressure lag error constant	
λ_T	Total pressure lag error constant	
ρ (Rho)	Density	slug/ ft^3

ρ_a	Ambient air density	slug/ ft^3
ρ_{ssl}	Standard sea level air density	0.0023769 slug/ ft^3
σ (Sigma)	Density ratio	
θ (Theta)	Angle	

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CHAPTER THREE

ENGINE ASSESSMENT

CHAPTER THREE

FIGURES

CHAPTER THREE

ENGINE ASSESSMENT

3.1 INTRODUCTION

3.1.1 General

In order to evaluate adequately the overall performance of the helicopter as an aircraft system, the individual components of the helicopter must be evaluated. The two major contributors to the overall aircraft performance are the engine performance (power available) and the airframe performance (power required). We must, therefore, evaluate both components of aircraft performance; engine and airframe.

In order to evaluate engine performance, we must first evaluate the pilot's interface with the engine and his control of that engine. This chapter, then, will deal with the pilot's interface with the engine, or engine assessment.

3.1.2 Types of Engines

In general, there are two main categories of helicopter propulsion systems, tip driven rotors, where the engine or part of it is blade mounted; and shaft driven rotors, with the engine in the fuselage. By far, the most common installation today is the shaft driven rotor. Of the shaft driven rotors, the engine may be a reciprocating engine, or a turbine engine. Of these two types of engines, most modern military helicopters are powered by the turbine engine. Within the turbine engine design, there are two types; the free power turbine and the fixed shaft turbine engine. The relative merits of the two designs are a matter of discussion; however, the free power turbine is by far the most widely used in the modern military helicopter. A few of the advantages are less power required to start the engine, the engine can be started with the rotor stopped, engine rotor matching is easier to achieve, and simple load sharing is possible in a multi-engine installation. This chapter will deal with the free power turbine engine.

Different engine manufacturers use varying shorthand notations to refer to the internal parts of a turbine engine. Engine gas generator speed may be referred to as N_1 or N_g. Engine power turbine speed may be referred to as N_2 or N_f. The engine temperature which the manufacturer chooses to measure and display in the cockpit may be power turbine inlet temperature (TIT), power turbine outlet temperature (TOT), exhaust gas temperature (EGT), or turbine gas temperature (TGT). Within this manual, N_1 will be used for engine gas generator speed, N_2 will be used for engine power turbine speed, and T_5 will be used to represent the measured and displayed engine temperature and is further defined as power turbine inlet temperature.

3.2 PURPOSE OF TEST

The purpose of these tests is to evaluate the engine/rotor compatibility and suitability for the mission of the host helicopter. Specific tests that are conducted include an evaluation of the engine controls and displays; engine operating procedures, both normal and emergency; engine start and shutdown characteristics; engine acceleration characteristics; engine trim response; engine/rotor stability, both static droop and transient droop; engine torque matching; engine limiting characteristics; and engine power contribution during minimum power descents.

3.3 THEORY

3.3.1 General

The theory that applies to the collection of engine assessment tests is no more than an understanding of the principles of operation and the requirements for the various systems under investigation.

3.3.2 Engine

The engine should be a very reliable, available, and maintainable engine capable of operation in all environments and with almost all fuels. It should have a high power output under all ambient conditions and should have a low specific fuel consumption. An ideal engine would be capable of operation in adverse environments without foreign object damage (FOD) protection and ice protection or noticeable degradation from sand, salt, or rain ingestion.

The engine should be inexpensive both to procure and to operate. It should have a low life cycle cost. It must be maintainable and supportable in the environment in which it will be operated. It should have a low parts count and be maintained with a few common tools.

On multi-engine installations, the engines should have good load sharing throughout the operating ranges and a high emergency power rating to cope with an engine failure situation.

The engine must be started easily and rapidly and operated throughout its range without constant operator monitoring. The engine monitoring systems should be automatic, error free, and not provide false alarms to the operator.

Although an ideal engine would meet all of these requirements, real engines can be expected to be deficient in one or several of these areas.

3.3.3 Governor

The governor must maintain constant rotor rpm under all operating conditions and not allow for an overspeed condition. The response from the engine/rotor governing system must be rapid enough to maintain rotor rpm during power changes. The governor should maintain stable control of the engine with the engine powering the rotor and during autorotation. In multi-engine installations, the individual engine governors must have the same operating characteristics throughout their operating ranges and engine matching must be achieved easily.

3.3.3.1 STATIC DROOP

The most common type of free power turbine governor is the proportional type where an error signal in the rotor rpm is used to change the fuel flow. This type of governor is also called an N_2 governor. The free power turbine is geared directly to the rotor; therefore, changes in the main rotor speed (N_R) are reflected as changes in the free power turbine speed. These changes in turbine speed or error signals from the desired turbine speed are sensed and a corresponding correction in fuel flow is made. For this type of governor to work there must be a change in rotor rpm with load or demand. The difference between initial N_R and final steady state N_R with power demand is known as STATIC DROOP (Figure 3.1).

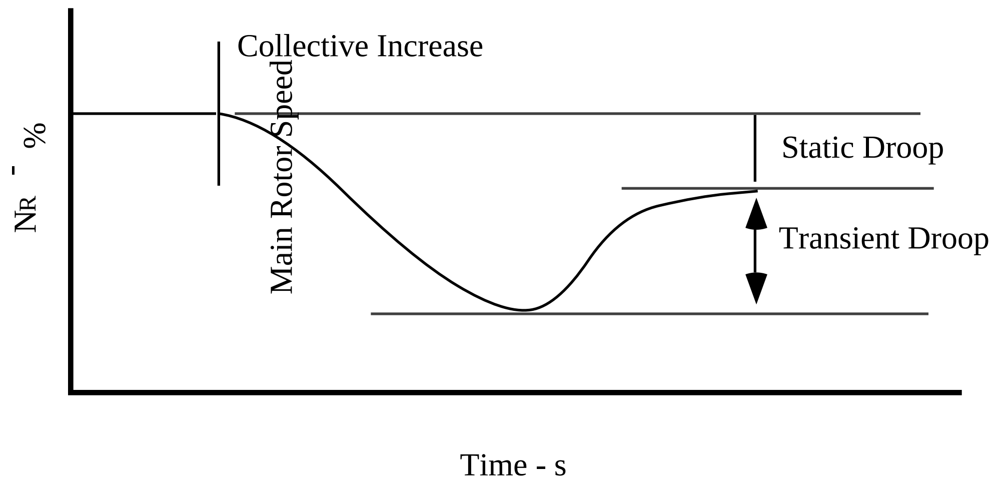

Figure 3.1
Main Rotor Droop

The governor gain is the fuel flow (W_f) per unit rotor rpm change (Figure 3.2). The governor gain defines the slope of the curve in Figure 3.2. This gain is unlikely to have a constant value or slope throughout the operating range of the engine.

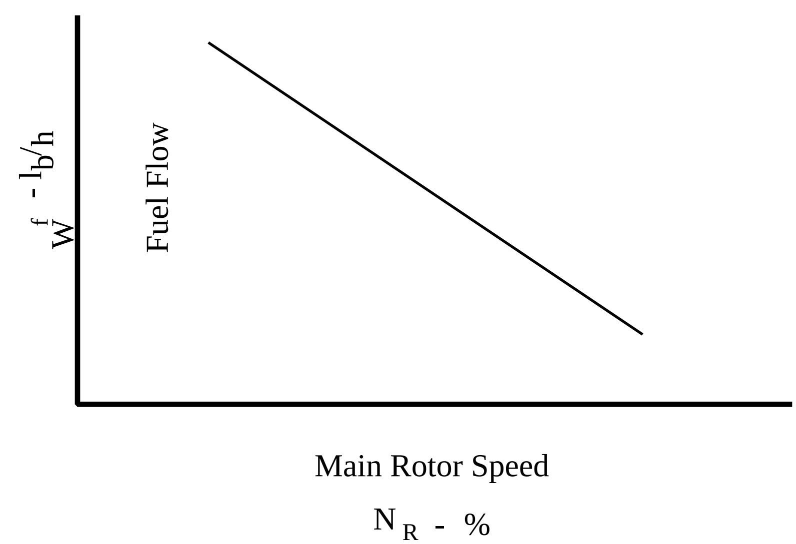

Figure 3.2
Governor Gain

The stability of the governing system is of great importance to the pilot. Poor stability will result in rotor over and underspeeds. If the governor gain is too high, the system response will be very rapid. The governor will over correct and induce speed oscillations which could reach dangerous proportions. The static droop will be small, however. If the governor gain is too low, engine response will be sluggish. The system will be stable but the static droop will be large.

The governor gain at which instability occurs may be influenced by the dynamic characteristics of the engine, transmission, and rotor system. Similarly, the stability of the governor system may be affected if any part of the power train has a natural frequency that can be excited.

A mechanical device that can be incorporated in the system to alleviate the static droop is a droop canceler. A droop canceler resets the datum of the governor so that the static droop may be reduced or eliminated (Figure 3.3). The canceler can be operated automatically through a mechanical interconnection with the collective lever. The effect of the canceler can be obtained manually through pilot operated variable rotor speed datum selection such as speed select. The effect in either case is to shift the governor gain curve in a parallel manner.

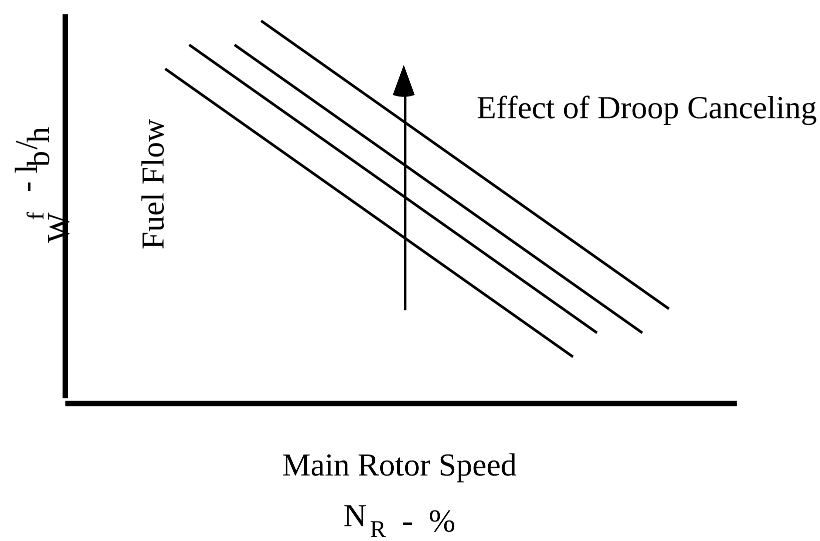

Figure 3.3
Droop Canceler

3.3.3.2 TRANSIENT DROOP

For any change in the power demanded, the governor will not take action until the rotor and the power turbine rpm changes. This is true for both an increase in the power demanded and for a reduction in the power demand. In the case of an increase in the power demand, the rotor rpm will decrease. The N_R decrease is sensed and an increase in the fuel flow is made. The rotor rpm will continue to decrease temporarily until the engine rpm is increased in response to the increase in fuel flow. The engine rpm will stabilize and the rotor rpm will also stabilize. This difference between the minimum rotor rpm achieved and the final steady state rpm is known as the TRANSIENT DROOP and is a function of the rate of power demand (Figure 3.1).

A similar difference in maximum rotor rpm and steady state rotor rpm can occur when a power reduction is made. The rotor rpm is increased when a reduction in the power demand is made. While the fuel flow is reduced and the engine speed reduced, the rotor rpm will continue to increase. The engine rpm will stabilize and the rotor rpm will stabilize. The difference or delta in rotor rpm between the maximum rotor rpm and the final steady state rpm is known as TRANSIENT OVERSHOOT.

The size of the transient droop or overshoot will depend on:

1. The size of the load change.
2. The rate of the load change.
3. The governor gain.
4. Basic engine response.
5. Inertia of the rotor.

Much as the droop canceler is added to help alleviate the static droop, a mechanical load anticipator is added to help alleviate the transient droop. The load anticipator is designed to give a momentary increase to the fuel flow as the collective is increased. This momentary increase due to the load anticipator effectively steepens the slope of the governor gain curve (Figure 3.4). The engine acceleration is begun in advance of the rotor droop and the effect is to reduce transient droop. Using a mechanical load anticipator , the amount of fuel flow increase is difficult to set correctly for all situations.

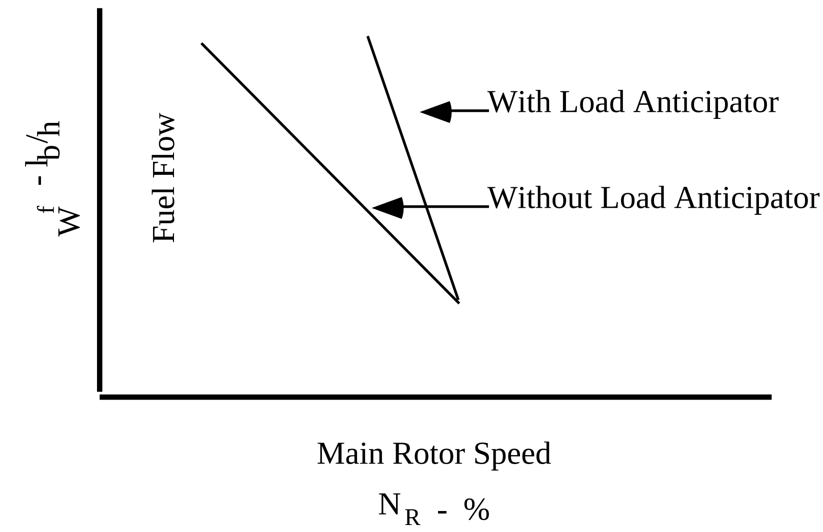

Figure 3.4
Load Anticipator

Electronic digital fuel control systems now allow more flexibility in providing load anticipation for a wide variety of situations. Unfortunately, these devices are more difficult to evaluate and document due to the increased variables effecting engine/rotor governing.

3.3.3.3 MULTI-ENGINE INSTALLATIONS

Multi-engine installations have an additional consideration. All engine governors must be matched; that is, they must have the same gain. The slopes of the governor gain curves, fuel flow versus rotor rpm, must be the same. If the governor gains are not the same, the engine load sharing will not be equal throughout the operating range.

If the mismatch occurs at low power (Figure 3.5), the helicopter will have poor flight idle characteristics with higher power contribution in a flight idle glide. The engines and rotor acceleration will be unpredictable.

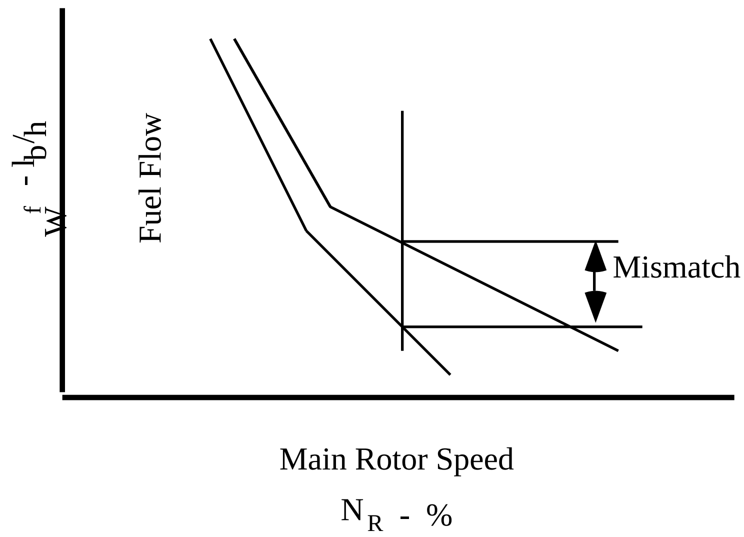

Figure 3.5
Governor Mismatch at Low Power

If the mismatch occurs at high power (Figure 3.6), total engine power will be limited. One engine will be operating at one of the engine limits while the other(s) will be operating below the limits, reducing total power available.

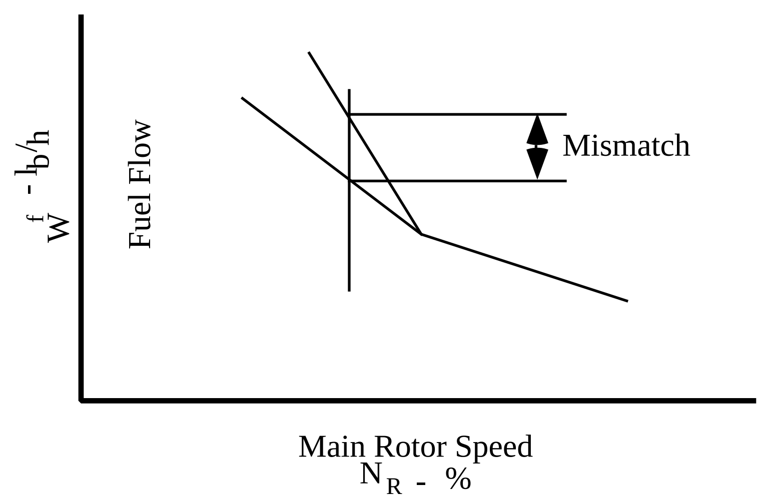

Figure 3.6
Governor Mismatch at High Power

Another type of mismatch occurs when the governor gain curves have the same slope but are parallel (Figure 3.7, 3.8). When the slope is greater, the mismatch is greater (Figure 3.7). This is the more severe case. When the slope is lower, the mismatch is less; but the static droop will be larger (Figure 3.8). Since the engines in a multi-engine installation normally do not have to accelerate quickly over a wide range of N_2 the shallow slope may be the more acceptable situation.

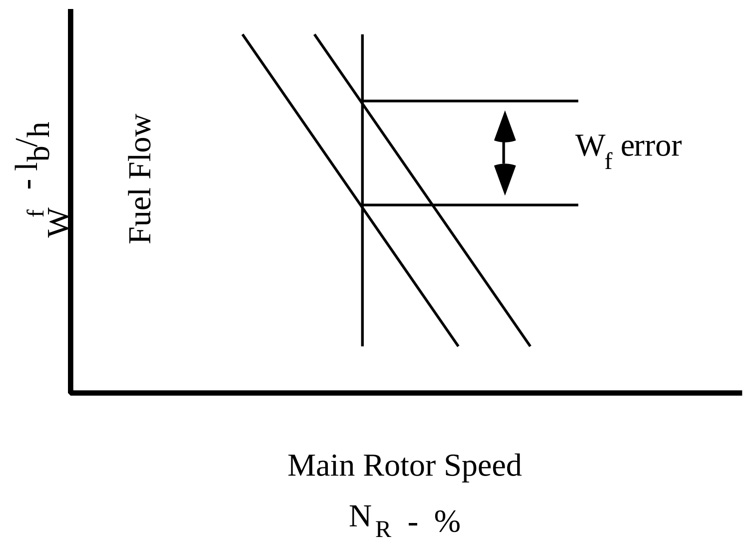

Figure 3.7
Governor Mismatch, High Gain

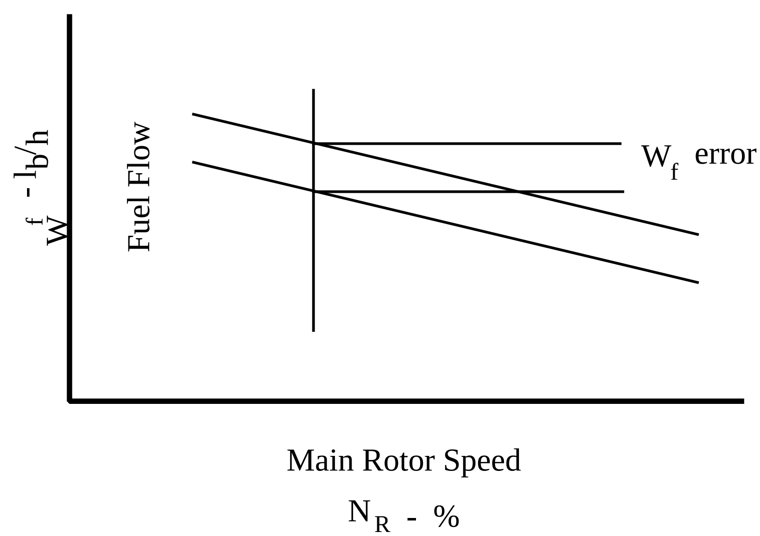

Figure 3.8
Governor Mismatch, Low Gain

3.3.4 Rotor Systems

Rotor system characteristics have an effect on the engine/rotor matching. For example, the basic minimum collective pitch settings must permit sufficient autorotative rotor rpm to enable an engine off landing. This characteristic will enhance training opportunities since practice autorotative descents will resemble closely actual autorotations. The corresponding flight idle engine speed must be self sustaining without producing a large power contribution while in flight idle glide. Additionally, the rotor system inertia can present a problem for the governor. Generally, the higher the inertia of the rotor system, the higher the governor gain required to compensate for the rotor rpm fluctuations. The rotor system configuration is also a consideration in the governor gain. A conventional rotor configuration, main and tail rotor, can require rapid engine response to accommodate frequent tail rotor power demands necessary to control uncommanded yaw rates. Limits can be placed on the rotor rpm for structural considerations, handling qualities, or because of accessory drive systems such as alternators. These limits may dictate the maximum allowable static droop, or the limits of transient droop and overshoot.

3.3.5 Engine Protection

Many types of devices have been fitted to protect the engine or the airframe from damage.

3.3.5.1 FREE POWER TURBINE OVERSPEED

The free power turbine overspeed protection is designed to protect the engine in the event of a governor runaway or a failure of the output shaft from the engine. This protective design must be very quick acting and usually functions through a fuel cutoff. Therefore, the design must be very reliable and not subject to false alarms. Usually this device is not used on single engine helicopters. Overspeed protection is installed on most multi-engine helicopters. To prevent a nominal engine from attempting to match a runaway engine, a discriminator must be part of the system.

3.3.5.2 GAS GENERATOR OVERSPEED

A mechanical maximum speed stop can be provided to prevent gas generator overspeed. This is especially important in a multi-engine installation to permit short periods of operation at maximum contingency power in the event of single engine failure. In some installations, a two position stop is provided to permit training or deliberate operation at a lower power level than would be used in an emergency. In twin turbine gas generators, it is common for only one turbine, the high pressure, to be controlled by the governor. However, the low pressure turbine may have rpm limits at high altitudes. In this case, a separate monitor and governor may be incorporated.

3.3.5.3 TEMPERATURE LIMITING

An engine temperature sensing device may be incorporated to sense power turbine inlet temperature (T_5) and to limit fuel flow and thus temperature. These devices may incorporate two datums or limits, one for starting when permissible temperatures may be higher and one for normal operations.

3.3.5.4 FLIGHT IDLE STOP

Many engine control systems incorporate a flight idle stop. The flight idle stop corresponds to the minimum fuel flow which can be delivered by the governor with the throttle in the fly position. There are two basic reasons to incorporate a flight idle stop. The first is to prevent the engine from decelerating to a speed where the engine is not self sustaining or susceptible to surge. The second reason is to prevent the engine from decelerating to a speed from which engine response is unacceptably slow.

There can be some problems associated with a fixed stop, particularly when considering operations over the entire gross weight range. One problem is the possibility of exceeding the rotor rpm limits or the N_2 limits when entering autorotation or during flight idle glide (minimum power descent). A second problem is a high power contribution during flight idle glide which reduces the rate of descent compared to a true autorotation. These problems may become apparent during operational maneuvers such as a quickstop.

3.4 TEST METHODS AND TECHNIQUES

3.4.1 General

The test methods and techniques employed for engine assessments are a continuation of engine bench and test cell testing. Additionally, the evaluation of the engine controls and displays is a continuation of a general cockpit evaluation with emphasis on the engine. The test methods that are discussed must be tailored to the specific engine installation. Thorough knowledge of the test helicopter with special emphasis on the engine/rotor, engine/rotor controls, and engine/rotor limits is mandatory for the specific application of the test methods and the success of the test program.

Prior to the start of the engine tests, it is essential that all rotor and engine adjustments and rigging be set to the manufacturer's recommendations. During the course of these rigging checks, the rigging procedures and the extremes of the rigging tolerances are checked.

Normally the engine tests are flown at a representative mission gross weight and center of gravity. Rotor rpm droop may be checked at maximum gross weight and the power contribution in flight idle glide may be checked at minimum gross weight.

During all flight tests, safety of the flight and ground crew is of utmost importance. Engine assessment testing is no different. The basic principle of incremental build up is followed. All limits for the engine, rotor, and transmission are observed and any approach to those limits are monitored. The crew must be thoroughly familiar with all applicable normal and emergency procedures. Thorough briefings covering all aspects of the tests must be conducted and attended by all flight, ground, and support crew. Contingencies for abort and emergencies must be considered.

3.4.2 Engine/Rotor Controls and Displays

An evaluation of the engine/rotor controls and displays is conducted on the ground and during all operating phases, both normal and emergency. The assessment should start on the ground and be continued throughout start, run-up, operating, and shutdown phases. Take into consideration all controls, instruments, displays, warning devices, warning lights, and warning sounds. Consider the location, functioning, labeling, access, grouping, and view of the controls and displays. Evaluate the standardization of the controls and displays within the test helicopter and with other fleet helicopters. An example is the UH-60 and the AH-64 which use similar audio tones for different warnings.

The evaluation should consider the operating procedures, both normal and emergency, in relation to the controls and displays. The adequacy and necessity of the controls and displays should be evaluated. This determination must be made with thorough knowledge of the systems. Similarly, the warnings must be evaluated for necessity and adequacy. Is a warning necessary or will a caution suffice? Can or must the pilot react to the warning or is it status information? Is a display necessary or will a segmented light be sufficient? The criticality of the information displayed must be evaluated and the pilot's ability to process and act on the information must be considered.

3.4.2.1 DATA REQUIRED

The data required for the evaluation of the controls and displays are a narrative description supported by pictures, photographs, or measurements.

3.4.3 Engine/Rotor Operating Procedures

The pilot and copilot operation of the engine and rotor is evaluated, both on the ground during simulated operations and during actual operations. The evaluation is conducted along with an evaluation of the controls and displays. Here, however, the manual and mental operations are evaluated, singularly and in combination. Attention is paid to the control movements that are required for all normal and emergency procedures. Special attention is paid to combinations of control movements that must be accomplished, especially if more than one crew member is required to accomplish a task. Also, special attention is given to monitoring displays or multiple displays while performing control movements. The evaluation must be conducted under realistic conditions. A task that appears to be performed easily while static, in the chocks, may be difficult to perform while in flight under adverse conditions.

3.4.3.1 DATA REQUIRED

The data required are a narrative description of the operations, supported by figures, photographs, or measurements as necessary.

3.4.4 Engine/Rotor Start and Shutdown Characteristics

The evaluation of the engine and rotor starting characteristics begins with an evaluation of the manufacturer's recommended method, under favorable conditions. The conditions that would be most favorable will generally be an assisted start, with a cold engine, facing into a light wind on a moderate temperature day. If these conditions cannot be met, deviations from a nominal start should be anticipated. In a logical build up fashion the engine starts should progress toward more extreme conditions, using all possible combinations of starting methods or techniques. For a single engine helicopter normally started with a battery or auxiliary power unit (APU), the cold engine into the wind start with APU would be made first, building up to a hot engine start on the battery, while facing down wind. If a battery, external APU, internal APU, or cross engine start are possible, all combinations of starting must be evaluated for practicality, ease, and consistent results. The cause of any starting abnormalities should be investigated. These investigations should involve the maintenance experts as well as the design engineers.

Time histories of engine parameters and control positions are made. Don't influence the data by powering the automatic recording equipment from the same source used to power the start. If a battery start is attempted, the data acquisition package should not be powered from the battery. The consequent drain of battery power may influence the start. In this case, an alternate power source for the data package or an alternate means of recording engine data must be used. One simple method is a hand held video of the appropriate cockpit instrumentation and control positions. While more difficult to reduce, this data may be sufficient for the intended use.

Additional areas of interest include starter function or dropout. During the start sequence, all equipment should be in the operating condition specified by the manufacturer. How the associated equipment is effected by the start should be evaluated. If the manufacturer provides an option on the operational condition of other equipment, then all possible combinations should be evaluated. If other equipment may be operated inadvertently in a manner not specified, then the effect of engine starting should be evaluated for this combination.

For rotor starting or engagement, the first method evaluated should be the manufacturer's recommended procedure. This may involve nothing more than starting the engine on a small helicopter without a rotor brake or clutch. The rotor may be engaged with one or more engines set to various power levels. Again, the principle of incremental build up is the governing principle. Always start with the most benign conditions and progress toward the more critical, stopping if an unfavorable trend exists or if unforecast results occur.

Engine/rotor shutdown characteristics are evaluated in much the same fashion. Initially investigate the manufacturer's recommendation, followed by approved optional means, and finally by all reasonable combinations. Engine and rotor deceleration from normal operating conditions are evaluated as part of the shutdown evaluation. The effect of engine/rotor shutdown on other helicopter systems is evaluated. Again, the ambient conditions such as temperature, cooling period of the engine(s), and relative wind are varied. The relevant data are time histories of engine and rotor parameters.

3.4.4.1 DATA REQUIRED

Time history of engine/rotor parameters: Q, N_1, N_2, T_5, W_f, and N_R. Annotate engine/rotor control activation. Annotate system operation. Note ambient conditions: OAT, H_p, wind speed and direction (relative to the nose of the helicopter), wind gust spread, and previous cooling period of the engines (if applicable).

3.4.4.2 SAFETY CONSIDERATIONS/RISK MANAGEMENT

The crew must be thoroughly familiar with abort start procedures and hot start/shutdown procedures. The area for the test must be suitable for engine and rotor starting and free of all FOD hazards. All ground support personnel must be briefed and all personnel must be thoroughly familiar with standard hand signals. Adequate fire protection and fighting means must be on hand. Interested observers and auxiliary equipment, such as power for the data recording device, must be well clear. Appropriate communications among all cockpit, aircrew, and ground crew must be used.

3.4.5 Engine Acceleration Characteristics

The rate at which the engine(s) can be accelerated from flight idle or ground idle to full throttle should be evaluated. Engine parameters as well as rotor speed should be recorded and observed as a function of time and rate of throttle application. The rate of throttle application should be varied as a function of engine trim (beep) setting. Engines should be advanced individually and collectively. Use the rotor brake(s) within the design tolerances. That may mean single engine acceleration to maximum with the other engines at idle while the rotor brake is on or, it may mean no accelerations beyond idle with the rotor brake on. In any event, the incremental build up is used and operations are conducted from the manufacturer's recommended base. As with engine and rotor starting, the effect of engine and rotor acceleration on associated helicopter equipment should be noted. Areas of possible concern are alternator/generator operation, hydraulic system operation, and structural loads (especially in the landing gear).

3.4.5.1 DATA REQUIRED

Time history of Q, N_1, N_2, T_5, W_f, N_R, and throttle position. Annotate system operation. Ambient conditions.

3.4.5.2 SAFETY CONSIDERATIONS/RISK MANAGEMENT

During rapid engine and rotor acceleration avoid exceeding system limitations. Wheel brakes, locks, and chocks should be employed. As the datum engine/rotor trim setting is advanced to the maximum, avoid exceeding system limitations. Avoid inadvertent movement on the ground during accelerations. The same site and crew precautions that are observed during engine start should be observed during the accelerations.

3.4.6 Engine Trim Response

In determining engine trim response two characteristics of system function time (trim delay) and trim rate are evaluated. Trim delay is the period of time between the actuation of the cockpit engine speed control and an engine response, as measured by fuel flow or engine rpm (N_2). Trim rate is the engine response, usually measured in rpm (N_2) per unit input of trim control (one second input). To evaluate these characteristics, three or more time histories of engine response to increasing duration trim inputs are made. The speed trim inputs are made from minimum engine speed or minimum "beep" settings. Reducing engine rpm from maximum engine rpm is evaluated also. Any difference in system function should be noted as it may cause problems for the pilot in selecting the desired rpm. In a multi-engine helicopter, the test is performed with multi-engine trim inputs to vary the N_2/N_R. Single engine trim inputs are used for torque matching (Paragraph 3.4.8).

3.4.6.1 DATA REQUIRED

Time history of N_2/N_R and engine trim speed or beep position. Ambient conditions.

3.4.6.2 TEST CRITERIA

Start trim input from minimum and maximum settings.

3.4.6.3 SAFETY CONSIDERATIONS/RISK MANAGEMENT

When performing these tests, do not exceed engine/rotor limitations, particularly during long increasing trim applications. Again, all the normal precautions of ground operations must be observed. If the trim rate is unusually fast, care must be exercised so that the helicopter does not move on the ground. Use the brakes, chocks, and wheel locks. If the helicopter has moved during this test, difficulty in unlocking the wheel locking device may be encountered.

3.4.7 Engine/Rotor Stability (Static and Transient Droop)

When testing engine/rotor stability, the characteristics of static and transient droop are evaluated. Static droop is the steady state variation in rotor rpm with a change in engine power demand. For increasing engine power demand, measured as increasing engine torque, the variation in rotor rpm, usually an overall decrease in rotor rpm, is static droop. For decreasing engine power demand or reduced torque, the variation in rotor rpm, usually an overall increase in rotor rpm, is termed static overshoot. The static droop and static overshoot can be influenced by forward airspeed. For this reason, a complete evaluation of static droop and overshoot is usually performed on the ground, at a hover, and at several representative forward airspeeds.

The technique is to stabilize at the desired airspeed and normal operating rotor rpm. It is imperative that once the collective is moved from the initial conditions, it is moved in the same direction until the test is complete. At a hover, the evaluation begins on the deck facing into the wind at minimum collective and normal operating rotor rpm. When stabilized at the initial conditions, record engine/rotor parameters and airspeed. Increase the collective in increments of approximately 5 % Q. When stabilized, record the data. Manual data recording usually suffices for this test; however, automatic data recording can be used. The process is repeated over the range of available power, without exceeding any limitations. The initial airspeed (i.e. wind speed) is maintained while the power is increased and a vertical velocity is accepted. No adjustments in the engine/rotor trim speed is made. The rotor rpm is allowed to vary according to the governor schedule. At maximum allowable power, normal operating rotor rpm is selected with the engine speed control. If maximum power for the desired airspeed will not produce normal operating rotor rpm, select maximum “beep” and note the rotor rpm. Data are recorded. The test is repeated in forward flight for collective reductions. Incremental decreases in engine power are made and the resultant rotor rpm is noted until minimum collective is reached. Again, once the collective is moved from the initial conditions, it must be moved in the same direction until the test is complete. The data must be recorded when all engine and rotor parameters are stable. Activation of engine/rotor caution or warnings are noted. These tests are often used to gather engine performance data (Chapter 4) as well as static droop data. If this is the case, a complete set of engine and rotor parameters are recorded at each condition (Q, N_1, N_2, T_5, N_R, W_f) along with aircraft and atmospheric data. Engine performance will be covered in more detail in Chapter 4.

Transient droop is the variation in rotor rpm with variations in rate of change in engine power demand. Normally this test is performed in forward flight, although the test may be performed in an OGE hover as well. (The control coordination required to perform this test in a hover may be such that the tests are restricted to forward flight airspeeds. However, modern aircraft with flight control augmentation may make performance of this test in a hover practical.) The effect of rapid collective increases on the flight control system may be of interest. The technique is to stabilize at an airspeed and altitude of interest and set the normal operating rotor rpm. Climb the aircraft above the test altitude. Descend at minimum collective or the collective setting that permits the engine and rotor rpm to be "just joined". This is the initial collective setting for the tests. The N_R may stabilize higher than that set in level flight, although care should be taken to avoid exceeding N_R limits. A climb at the test altitude is made to establish the maximum collective setting that will be used. This setting is one that corresponds to maximum power plus some margin for safety. The collective setting usually corresponds to 80 to 90% of maximum torque or temperature, or whichever limit is most restrictive. Once these two collective settings are noted the test begins. With the aircraft above the test altitude, descend at the previously determined lower collective setting. At the test altitude a slow smooth collective increase is made to the previously determined upper collective setting. During this increase automatic data recording is used to record engine and rotor rpm. The test runs are repeated for the airspeed of interest with increasing rates of collective application as the aircraft descends to the target altitude. The tests are continued until a maximum rate of collective application is achieved or until another aircraft limit is reached. The test series may be repeated for additional airspeeds of interest. During the tests, note activation of engine/rotor cautions or warnings and other variations from nominal operation.

Transient overshoot tests are conducted following the same principles for a collective reduction from maximum power at the airspeed of interest.

The effects of engine/rotor stability on the performance of mission maneuvers are documented by using automatic data recording during mission maneuvers. Such maneuvers are quick stops, bob-ups, remasking, rapid accelerations, and dashes. Performance of these maneuvers may be linked to the engine/rotor stability.

3.4.7.1 DATA REQUIRED

Static Droop: Stabilized Q, N_R, airspeed. Annotate system or warning operation. Ambient conditions.

Transient Droop: Time history of Q, N_R, and collective position. Airspeed. Annotate system or warning operation. Ambient conditions.

3.4.7.2 TEST CRITERIA

1. Fixed engine trim setting during tests.
2. Static droop, collective movement in one direction only.
3. Static droop, stable engine power demand.

3.4.7.3 DATA REQUIREMENTS

1. Static droop, stabilize 30 s minimum prior to recording data.
2. $V_o \pm 1$ kn.

3.4.7.4 SAFETY CONSIDERATIONS/RISK MANAGEMENT

During these tests, observe engine/rotor and aircraft limitations. The test crew must have thorough knowledge of all applicable limitations. Use extreme caution in the transient droop test to avoid low rotor rpm situations that may present system or handling qualities problems. Consider the engine power indicating system and any lags in that system for the rate of collective application and the maximum collective. Since these tests are performed at conditions of high collective settings and possible low rotor rpm, they must be conducted over an area suitable for autorotation. This is especially true for the hover tests. Perform the forward flight tests at a test altitude that favors an autorotation considering that both climbs and descents from the test altitude will be made.

3.4.8 Engine Torque Matching

For multi-engine installations, engine torque matching and load sharing throughout the operating envelope are of interest. The ease with which the pilot can match the engine power will be determined as part of the evaluation of normal and emergency operating procedures as described in paragraph 3.4.3. In conjunction with trim response tests, evaluate the ease of matching engine torque by using single engine trim inputs. Equal load demand between or among the engines is of interest. Consider engine power control during actual and simulated single engine situations. Additional special techniques are not required for this evaluation. Observation during normal and simulated emergency operation will reveal any problems. If a problem with torque matching is encountered, make a time history of engine/rotor parameters to document the problem.

3.4.8.1 DATA REQUIRED

Time history of Q, N_1, N_2, T_5, N_R, W_f. Ambient conditions. Maneuver description. Time history of Q and engine trim position for pilot engine torque matching.

3.4.9 Engine Limiting Characteristics

Many modern helicopter engines incorporate automatic limiting features. These features may limit engine torque, engine temperature, or engine speed (N_1 and/or N_2). If these features are included, they should be tested for proper operation. The technique used is to approach the limiting condition slowly and note the operation of the feature. If the operation of the limiting feature presents a problem for the pilot, document with a time history.

3.4.9.1 DATA REQUIRED

Time history of Q, N_1, N_2, T_5, W_f, N_R as appropriate. Ambient conditions. Description of limiting feature.

3.4.9.2 SAFETY CONSIDERATIONS/RISK MANAGEMENT

By the very nature of these tests, limitations are going to be reached. Approach the limits slowly and carefully. Thorough knowledge of the limiting feature is necessary to anticipate aircraft response. If an unanticipated response is found, the tests should be suspended until the reason for the response is understood.

3.4.10 Engine Power Contribution

The rate of descent with the collective at the minimum pitch setting and with the engine(s) on line is evaluated and compared to the rate of descent obtained in full autorotation. Make a descent with the collective at minimum and 100 % N_R or normal operating rotor speed for a given airspeed, gross weight, and altitude band. The rate of descent is determined by timing through the altitude band, normally 1000 ft. The descent is repeated in full autorotation for the same test conditions of airspeed, gross weight, and altitude band. Again, the rate of descent is determined by timing through the altitude band. The tests can be repeated for additional airspeeds throughout the normal operating range.

Start the descent above the altitude band of interest to insure that the aircraft is stabilized while passing through the band. Time splits are usually taken at 500 ft increments. This is done as a backup measure in case the entire 1000 ft run is not stable. If that is the case, the first 500 ft or the second 500 ft, may provide usable data.

If the helicopter power controls may not be retarded to ground idle in flight, full autorotation must be simulated. Split the N_2 and N_R by using the engine speed trim control to reduce the N_2.

3.4.10.1 DATA REQUIRED

Altitude band, time through band, airspeed, N_R, gross weight (fuel used or remaining) and ambient conditions.

3.4.10.2 TEST CRITERIA

1. Stabilized N_R.
2. Balanced, wings level, unaccelerated flight.

3.4.10.3 DATA REQUIREMENTS

1. $V_o \pm 1$ kn.
2. $N_R \pm 1$ %.

3.4.10.4 SAFETY CONSIDERATIONS/RISK MANAGEMENT

The floor of the altitude band should be high enough to allow time during the descent to bring the engine back on line to power the rotor and still maintain a margin of altitude safety. Complete the recovery to powered flight no lower than 1000 ft AGL. The area must be suitable to continue the descent to touchdown if the power can not be brought back on line for any reason. The first airspeed investigated should be the recommended airspeed for minimum rate of descent. An incremental build up is conducted to expand the airspeed range. Keep in mind that additional recovery altitude is necessary at greater rates of descent.

3.5 DATA REDUCTION

3.5.1 General

The numerical data reduction for the engine assessment tests is quite simple. The appropriate instrument calibration factors are applied to the observed data recorded from cockpit instrumentation to obtain indicated values for the data.

3.5.2 Engine/Rotor Controls and Displays

Not applicable. Narrative data presentation.

3.5.3 Engine/Rotor Operating Procedures

Not applicable. Narrative data presentation.

3.5.4 Engine/Rotor Start and Shutdown Characteristics

Time history of engine/rotor parameters: Q, N_1, N_2, T_5, W_f, N_R are presented for each of the starts and shutdowns. Annotate ambient conditions and method of start and shutdown. Annotate any anomalies in system functioning. Annotate engine and starting control operation (Figure 3.9).

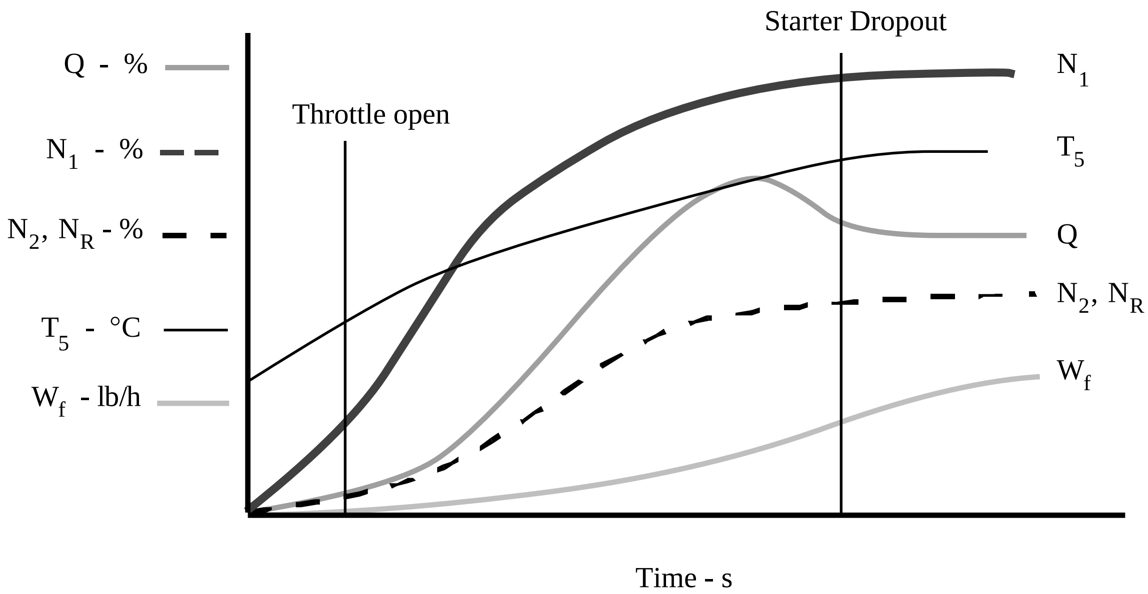

Figure 3.9
Engine Start

3.5.5 Engine Acceleration Characteristics

Time history of engine/rotor parameters: Q, N_1, N_2, T_5, W_f, N_R and throttle position are presented for each acceleration. Annotate datum engine trim speed or beep setting (Figure 3.10).

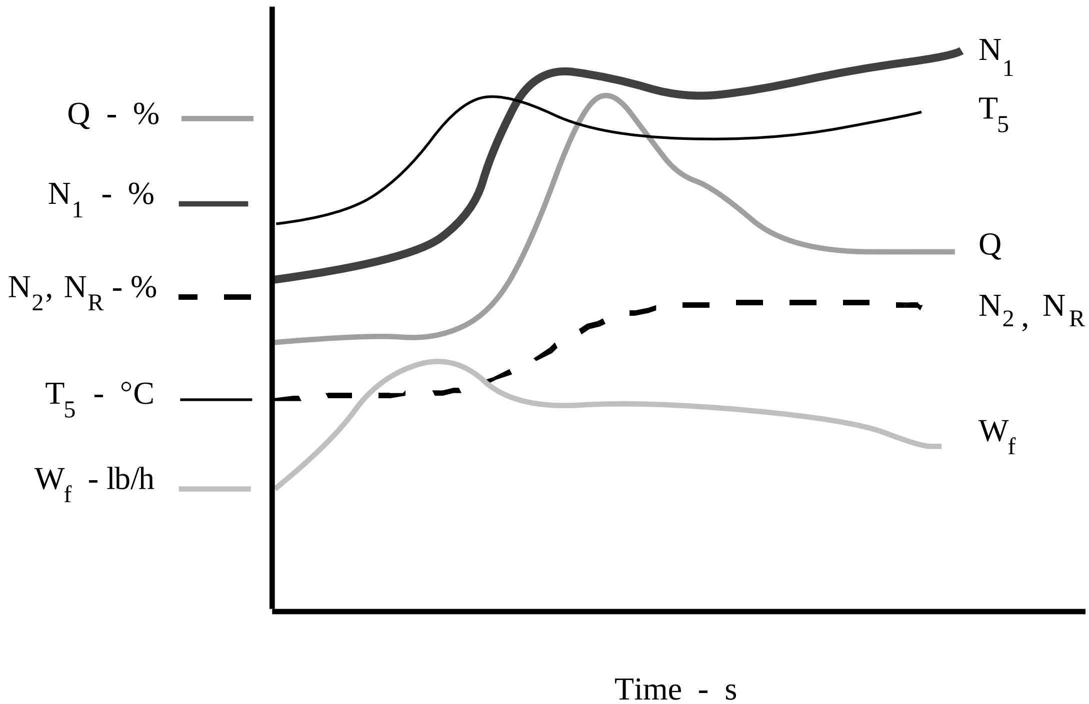

Figure 3.10
Engine Acceleration, 3 Seconds From Idle to Fly

3.5.6 Engine Trim Response

Time history of N_2/N_R for engine trim speed or beep increase and decrease are presented. Annotate trim delay time (Figure 3.11).

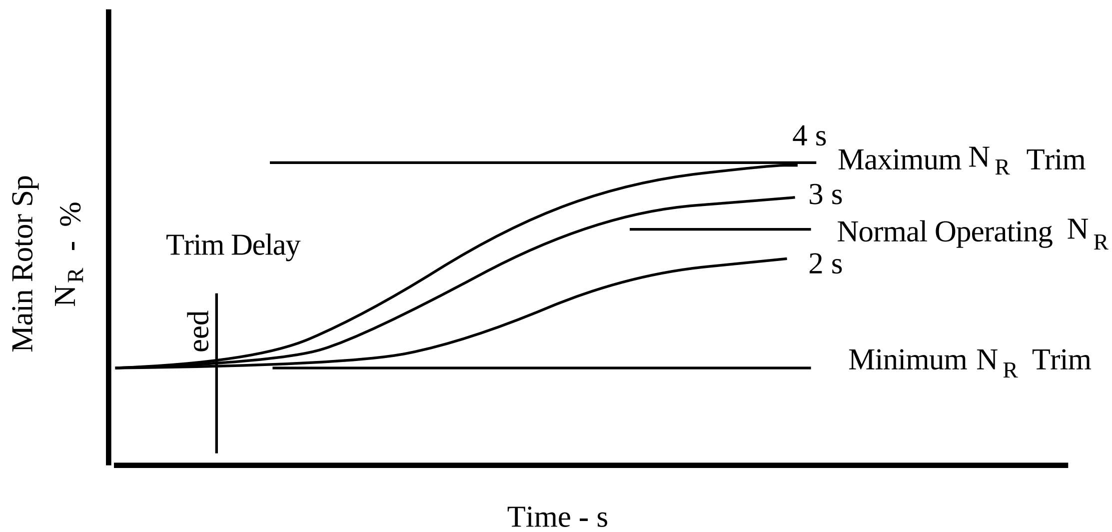

Figure 3.11
Engine Trim Response; 2, 3, 4 Second Application

3.5.7 Engine/Rotor Stability (Static and Transient Droop)

Static droop. Plot main rotor speed versus engine torque. Annotate test airspeed and conditions. Annotate normal, minimum, and maximum rotor speed. Annotate rotor speed warnings (Figure 3.12).

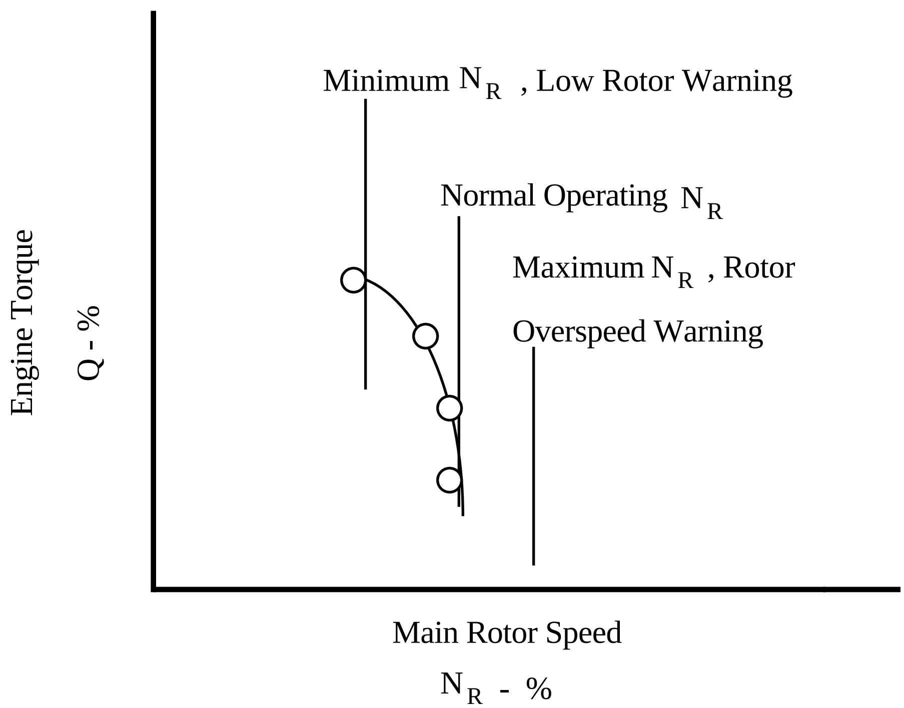

Figure 3.12
Static Droop

Transient droop. Time history of N_R. Annotate test conditions. Annotate maximum and minimum rotor speed. Annotate rotor speed warnings (Figure 3.13). Summary plot of transient droop and/or minimum N_R versus collective application time (Figure 3.14).

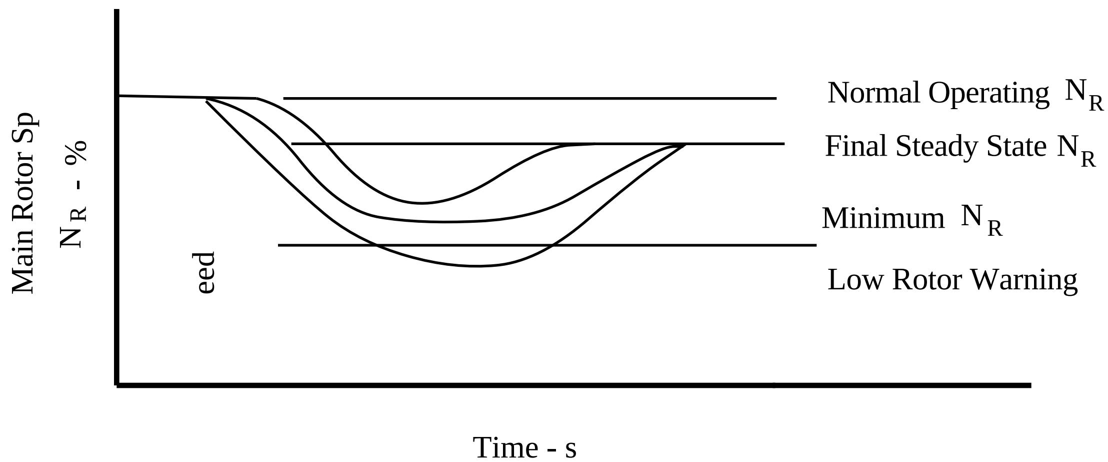

Figure 3.13
Transient Droop Time History

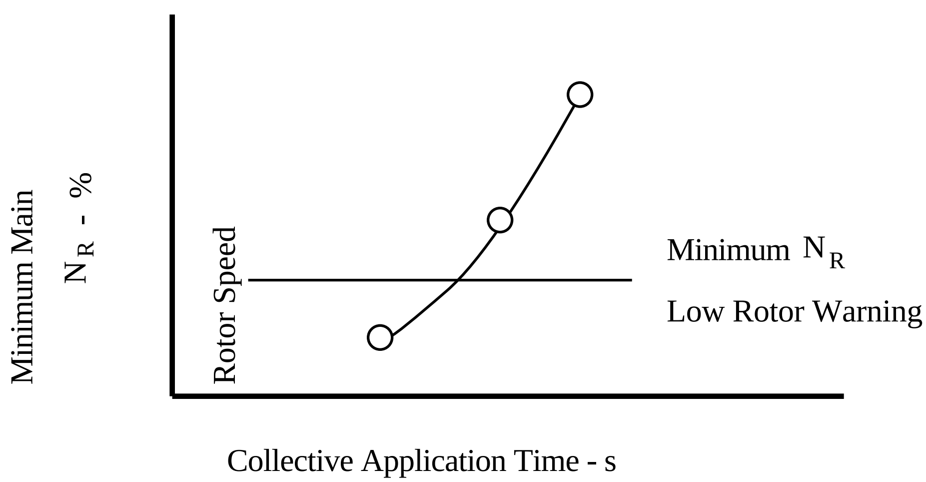

Figure 3.14
Transient Droop

3.5.8 Engine Torque Matching

Time history of engine/rotor parameters: Q, N_1, N_2, T_5, W_f, N_R for problem maneuver (Figure 3.15).

Time history of engine torques for pilot engine torque matching in a multi-engine installation (Figure 3.16).

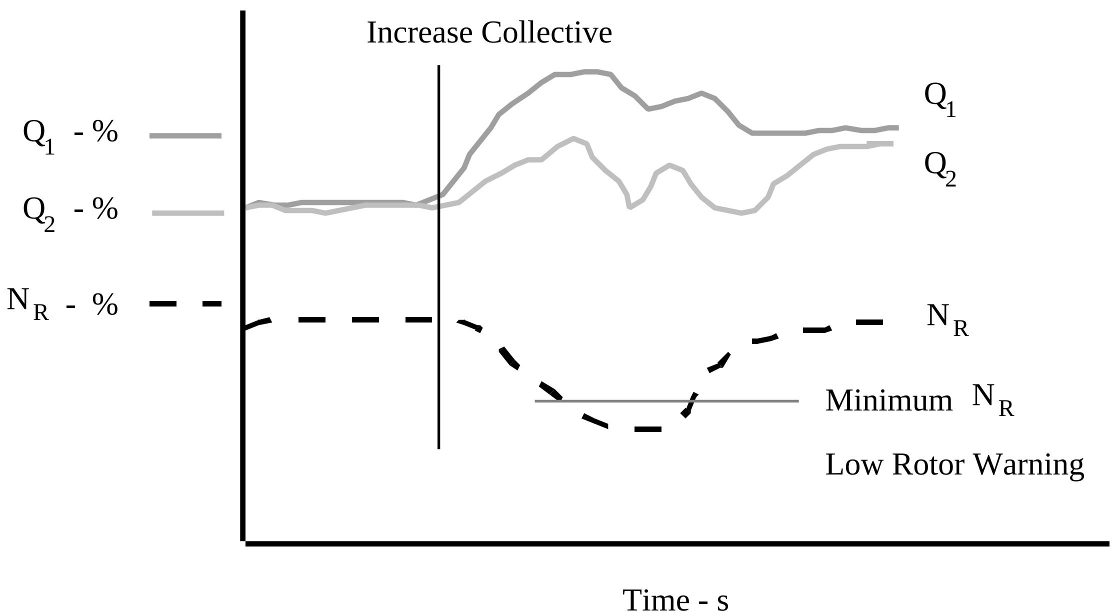

Figure 3.15
Engine Torque Matching During Bob-Up

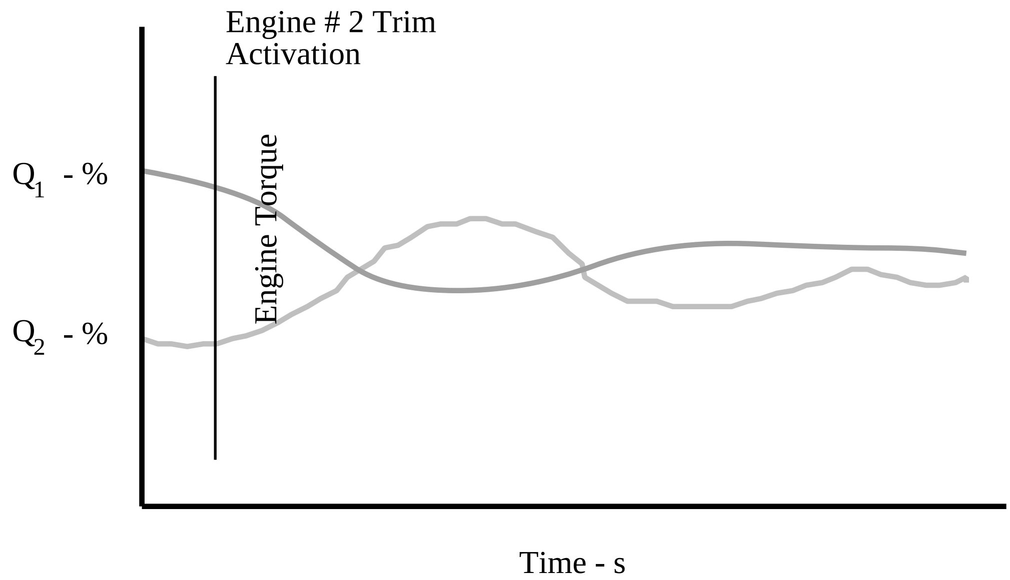

Figure 3.16
Engine Torque Matching

3.5.9 Engine Limiting Characteristics

Time history of engine/rotor parameters: Q, N_1, N_2, T_5, W_f, N_R. Annotate limiting function (Figure 3.17).

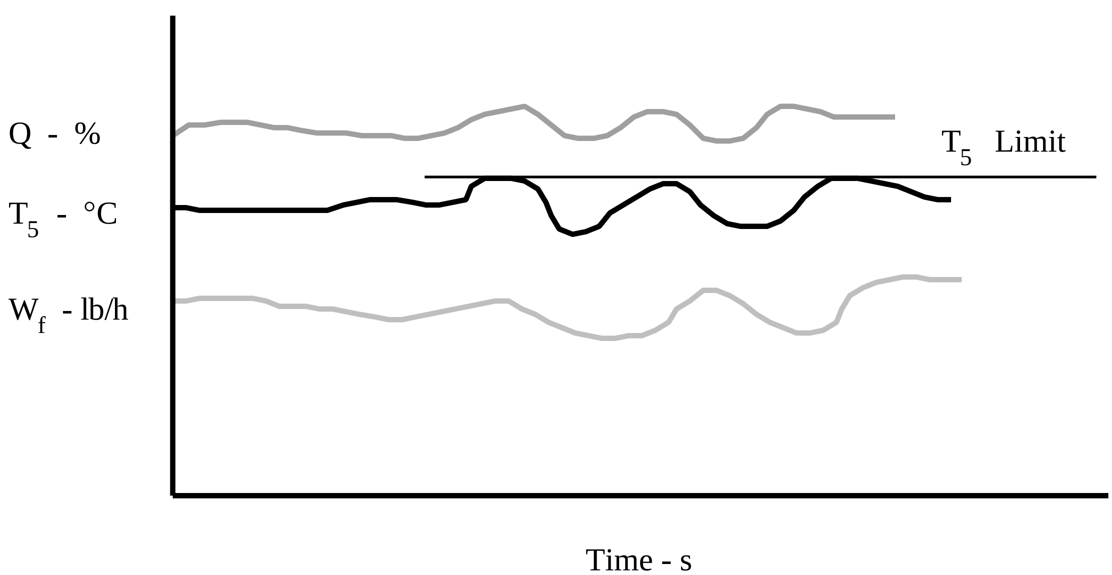

Figure 3.17
Engine T_5 Limiting

3.5.10 Engine Power Contribution

Plot of rate of descent for both minimum collective descent and full or simulated autorotation versus airspeed. Annotate airspeed for minimum rate of descent and maximum glide (recommended or actual) (Figure 3.18).

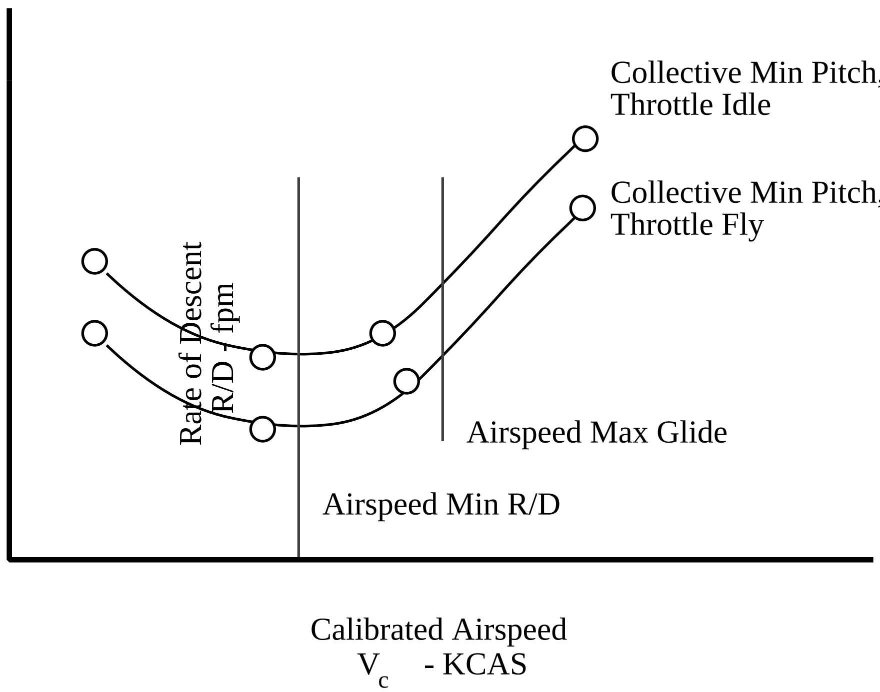

Figure 3.18
Engine Power Contribution During Descent

3.6 DATA ANALYSIS

Engine assessment tests are made to document characteristics (i.e. static and transient droop) or to uncover possible problem areas (i.e. engine torque matching or engine limiting characteristics). Data is presented to support these characteristics or problems. An analysis of the impact on mission performance is made. Representative mission maneuvers and functions are performed to support the mission considerations.

3.7 MISSION SUITABILITY

Mission suitability is the primary reason for engine assessment testing. How do the engine characteristics effect the mission performance? Operation of the engine/rotor in simulated mission tasks and under simulated mission conditions will reveal the overall suitability of the engine/rotor installation. The pilot and crew should be able to direct their attention to the employment of the helicopter as a weapon system without being diverted to the operation of the engine/rotor. The operation of the engine/rotor should not distract from essential tasks such as navigation, target acquisition, and weapons delivery.

3.8 SPECIFICATION COMPLIANCE

Engine assessment tests generally are not covered in the Military Specifications. Each helicopter and engine detail or development specification may contain provisions for engine assessment. Each specification must be researched for applicability and a determination made if the characteristics meet the specification.

3.9 GLOSSARY

3.9.1 Notations

AGL	Above ground level	
APU	Auxiliary power unit	
FOD	Foreign object damage	
IGE	In-ground effect	
N_1,N_g	Engine gas generator speed	%, rpm
N_2, N_f	Engine power turbine speed	%, rpm
N_R	Main rotor speed	%, rpm
Q	Engine torque	%, psi
T_5	Power turbine inlet temperature	°C
V_c	Calibrated airspeed	KCAS
V_o	Observed airspeed	KOAS
W_f	Fuel flow	lb/h

3.9.2 Terms

Flight idle glide	Minimum power descent.
Flight idle stop	The minimum fuel flow which can be delivered with the throttle in the fly position.
Static droop	The change in N_R, from initial to final steady state, caused by a change in power demand.
Transient droop	The change in N_R, from minimum to final steady state, caused by the rate of change in power demand.

3.10 REFERENCES

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CHAPTER FOUR

ENGINE PERFORMANCE

CHAPTER FOUR

FIGURES

CHAPTER FOUR

EQUATIONS

$$\delta = \frac{P}{P_{ssl}}$$ *eq 4.1*

$$\theta = \frac{T}{T_{ssl}}$$ *eq 4.2*

$$\delta_a = \frac{P_a}{P_{ssl}}$$ *eq 4.3*

$$\theta_a = \frac{T_a}{T_{ssl}}$$ *eq 4.4*

$$\delta_{s_2} = \frac{P_{s_2}}{P_{ssl}}$$ *eq 4.5*

$$\theta_{s_2} = \frac{T_{s_2}}{T_{ssl}}$$ *eq 4.6*

$$\delta_{T_2} = \frac{P_{T_2}}{P_{ssl}}$$ *eq 4.7*

$$\theta_{T_2} = \frac{T_{T_2}}{T_{ssl}}$$ *eq 4.8*

$$\delta_{T_2} = \frac{P_{T_2}}{P_{ssl}} = f\left(V_c,\ \delta_a\right)$$ *eq 4.9*

$$\theta_{T_2} = \frac{T_{T_2}}{T_{ssl}} = f\left(V_c, \theta_a\right)$$ *eq 4.10*

$$\frac{ESHP}{\delta\sqrt{\theta}} = f\left(\frac{N_1}{\sqrt{\theta}}, \frac{N_2}{\sqrt{\theta}}, \frac{V_T}{\sqrt{\theta}}, R_n\right)$$ *eq 4.11*

$$\frac{W_f}{\delta\sqrt{\theta}} = \left(\frac{N_1}{\sqrt{\theta}}, \frac{N_2}{\sqrt{\theta}}, \frac{V_T}{\sqrt{\theta}}, R_n\right)$$ *eq 4.12*

$$ESHP_{corr} = \frac{ESHP}{\delta_{T_2}\sqrt{\theta_{T_2}}}$$ *eq 4.13*

$$W_{f_{corr}} = \frac{W_f}{\delta_{T_2}\sqrt{\theta_{T_2}}}$$ *eq 4.14*

$$T_{5_{corr}} = \frac{T_5}{\theta_{T_2}}$$ *eq 4.15*

$$N_{1_{corr}} = \frac{N_1}{\sqrt{\theta_{T_2}}}$$ *eq 4.16*

$$ESHP_{corr} = f(N_{1_{corr}})$$ *eq 4.17*

$$W_{f_{corr}} = f(N_{1_{corr}})$$ *eq 4.18*

$$T_{5_{corr}} = f(N_{1_{corr}})$$ *eq 4.19*

$$W_{f_{corr}} = f(ESHP_{corr})$$ *eq 4.20*

$$SFC = \frac{W_{f_{corr}}}{ESHP_{corr}}$$ *eq 4.21*

$$\overline{P}_{s_2} = \frac{\sum P_{s_2}}{n}$$ *eq 4.22*

$$\overline{P}_{T_2} = \frac{\sum P_{T_2}}{n}$$ *eq 4.23*

$$M_2 = \sqrt{\left(\frac{2}{\gamma - 1}\right)\left[\left(\frac{\overline{P}_{T_2}}{\overline{P}_{s_2}}\right)^{\frac{\gamma - 1}{\gamma}} - 1\right]}$$ *eq 4.24*

$$T_{T_2} = T_{s_2}\left[1 + \left(\frac{\gamma - 1}{2}\right)M_2^2\right]$$ *eq 4.25*

$$\delta_{T_2} = \frac{\overline{P}_{T_2}}{P_{ssl}} = \left(\frac{\overline{P}_{T_2}}{P_a}\right)\left(\frac{P_a}{P_{ssl}}\right) = \left(\frac{\overline{P}_{T_2}}{P_a}\right)\delta_a$$ *eq 4.26*

$$\theta_{T_2} = \frac{T_{T_2}}{T_{ssl}} = \frac{\left(T_{T_2} - T_a\right)}{T_{ssl}} + \frac{T_a}{T_{ssl}} = \frac{\left(T_{T_2} - T_a\right)}{T_{ssl}} + \theta_a$$ *eq 4.26a*

$$V_c = V_o + \Delta V_{ic} + \Delta V_{pos}$$ *eq 4.27*

$$ESHP = (K_Q)(Q)(N_R)$$ *eq 4.28*

$$\left(N_{1_{corr}}\right)_{max} = \frac{N_{1_{max}}}{\sqrt{\theta_{T_2}}}$$ *eq 4.29*

$$\left(T_{5_{corr}}\right)_{max} = \frac{T_{5_{max}}}{\theta_{T_2}}$$ *eq 4.30*

$$\left(W_{f_{corr}}\right)_{max} = \frac{W_{f_{max}}}{\delta_{T_2}\sqrt{\theta_{T_2}}}$$ *eq 4.31*

$$ESHP_A = \left(\frac{ESHP}{\delta_{T_2}\sqrt{\theta_{T_2}}}\right)_{max}\left(\delta_{T_2}\sqrt{\theta_{T_2}}\right)$$ *eq 4.32*

$$\frac{P_{T_2}}{P_a} = f(V_c)$$ *eq 4.33*

$$\left(T_{T_2} - T_a\right) = f(V_c)$$ *eq 4.34*

$$\frac{P_{T_2}}{P_a} = A_0 + A_1 V_c + A_2 V_c^2 + A_3 V_c^3$$ *eq 4.35*

$$\left(T_{T_2} - T_a\right) = B_0 + B_1 V_c + B_2 V_c^2 + B_3 V_c^3$$ *eq 4.36*

$$\frac{P_a}{P_{ssl}} = \left[1 - \left(\frac{\lambda_{ssl} H_P}{T_{ssl}}\right)\right]^{\frac{g_{ssl}}{g_c \lambda_{ssl} R}}$$ *eq 4.37*

$$T_a = \frac{OAT + 273.15}{1 + 0.2K_T\left[\frac{\left(\frac{V_c^2}{\delta}\right)}{a_{ssl}^2}\right]}$$ *eq 4.38*

$$P_{T_2} = \left(\frac{P_{T_2}}{P_a}\right)\left(P_a\right)$$ *eq 4.39*

$$T_{T_2} = (T_{T_2} - T_a) + T_a$$ *eq 4.40*

$$ESHP_{corr} = C_0 + C_1(N_{1_{corr}}) + C_2(N_{1_{corr}})^2 + C_3(N_{1_{corr}})^3$$ *eq 4.41*

$$W_{f_{corr}} = D_0 + D_1(N_{1_{corr}}) + D_2(N_{1_{corr}})^2 + D_3(N_{1_{corr}})^3$$ *eq 4.42*

$$T_{5_{corr}} = E_0 + E_1(N_{1_{corr}}) + E_2(N_{1_{corr}})^2 + E_3(N_{1_{corr}})^3$$ *eq 4.43*

CHAPTER FOUR

ENGINE PERFORMANCE

4.1 INTRODUCTION

4.1.1 General

The thrust of this manual is the evaluation of helicopter performance for mission suitability and specification compliance. Chapter 2 is concerned with the measurement of airspeed and altitude and the performance of the measurement systems. Knowledge of accurate airspeed and altitude is important both for the employment of the helicopter and for flight tests. Chapter 3 covers the area of pilot interface with the engine, also known as engine assessment. The remaining chapters of this manual discuss the measurement of airframe power required to perform in various flight regimes: hover, vertical climb, level flight, and forward flight climb and descents. In order to determine aircraft performance, power available must be measured and combined with airframe power required to obtain aircraft performance parameters. Aircraft performance parameters of interest are, among others, maximum level flight airspeed, hover ceiling, and maximum rate of climb.

4.1.2 Inlet Performance

Inlet performance as used in this chapter refers to engine inlets. The inlet is the entire ducting, shaping, guiding, or other devices between the engine compressor face and the free air stream. The design of the inlet may be plain or it may be complex. The most important aspect of inlet design is the impact on engine performance.

The engine is often designed well in advance of the airframe it is eventually placed in. Often an engine will be employed in many airframes, such as the T-700 in the AH-64 and UH-60. The engine manufacturer's knowledge of future applications is often limited. Therefore, in the design stage one can say that there will be inlet effects which may detract from the test cell engine performance.

Inlets are very difficult to design for optimum performance (minimum detraction from engine power available) over a wide range of operating environments. The actual airflow around the helicopter will be very disturbed compared to the free stream conditions. The disturbances will vary with flight regime such as level flight, hover, and climb. Also, overall helicopter design parameters play an important part. Engines are placed beneath the rotor, the airflow is less than ideal due to flow separation and boundary layer ingestion.

4.1.3 Engine Performance

The total performance evaluation of the helicopter requires determination of engine performance characteristics. Engine theory and test cell engine power charts can be used to determine engine performance. However, these data are developed without regard for the installation or application of the engine, and may be in error by as much as ± 5 %. This accuracy is not sufficient for flight test application or for translation into handbook performance information.

The engine performance characteristics of corrected engine shaft horsepower ($ESHP_{corr}$), corrected engine gas generator speed ($N1_{corr}$), corrected power turbine inlet temperature ($T5_{corr}$), corrected fuel flow ($W_{f_{corr}}$), and specific fuel consumption (SFC) must be determined. These parameters define engine performance over the range of operating conditions and are corrected to standard sea level conditions for comparison purposes.

4.1.4 Power Available

Having determined inlet performance and engine performance, and armed with a knowledge of engine limitations, engine power available for the combination of desired conditions (altitude, temperature, and airspeed) can be determined. Power available data can then be combined with power required data obtained from tests discussed in later chapters to obtain overall aircraft performance.

4.2 PURPOSE OF TEST

The purpose of these tests is to determine power available. To accomplish this end, inlet performance and engine performance will be determined.

4.3 THEORY

4.3.1 General

The engine is developed separately from the airframe it is used in. Often the engine is used in more than one airframe. The distortion of the airflow at the engine compressor face, caused by the specific airframe and the inlet design, will affect engine performance. Engine performance must be determined for this distortion in airflow and corrected to standard sea level conditions to obtain power available. Once the corrections to standard sea level conditions are made, power available can be determined for any conditions of interest.

4.3.2 Inlet Performance

Inlet performance is defined as the influence on free stream air flow caused by the engine inlet. The distortion in flow, pressure and temperature, reaching the engine compressor face is the measure of inlet performance. The temperature and pressure are measured across the engine compressor (Figure 4.1) and compared to the free stream air flow.

The pressure ratio (δ) and temperature ratio (θ) can be determined for the engine inlet. The pressure ratio,

$$\delta = \frac{P}{P_{ssl}} \qquad \text{eq 4.1}$$

Where:

δ	- Pressure ratio
P	- Pressure
P_{ssl}	- Standard sea level pressure

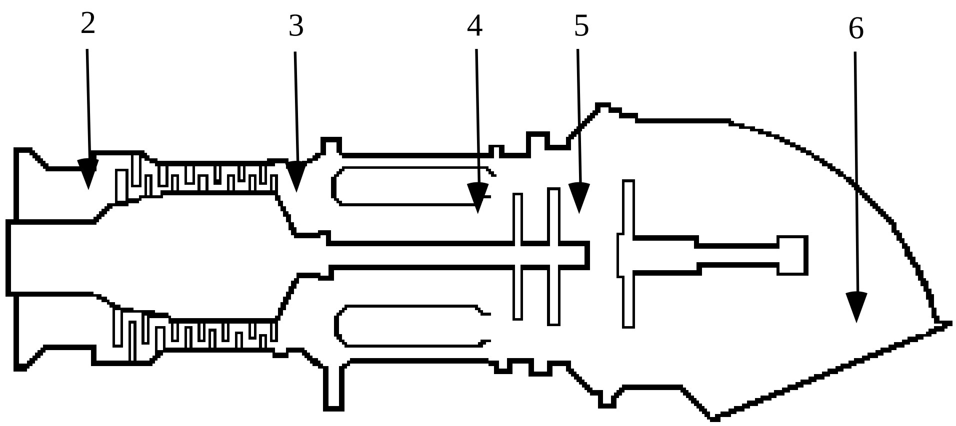
and temperature ratio,

$$\theta = \frac{T}{T_{ssl}} \qquad \text{eq 4.2}$$

Where:

θ	- Temperature ratio
T	- Temperature
T_{ssl}	- Standard sea level temperature, 288.15 °K

can be based on ambient conditions; P_a, T_a; static conditions at the engine compressor face; P_{s_2} , T_{s_2}; or total conditions at the engine compressor face; P_{T_2} , T_{T_2}.



2. Compressor Inlet
3. Compressor Discharge
4. Gas Generator Turbine Inlet
5. Power Turbine Inlet
6. Exhaust Gas Discharge

Figure 4.1
Engine Stations

If the ratios are based on ambient conditions, then the pressure ratio is the ambient pressure ratio,

$$\delta_a = \frac{P_a}{P_{ssl}} \qquad \textit{eq 4.3}$$

Where:

δ_a - Ambient pressure ratio

P_a - Ambient pressure

P_{ssl} - Standard sea level pressure

and the temperature ratio is the ambient temperature ratio,

$$\theta_a = \frac{T_a}{T_{ssl}}$$ *eq 4.4*

Where:

θ_a - Ambient temperature ratio

T_a - Ambient temperature

T_{ssl} - Standard sea level temperature, 288.15 °K .

These ratios take into consideration the effect of altitude and temperature variation from standard sea level conditions on engine performance but do not address the impact of the inlet on the pressure and temperature present at the engine compressor face.

The ratios can be based on the static conditions that exist at the engine compressor face. The pressure ratio becomes the static pressure ratio,

$$\delta_{s_2} = \frac{P_{s_2}}{P_{ssl}}$$ *eq 4.5*

Where:

δ_{s_2} - Static pressure ratio

P_{s_2} - Inlet static pressure

P_{ssl} - Standard sea level pressure

and the temperature ratio becomes the static temperature ratio,

$$\theta_{s_2} = \frac{T_{s_2}}{T_{ssl}} \qquad \textit{eq 4.6}$$

Where:

θ_{s_2}	- Static temperature ratio
T_{s_2}	- Inlet static temperature
T_{ssl}	- Standard sea level temperature, 288.15 °K .

These ratios take into consideration the effect of altitude and temperature variation from standard sea level conditions and the impact of the inlet on the static conditions at the engine compressor face. However, the static ratios do not consider the total conditions that exist at the engine compressor face.

The pressure and temperature ratio that best defines the impact of the inlet on the air flow at the engine compressor face is the inlet pressure ratio,

$$\delta_{T_2} = \frac{P_{T_2}}{P_{ssl}} \qquad \textit{eq 4.7}$$

Where:

δ_{T_2}	- Inlet pressure ratio
P_{T_2}	- Inlet total pressure
P_{ssl}	- Standard sea level pressure

and the inlet temperature ratio,

$$\theta_{T_2} = \frac{T_{T_2}}{T_{ssl}} \qquad \textit{eq 4.8}$$

Where:

θ_{T_2} - Inlet temperature ratio

T_{T_2} - Inlet total temperature

T_{ssl} - Standard sea level temperature, 288.15 °K .

These ratios compare the total pressure and temperature at the engine compressor face to the standard sea level pressure and temperature. The inlet pressure and temperature ratios account for inlet performance as well as altitude and temperature variations from standard sea level conditions.

The inlet pressure and temperature ratios have been found to be functions of calibrated airspeed, V_c, and the ambient pressure and temperature ratios,

$$\delta_{T_2} = \frac{P_{T_2}}{P_{ssl}} = f\left(V_c,\ \delta_a\right) \qquad \textit{eq 4.9}$$

Where:

δ_{T_2} - Inlet pressure ratio

P_{T_2} - Inlet total pressure

P_{ssl} - Standard sea level pressure

V_c - Calibrated airspeed

δ_a - Ambient pressure ratio

and,

$$\theta_{T_2} = \frac{T_{T_2}}{T_{ssl}} = f\left(V_c, \theta_a\right) \qquad \textit{eq 4.10}$$

Where:

θ_{T_2} - Inlet temperature ratio

T_{T_2} - Inlet total temperature

T_{ssl} - Standard sea level temperature, 288.15 °K

V_c - Calibrated airspeed

θ_a - Ambient temperature ratio.

4.3.3 Engine Performance

Dimensional analysis shows that engine shaft horsepower and fuel flow are a function of N_1, N_2, V_T, δ, θ, and R_n ,

$$\frac{ESHP}{\delta\sqrt{\theta}} = f\left(\frac{N_1}{\sqrt{\theta}}, \frac{N_2}{\sqrt{\theta}}, \frac{V_T}{\sqrt{\theta}}, R_n\right) \qquad \textit{eq 4.11}$$

$$\frac{W_f}{\delta\sqrt{\theta}} = \left(\frac{N_1}{\sqrt{\theta}}, \frac{N_2}{\sqrt{\theta}}, \frac{V_T}{\sqrt{\theta}}, R_n\right) \qquad \textit{eq 4.12}$$

Where:

ESHP - Engine shaft horsepower

W_f - Fuel flow

N_1 - Engine gas generator speed

N_2 - Engine power turbine speed

V_T - True airspeed

R_n - Reynolds number

δ - Pressure ratio

θ - Temperature ratio.

Engine performance data generalizes if the inlet pressure ratio, δ_{T_2}, is substituted for δ, and the inlet temperature ratio, θ_{T_2}, is substituted for θ in the above expression.

Generalized data are data obtained at different test conditions such as airspeeds, altitudes, and temperatures which, when reduced, will define one curve. The generalized engine performance parameters have been termed “corrected”; they are corrected for differences in total temperature and pressure at the engine compressor face. The inlet pressure and temperature ratios are used to correct engine performance data to standard sea level conditions. The corrected engine performance parameters are:

$$ESHP_{corr} = \frac{ESHP}{\delta_{T_2}\sqrt{\theta_{T_2}}}$$

eq 4.13

$$W_{f_{corr}} = \frac{W_f}{\delta_{T_2}\sqrt{\theta_{T_2}}}$$

eq 4.14

$$T_{5_{corr}} = \frac{T_5}{\theta_{T_2}}$$

eq 4.15

$$N_{1_{corr}} = \frac{N_1}{\sqrt{\theta_{T_2}}}$$

eq 4.16

Where:

$ESHP$	- Engine shaft horsepower
$ESHP_{corr}$	- Corrected engine shaft horsepower
W_f	- Fuel flow
$W_{f_{corr}}$	- Corrected fuel flow
T_5	- Power turbine inlet temperature
$T_{5_{corr}}$	- Corrected power turbine inlet temperature
N_1	- Engine gas generator speed
$N_{1_{corr}}$	- Corrected engine gas generator speed
δ_{T_2}	- Inlet pressure ratio
θ_{T_2}	- Inlet temperature ratio.

The corrected engine performance parameters are a function of $N_{1_{corr}}$:

$$ESHP_{corr} = f(N_{1_{corr}}) \qquad \textit{eq 4.17}$$

$$W_{f_{corr}} = f(N_{1_{corr}}) \qquad \textit{eq 4.18}$$

$$T_{5_{corr}} = f(N_{1_{corr}}) \qquad \textit{eq 4.19}$$

Where:

$ESHP_{corr}$	- Corrected engine shaft horsepower
$W_{f_{corr}}$	- Corrected fuel flow
$T_{5_{corr}}$	- Corrected power turbine inlet temperature
$N_{1_{corr}}$	- Corrected engine gas generator speed.

Since both $ESHP_{corr}$ and $W_{f_{corr}}$ are functions of $N_{1_{corr}}$, a relation of $W_{f_{corr}}$ to $ESHP_{corr}$ can be determined,

$$W_{f_{corr}} = f(ESHP_{corr}) \qquad \textit{eq 4.20}$$

Where:

$W_{f_{corr}}$	- Corrected fuel flow
$ESHP_{corr}$	- Corrected engine shaft horsepower.

A calculated engine performance parameter, specific fuel consumption, is often determined and used as a measure of engine performance. Specific fuel consumption (SFC) is often listed as a specification compliance parameter for the engine and airframe manufacturer. SFC is defined as follows:

$$SFC = \frac{W_{f_{corr}}}{ESHP_{corr}} \qquad \textit{eq 4.21}$$

Where:

SFC	- Specific fuel consumption
$W_{f_{corr}}$	- Corrected fuel flow
$ESHP_{corr}$	- Corrected engine shaft horsepower.

4.3.4 Power Available

The maximum power available for a given airspeed and set of atmospheric conditions (H_P and OAT) can be determined from inlet performance and engine performance. Knowing:

$$\delta_{T_2} = \frac{P_{T_2}}{P_{ssl}} = f\left(V_c,\ \delta_a\right) \qquad \textit{eq 4.9}$$

$$\theta_{T_2} = \frac{T_{T_2}}{T_{ssl}} = f\left(V_c, \theta_a\right) \qquad \textit{eq 4.10}$$

$$N_{1_{corr}} = \frac{N_1}{\sqrt{\theta_{T_2}}} \qquad \textit{eq 4.16}$$

$$ESHP_{corr} = f(N_{1_{corr}}) \qquad \textit{eq 4.17}$$

$$W_{f_{corr}} = f(N_{1_{corr}}) \qquad \textit{eq 4.18}$$

$$T_{5_{corr}} = f(N_{1_{corr}}) \qquad \textit{eq 4.19}$$

and having the manufacturer's supplied engine limits:

$ESHP_{max}$	- Maximum permissible engine shaft horsepower
$T_{5_{max}}$	- Maximum permissible power turbine inlet temperature
$W_{f_{max}}$	- Maximum permissible fuel flow
$N_{1_{max}}$	- Maximum permissible engine gas generator speed

the power available (ESHP) can be determined for a given V_c, H_P, and OAT.

4.4 TEST METHODS AND TECHNIQUES

4.4.1 General

Engine performance data are recorded during any steady, unaccelerated flight conditions. Special tests often are not conducted to acquire engine and inlet performance data, although when they are, they typically focus on the extremes of engine performance. During steady-state conditions, power is set for the test point and engine data are recorded along with the data supporting other test purposes. For example, engine and inlet performance data can be obtained along with pitot static system performance data during the course of the airspeed calibration. At each test point, the engine must be allowed to stabilize and sufficient time elapse to obtain a representative fuel flow. All parameters should be monitored during the recording interval for any unusual changes.

4.4.2 Inlet Performance

USNTPS helicopters are not instrumented for inlet performance measurements. However, inlet performance test methods are discussed in this manual.

The inlet performance tests should be planned early in the test program so that test instrumentation, consisting of static and total pressure and temperature sensors, can be incorporated into the total test planning effort. Sufficient lead time must be allowed for the design, fabrication, and airworthiness qualification of the inlet hardware. During flight testing, inlet performance data requirements should be coordinated with the total data recording requirements to include pre-flight and post-flight instrumentation checks. The data recording process must be compatible with the total data collection effort.

The test instrumentation consists of pressure and temperature, static and total, sensors. The sensors must have the appropriate range and response times for the data to be gathered. The sensor probes are mounted on a rake and installed within the inlet (Figure 4.2). Placement of the sensors on the rake is based on profile data obtained during test cell calibrations where large, extensively instrumented engine bellmouths are used.

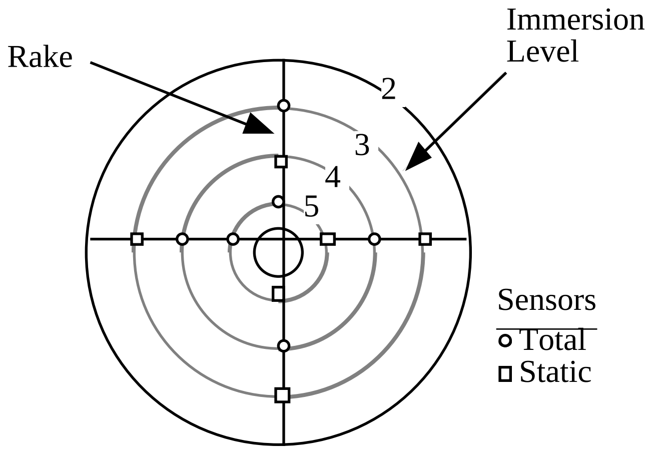


Figure 4.2

Inlet Instrumentation Rake

Experience has shown if averaged inlet performance is sufficient, a minimum of three probes for static and total, temperature and pressure, are used. However, if profile performance across the compressor face is required, a minimum of four azimuth positions with four immersion depths for each sensor is used. If averaged data are used, the average can be displayed on a cockpit visual indicator and manually recorded. Transients are lost. If profile data are required, automatic recording is needed.

Calibration and preflight procedures are important. Total system calibration is required. The calibration and data gathering should start with the static test cell calibration of the engine. A full range of airflows within the engine should be obtained with the specification inlet and exhaust configurations. The test cell configuration should be the same as that used for the flight tests. Inlet performance is a matter of minor variations. Configuration control and documentation of the test article must be maintained during the ground and flight tests.

4.4.2.1 DATA REQUIRED

P_{s_2}, P_{T_2}, T_{s_2}, V_o.

4.4.2.2 TEST CRITERIA

1. Balanced, wings level, unaccelerated flight.
2. Engine bleed air systems off or operated in normal flight mode.
3. Stabilized engine power demand.

4.4.2.3 DATA REQUIREMENTS

1. Stabilize 60 s minimum prior to recording data.
2. Record stabilized data 60 s.
3. $V_o \pm 1$ kn.

4.4.2.4 SAFETY CONSIDERATIONS/RISK MANAGEMENT

Inlet and engine performance data are gathered during all steady-state flight conditions. Usually these tests are combined with other helicopter testing. The conditions and procedures applicable to the associated flight tests are applicable to the inlet and engine performance flight tests. Of particular concern for inlet tests is the location and airworthiness of the inlet instrumentation rake. Prior to installation, the rake must be tested to insure it can withstand safely the dynamic pressures, vibrations, and temperatures that are present or anticipated in the inlet. For engine run ground tests, necessary helicopter restraining devices must be designed, fabricated, and installed.

4.4.3 Engine Performance

Engine performance data are required for all flight regimes and are obtained during steady-state, unaccelerated test points. Usually, inlet and engine performance data are recorded during the same test points. Data points at the extremes of engine power demand are required and obtained in high speed forward flight, high power climbs at high altitudes, at minimum power on the deck, and during power-on autorotational descents (flight idle glides).

4.4.3.1 DATA REQUIRED

Q, N_1, T_5, W_f, N_R, V_o.

4.4.3.2 TEST CRITERIA

1. Balanced, wings level, unaccelerated flight.
2. Engine bleed air systems off or operated in normal flight mode.
3. Stabilized engine power demand.

4.4.3.3 DATA REQUIREMENTS

1. Stabilize 60 s minimum prior to recording data.
2. Record stabilized data 60 s.
3. $N_R \pm 0.2\%$.
4. $N_1 \pm 0.1\%$.
5. $T_5 \pm 5$ °C.
6. $V_o \pm 1$ kn.

4.4.3.4 SAFETY CONSIDERATIONS/RISK MANAGEMENT

Know and adhere to normal operating procedures and all limitations. Exercise care at the high power points so engine limitations are not exceeded. Consider the cockpit displays and any lags or hysteresis present.

4.4.4 Power Available

Engine power available is computed from inlet and engine performance data. Separate tests are not required.

4.5 DATA REDUCTION

4.5.1 General

The USNTPS helicopters are not instrumented for inlet performance data. Therefore, inlet performance data are provided to the students to use for engine performance calculations.

USNTPS uses computer based data reduction for helicopter performance. Data reduction routines have been developed for engine performance and power available calculations. This section concentrates on manual data reduction for inlet and engine performance and power available. A general introduction to computer based data reduction is included.

Observed cockpit data must be reduced to indicated data by applying the appropriate instrument correction factors prior to further reduction.

4.5.2 Inlet Performance

The following equations are used to reduce inlet performance data:

$$\bar{P}_{s_2} = \frac{\sum P_{s_2}}{n} \qquad \textit{eq 4.22}$$

Where:

$\bar{P}_{s_2}$ - Average inlet static pressure

P_{s_2} - Inlet static pressure (one per probe)

n - Number of probes.

$$\bar{P}_{T_2} = \frac{\sum P_{T_2}}{n} \qquad \textit{eq 4.23}$$

Where:

$\bar{P}_{T_2}$ - Average inlet total pressure

P_{T_2} - Inlet total pressure

n - Number of probes.

$$M_2 = \sqrt{\left(\frac{2}{\gamma - 1}\right)\left[\left(\frac{\bar{P}_{T_2}}{\bar{P}_{s_2}}\right)^{\frac{\gamma - 1}{\gamma}} - 1\right]} \qquad \textit{eq 4.24}$$

Where:

M_2 - Inlet Mach number

γ - Ratio of specific heats (Air: γ = 1.4)

$\bar{P}_{T_2}$ - Average inlet total pressure

$\bar{P}_{s_2}$ - Average inlet static pressure.

$$T_{T_2} = T_{s_2}\left[1 + \left(\frac{\gamma - 1}{2}\right)M_2^2\right] \qquad \textit{eq 4.25}$$

Where:

T_{T_2} - Inlet total temperature

T_{s_2} - Inlet static temperature

γ - Ratio of specific heats (Air: $\gamma = 1.4$)

M_2 - Inlet Mach number.

$$\delta_{T_2} = \frac{\bar{P}_{T_2}}{P_{ssl}} = \left(\frac{\bar{P}_{T_2}}{P_a}\right)\left(\frac{P_a}{P_{ssl}}\right) = \left(\frac{\bar{P}_{T_2}}{P_a}\right)\delta_a \qquad \textit{eq 4.26}$$

Where:

δ_{T_2} - Inlet pressure ratio

$\bar{P}_{T_2}$ - Average inlet total pressure

P_a - Ambient pressure (free stream, ahead of inlet)

P_{ssl} - Standard sea level pressure

δ_a - Ambient pressure ratio

$$\theta_{T_2} = \frac{T_{T_2}}{T_{ssl}} = \frac{\left(T_{T_2} - T_a\right)}{T_{ssl}} + \frac{T_a}{T_{ssl}} = \frac{\left(T_{T_2} - T_a\right)}{T_{ssl}} + \theta_a \qquad \textit{eq 4.26a}$$

Where:

θ_{T_2} - Inlet temperature ratio

T_{T_2} - Average Inlet total temperature ratio

T_a - Ambient temperature (free stream, ahead of inlet)

T_{ssl} - Standard sea level temperature

θ_a - Ambient temperature ratio

$$V_c = V_o + \Delta V_{ic} + \Delta V_{pos}$$ *eq 4.27*

Where:

V_c	- Calibrated airspeed
V_o	- Observed airspeed
ΔV_{ic}	- Airspeed instrument correction
ΔV_{pos}	- Airspeed position error.

From the observed inlet performance data, compute δ_{T_2}, θ_{T_2}, and V_c as follows:

Parameter	Notation	Formula	Units	Remarks
(1) Inlet static pressure	P_{s_2}		psi	measured
(2) Inlet total pressure	P_{T_2}		psi	measured
(3) Average inlet static pressure	$\bar{P}_{s_2}$	eq 4.22	psi	
(4) Average inlet total pressure	$\bar{P}_{T_2}$	eq 4.23	psi	
(5) Inlet Mach number	M_2	eq 4.24		
(6) Inlet static temperature	T_{s_2}		°K	measured
(7) Inlet total temperature	T_{T_2}	eq 4.25	°K	
(8) Inlet temperature ratio	θ_{T_2}	eq 4.26		T_{ssl} = 288.15°K

(9) Inlet pressure ratio	δ_{T_2}	eq 4.26a	P_{ssl} = 14.7 psi
(10) Calibrated Airspeed	V_c	eq 4.27	KCAS

Plot inlet pressure recovery ratio, $\left(\frac{\bar{P}_{T_2}}{P_a}\right)$, as a function of calibrated airspeed, V_c (Figure 4.3).

Plot inlet temperature difference, $T_{T_2} - T_a$, as a function of calibrated airspeed, V_c (Figure 4.4).

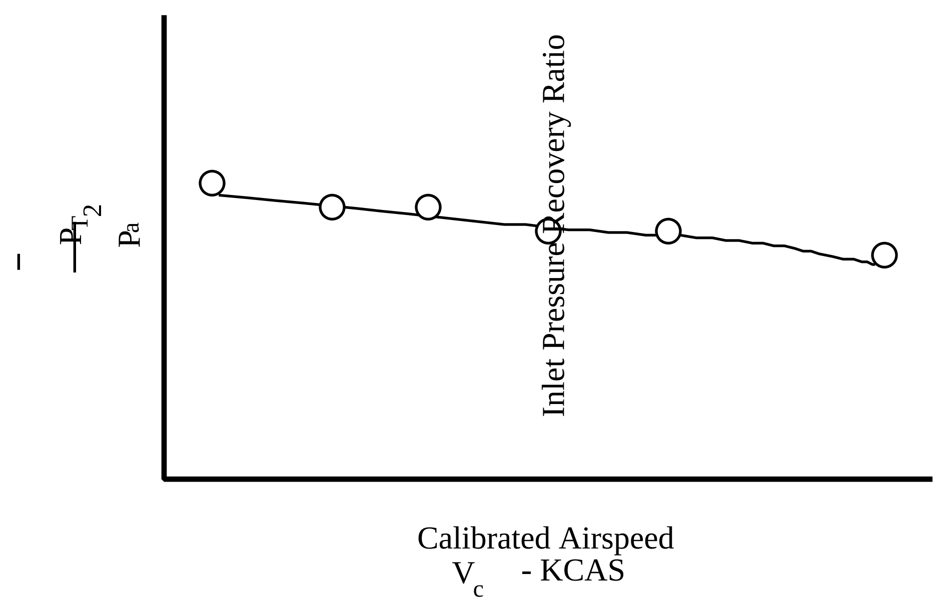


Figure 4.3
Inlet Pressure Recovery Ratio Characteristics

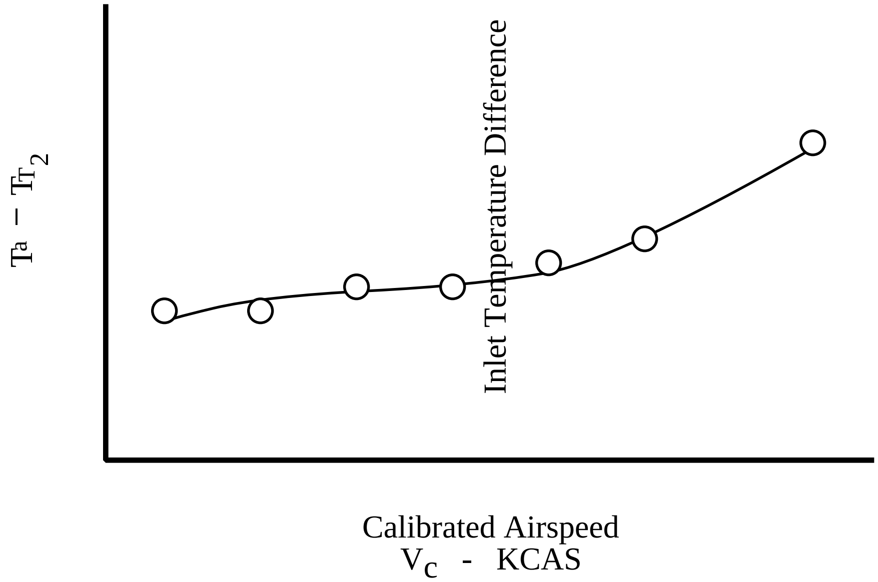


Figure 4.4
Inlet Temperature Difference Characteristics

4.5.3 Engine Performance

The following equations are used in engine performance data reduction:

$$ESHP = (K_Q)(Q)(N_R) \quad \textit{eq 4.28}$$

Where:

K_Q - Engine torque constant, with applicable units for the specific helicopter to convert measured Q and N_R to ESHP - hp

Q - Engine torque

N_R - Main rotor speed.

$$ESHP_{corr} = \frac{ESHP}{\delta_{T_2}\sqrt{\theta_{T_2}}} \quad \textit{eq 4.13}$$

$$N_{1_{corr}} = \frac{N_1}{\sqrt{\theta_{T_2}}}$$ *eq 4.16*

$$T_{5_{corr}} = \frac{T_5}{\theta_{T_2}}$$ *eq 4.15*

$$W_{f_{corr}} = \frac{W_f}{\delta_{T_2}\sqrt{\theta_{T_2}}}$$ *eq 4.14*

$$SFC = \frac{W_{f_{corr}}}{ESHP_{corr}}$$ *eq 4.21*

From observed engine performance data, calculate the following corrected engine parameters for a specific V_c and corresponding δ_{T_2} , θ_{T_2} (Figures 4.3 and 4.4 with Equations 4.26 and 4.26a):

Parameter	Notation	Formula	Units	Remarks
(1) Engine shaft horsepower	ESHP	eq 4.28	hp	
(2) Corrected engine shaft horsepower	$ESHP_{corr}$	eq 4.13	hp	θ_{T_2}, Figure 4.4 and Equation 4.26a δ_{T_2} , Figure 4.3 and Equation 4.26
(3) Corrected engine gas generator speed	$N_{1_{corr}}$	eq 4.16	%	
(4) Corrected power turbine inlet temperature	$T_{5_{corr}}$	eq 4.15	°C	

(5) Corrected fuel flow	$W_{f_{corr}}$	eq 4.14	lb/h
(6) Specific fuel consumption	SFC	eq 4.21	lb/h/hp

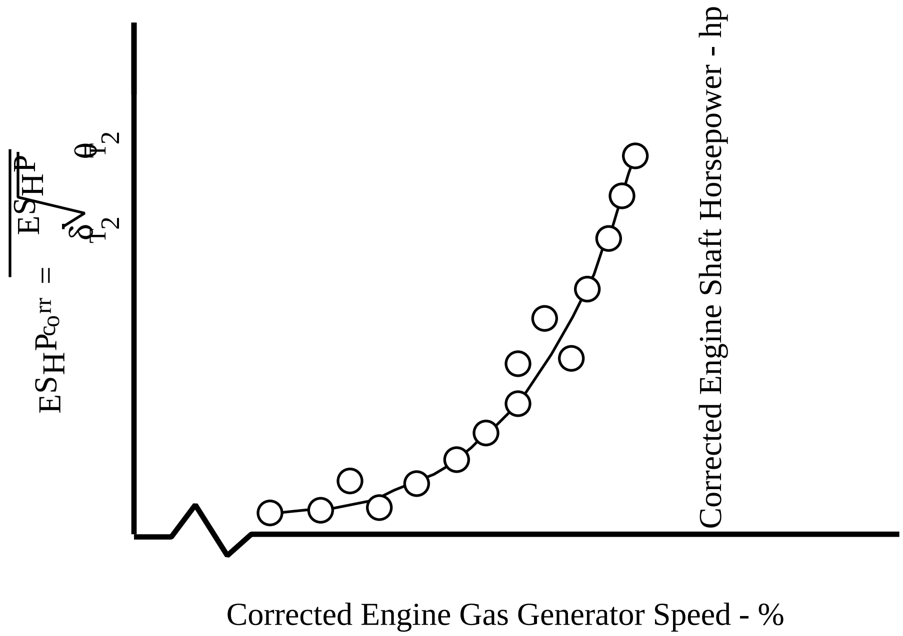
Plot $ESHP_{corr}$ as a function of $N1_{corr}$ (Figure 4.5).

Plot $T5_{corr}$ as a function of $N1_{corr}$ (Figure 4.6).

Plot $W_{f_{corr}}$ as a function of $N1_{corr}$ (Figure 4.7).

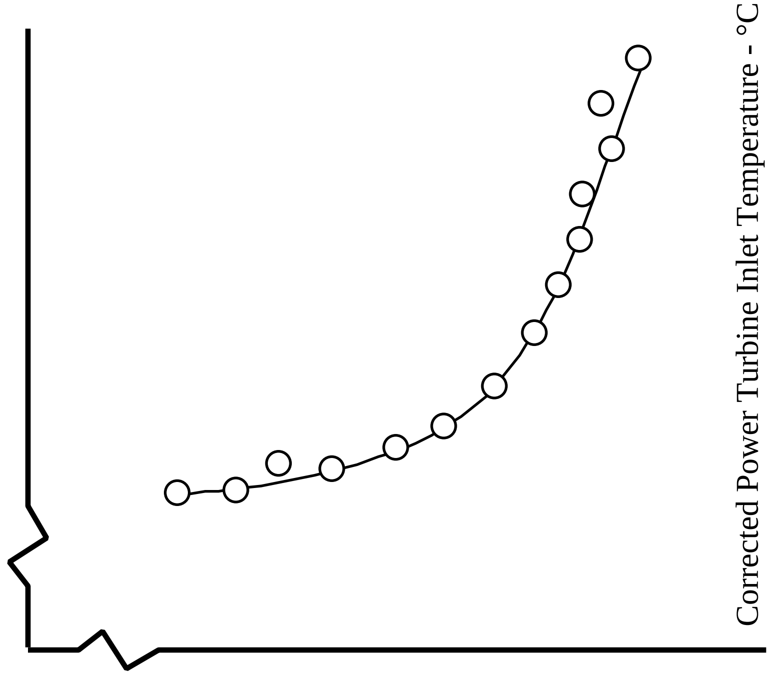
Cross plot $W_{f_{corr}}$ as a function of $ESHP_{corr}$ from Figures 4.5 and 4.7 (Figure 4.8).

Plot SFC as a function of $N1_{corr}$ (Figure 4.9).



$$N_{1\,corr} = \frac{N_1}{\sqrt{\theta_{T_2}}}$$

Figure 4.5
Corrected Engine Shaft Horsepower
with Corrected Engine Gas Generator Speed



$$T_{5_{corr}} = \frac{T_5}{\theta_{T_2}}$$

$$N_{1_{corr}} = \frac{N_1}{\sqrt{\theta_{T_2}}}$$

Figure 4.6
Corrected Power Turbine Inlet Temperature with Corrected Engine Gas Generator Speed

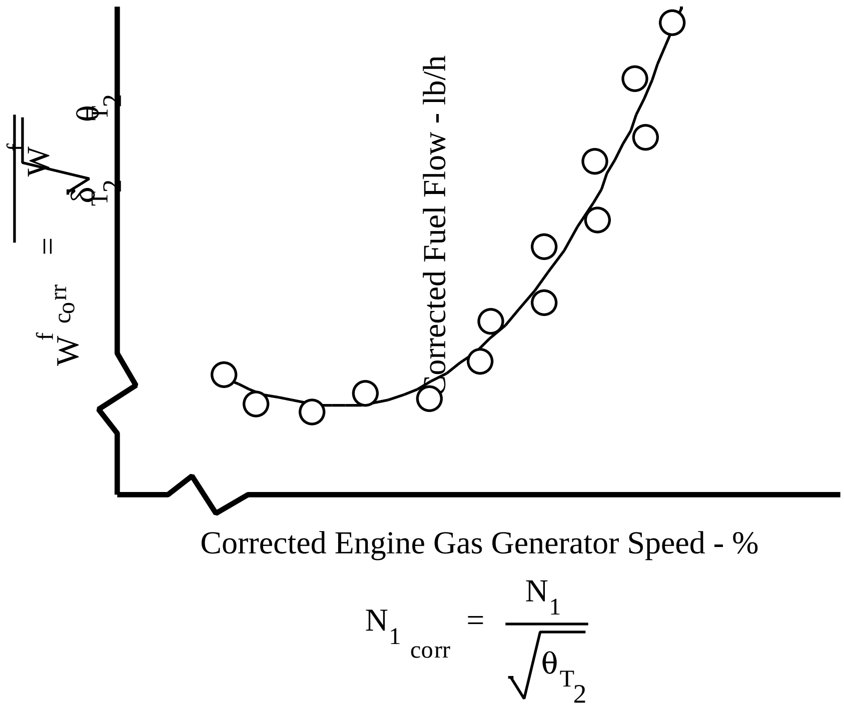
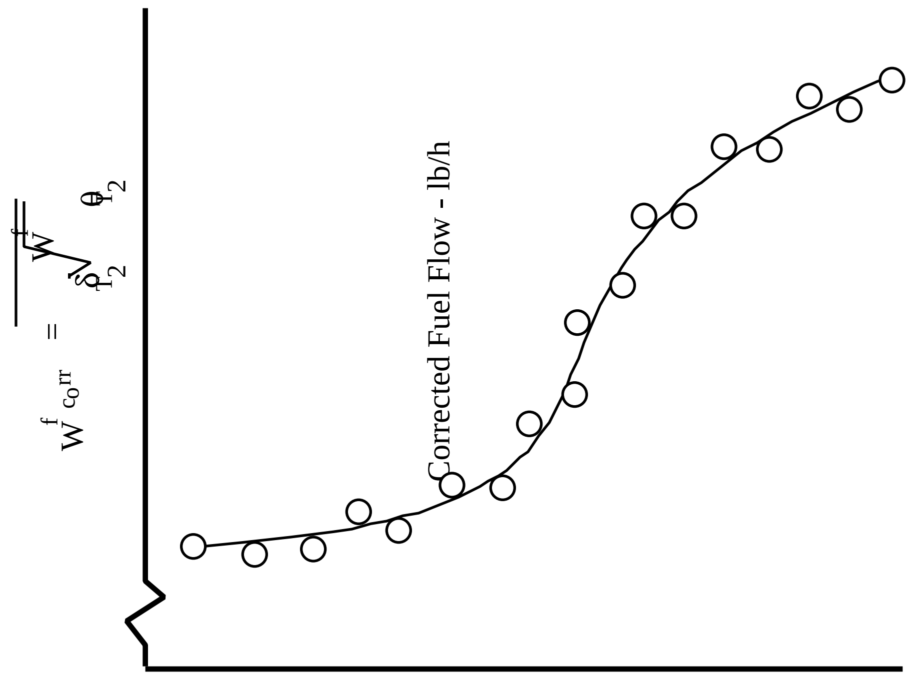


Figure 4.7
Corrected Fuel Flow with Corrected Engine Gas Generator Speed



$$ESHP_{corr} = \frac{ESHP}{\delta_{T_2}\sqrt{\theta_{T_2}}}$$

Figure 4.8
Corrected Fuel Flow with Corrected Engine Shaft Horsepower

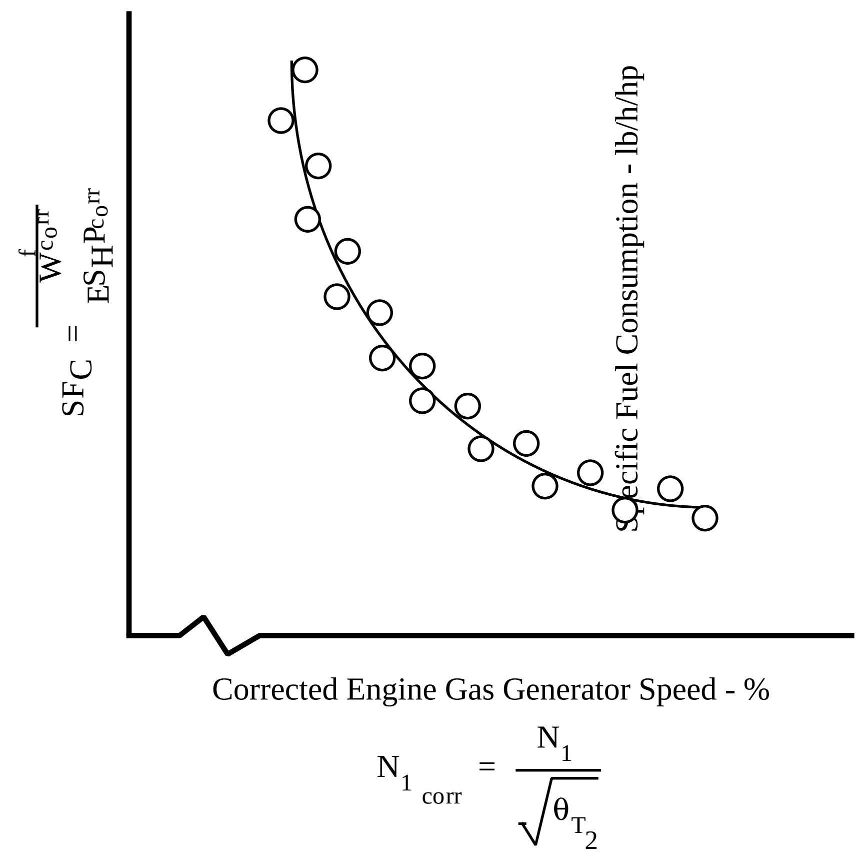


Figure 4.9
Specific Fuel Consumption with Corrected Engine Gas Generator Speed

4.5.4 Power Available

The following equations are used in power available data reduction:

$$\left(N_{1_{corr}}\right)_{max} = \frac{N_{1_{max}}}{\sqrt{\theta_{T_2}}} \qquad \text{eq 4.29}$$

$$\left(T_{5_{corr}}\right)_{max} = \frac{T_{5_{max}}}{\theta_{T_2}} \qquad \text{eq 4.30}$$

$$\left(W_{f_{corr}}\right)_{max} = \frac{W_{f_{max}}}{\delta_{T_2}\sqrt{\theta_{T_2}}} \qquad \text{eq 4.31}$$

Where:

$(N_{1_{corr}})_{max}$	- Maximum corrected engine gas generator speed
$(T_{5_{corr}})_{max}$	- Maximum corrected power turbine inlet temperature
$(W_{f_{corr}})_{max}$	- Maximum corrected fuel flow
$N_{1_{max}}$	- Maximum permissible engine gas generator speed
$T_{5_{max}}$	- Maximum permissible power turbine inlet temperature
$W_{f_{max}}$	- Maximum permissible fuel flow
δ_{T_2}	- Inlet pressure ratio
θ_{T_2}	- Inlet temperature ratio.

$$ESHP_A = \left(\frac{ESHP}{\delta_{T_2}\sqrt{\theta_{T_2}}}\right)_{max} \left(\delta_{T_2}\sqrt{\theta_{T_2}}\right) \qquad \textit{eq 4.32}$$

Where:

$ESHP_A$	- Engine shaft horsepower available
ESHP	- Engine shaft horsepower
δ_{T_2}	- Inlet pressure ratio
θ_{T_2}	- Inlet temperature ratio.

From corrected engine performance parameters, Figures 4.3, 4.4, 4.5, 4.6, and 4.7 and Equations 4.26 and 4.26a, calculate power available for a specific V_c as a function of H_P and calculate power available for a specific H_P as a function of V_T as follows:

Parameter	Notation	Formula	Units	Remarks
(1) Inlet pressure	δ_{T_2}	eq 4.26		$\left(\frac{\bar{P}_{T_2}}{P_a}\right)$ from Figure 4.3 ratio for specific V_c

(2) Inlet temperature	θ_{T_2}	eq 4.26a		$\left(T_{T_2} - T_a\right)$ from Figure 4.4 ratio for specific V_c
(3) Maximum corrected engine gas generator speed	$(N_{1_{corr}})_{max}$	eq 4.29	%	
(4) Maximum corrected power turbine inlet temperature	$(T_{5_{corr}})_{max}$	eq 4.30	°C	
(5) Maximum corrected fuel flow	$(W_{f_{corr}})_{max}$	eq 4.31	lb/h	
(6) Corrected engine gas generator speed limit	$(N_{1_{corr}})_{limit}$		%	Greatest value of $(N_{1_{corr}})_{max}$ corresponding to (3) or (4) using Figure 4.6 and (5) using Figure 4.7
(7) Maximum corrected ESHP	$(ESHP_{corr})_{max}$		hp	Value corresponding to (6) using Figure 4.5
(8) Engine shaft horsepower available	$ESHP_A$	eq 4.32	hp	
(9) Repeat for increasing H_P for a standard day. Use Standard Atmosphere Tables to reduce δ and θ.				
(10) Repeat for increasing H_P for a hot day. Use Standard Atmosphere Tables and hot day lapse rate.				
(11) True airspeed	V_T	See Chapter 2	KTAS	For a V_c of interest

Plot power available ($ESHP_A$) as a function of H_P for standard day conditions and hot day conditions for a specific V_c (Figure 4.10).

Plot power available as a function of V_T for a specific H_P (Figure 4.11).

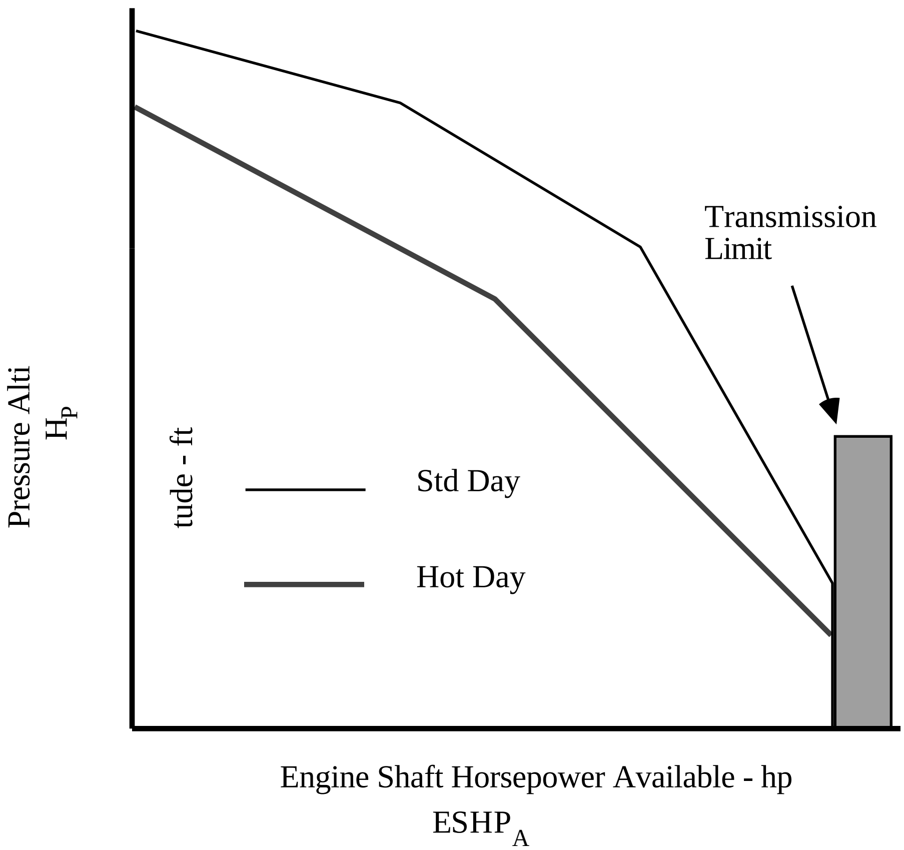


Figure 4.10
Engine Power Available with Pressure Altitude

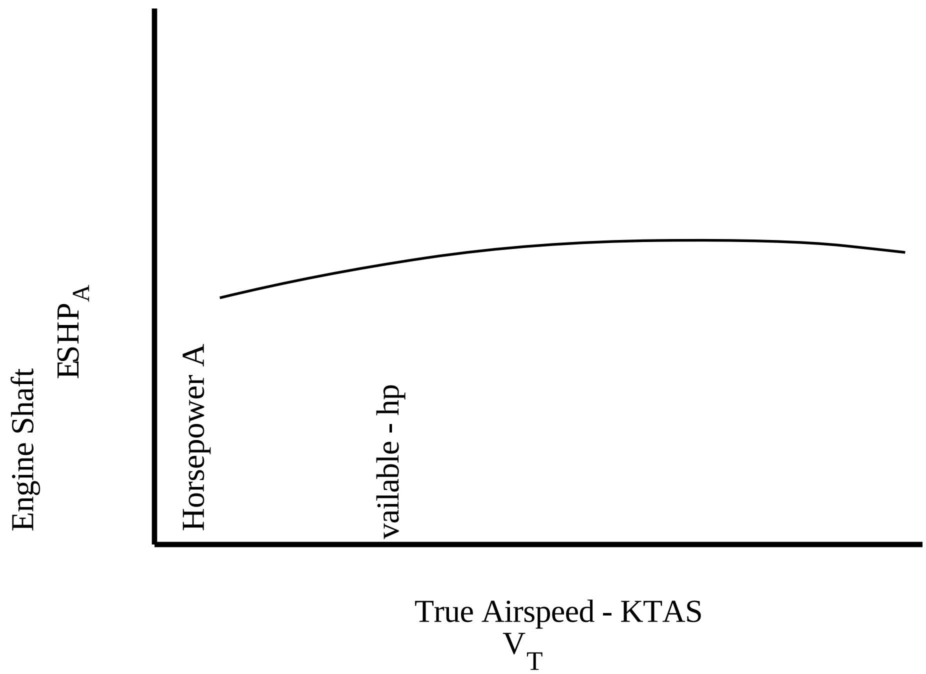


Figure 4.11
Engine Power Available with True Airspeed

4.5.5 USNTPS Engine Performance Computer Data Reduction

Inlet performance can be presented in the form:

$$\frac{P_{T_2}}{P_a} = f(V_c) \qquad \text{eq 4.33}$$

$$\left(T_{T_2} - T_a\right) = f(V_c) \qquad \text{eq 4.34}$$

Where:

P_{T_2}	- Inlet total pressure
P_a	- Ambient pressure
T_{T_2}	- Inlet total temperature
T_a	- Ambient temperature
V_c	- Calibrated airspeed.

USNTPS helicopters are not instrumented for inlet performance data. Therefore, inlet performance data are provided to the student. The data are in the form of the coefficients for the third degree polynomials:

$$\frac{P_{T_2}}{P_a} = A_0 + A_1 V_c + A_2 V_c^2 + A_3 V_c^3 \qquad \text{eq 4.35}$$

$$\left(T_{T_2} - T_a\right) = B_0 + B_1 V_c + B_2 V_c^2 + B_3 V_c^3 \qquad \text{eq 4.36}$$

Where:

P_{T_2}	- Inlet total pressure
P_a	- Ambient pressure
T_{T_2}	- Inlet total temperature
T_a	- Ambient temperature
A_0, A_1, A_2, A_3	- Curve fit coefficient constants
B_0, B_1, B_2, B_3	- Curve fit coefficient constants
V_c	- Calibrated airspeed.

Students input the inlet performance data in the coefficient form along with recorded data (N_1, Q, N_R, W_f, T_5, H_P , OAT, V_c) into the ENGINE computer program to compute $ESHP_{corr}$, $W_{f_{corr}}$, $T_{5_{corr}}$, $N_{1_{corr}}$, and SFC. Data are output to the line printer and furnished as Figures 4.5, 4.6, 4.7, 4.8, and 4.9. Third degree polynomial curve fit coefficients are provided as well.

The following logic is used to compute the corrected engine parameters.

(1) Ambient Pressure:

$$\frac{P_a}{P_{ssl}} = \left[1 - \left(\frac{\lambda_{ssl} H_P}{T_{ssl}}\right)\right]^{\frac{g_{ssl}}{g_c\ \lambda_{ssl}\ R}} \qquad \text{eq 4.37}$$

Where:

P_a - Ambient pressure

P_{ssl} - Standard sea level pressure

H_P - Pressure altitude

λ_{ssl} - Standard sea level lapse rate, 0.00198 °K/ft

g_{ssl} - Standard sea level gravity, 32.17 ft/ s^2

g_c - 32.17 lb - mass / slug

R - Engineering gas constant for air, 96.03 ft - lb/ lb_m- °K.

(2) Ambient Temperature:

$$T_a = \frac{OAT + 273.15}{1 + 0.2K_T\left[\frac{\left(\frac{V_c^2}{\delta}\right)}{a_{ssl}^2}\right]} \qquad \textit{eq 4.38}$$

Where:

T_a - Ambient temperature, °K

OAT - Outside air temperature (total temperature), °C

K_T - Temperature recovery factor (assume = 0.95)

V_c - Calibrated airspeed, KCAS

δ - Pressure ratio

a_{ssl} -Standard sea level speed of sound, 661.23 kn.

(3) From Equation 4.35, Equation 4.37, and P_{ssl}, calculate:

$$P_{T_2} = \left(\frac{P_{T_2}}{P_a}\right)(P_a) \qquad \textit{eq 4.39}$$

Where:

P_{T_2} - Inlet pressure ratio

P_a - Ambient pressure.

(4) From Equation 4.36 and Equation 4.38, calculate:

$$T_{T_2} = (T_{T_2} - T_a) + T_a \qquad \textit{eq 4.40}$$

Where:

T_{T_2} - Inlet total temperature

T_a - Ambient temperature.

(5) From Equation 4.39 and Equation 4.40, calculate:

$$\delta_{T_2} = \frac{P_{T_2}}{P_{ssl}} \qquad \textit{eq 4.7}$$

$$\theta_{T_2} = \frac{T_{T_2}}{T_{ssl}} \qquad \textit{eq 4.8}$$

(6) From observed data, calculate:

$$ESHP = (K_Q)(Q)(N_R) \qquad \textit{eq 4.28}$$

(7) The corrected engine parameters are then computed for a given airspeed, V_c:

$$ESHP_{corr} = \frac{ESHP}{\delta_{T_2}\sqrt{\theta_{T_2}}} \qquad \textit{eq 4.13}$$

$$W_{f_{corr}} = \frac{W_f}{\delta_{T_2}\sqrt{\theta_{T_2}}} \qquad \textit{eq 4.14}$$

$$T_{5_{corr}} = \frac{T_5}{\theta_{T_2}}$$ *eq 4.15*

$$N_{1_{corr}} = \frac{N_1}{\sqrt{\theta_{T_2}}}$$ *eq 4.16*

$$SFC = \frac{W_{f_{corr}}}{ESHP_{corr}}$$ *eq 4.21*

4.5.6 USNTPS Power Available Computer Data Reduction

Corrected engine performance parameters are in the form:

$$ESHP_{corr} = \frac{ESHP}{\delta_{T_2}\sqrt{\theta_{T_2}}}$$ *eq 4.13*

$$W_{f_{corr}} = \frac{W_f}{\delta_{T_2}\sqrt{\theta_{T_2}}}$$ *eq 4.14*

$$T_{5_{corr}} = \frac{T_5}{\theta_{T_2}}$$ *eq 4.15*

$$N_{1_{corr}} = \frac{N_1}{\sqrt{\theta_{T_2}}}$$ *eq 4.16*

and are a function of $N1_{corr}$ as follows:

$$ESHP_{corr} = f(N_{1_{corr}}) \qquad \textit{eq 4.17}$$

$$W_{f_{corr}} = f(N_{1_{corr}}) \qquad \textit{eq 4.18}$$

$$T_{5_{corr}} = f(N_{1_{corr}}) \qquad \textit{eq 4.19}$$

The nature of the functions are assumed to be a third degree polynomial in the form:

$$ESHP_{corr} = C_0 + C_1(N_{1_{corr}}) + C_2(N_{1_{corr}})^2 + C_3(N_{1_{corr}})^3 \qquad \textit{eq 4.41}$$

$$W_{f_{corr}} = D_0 + D_1(N_{1_{corr}}) + D_2(N_{1_{corr}})^2 + D_3(N_{1_{corr}})^3 \qquad \textit{eq 4.42}$$

$$T_{5_{corr}} = E_0 + E_1(N_{1_{corr}}) + E_2(N_{1_{corr}})^2 + E_3(N_{1_{corr}})^3 \qquad \textit{eq 4.43}$$

Where:

$ESHP_{corr}$	- Corrected engine shaft horsepower
$W_{f_{corr}}$	- Corrected fuel flow
$T5_{corr}$	- Corrected power turbine inlet temperature
$N1_{corr}$	- Corrected engine gas generator speed
C_0, C_1, C_2, C_3	- Curve fit coefficient constants
D_0, D_1, D_2, D_3	- Curve fit coefficient constants
E_0, E_1, E_2, E_3	- Curve fit coefficient constants.

Given:

$$\frac{P_{T_2}}{P_a} = f(V_c) \qquad \textit{eq 4.33}$$

$$\left(T_{T_2} - T_a\right) = f(V_c) \qquad \textit{eq 4.34}$$

$$\frac{P_{T_2}}{P_a} = A_0 + A_1 V_c + A_2\, V_c^2 + A_3\, V_c^3 \qquad \textit{eq 4.35}$$

$$\left(T_{T_2} - T_a\right) = B_0 + B_1 V_c + B_2\, V_c^2 + B_3\, V_c^3 \qquad \textit{eq 4.36}$$

and given the curve fit coefficients for Equations 4.35, 4.36, 4.41, 4.42 and 4.43 from the ENGINE program and the manufacturer's supplied limits:

$ESHP_{max}$	- Maximum permissible engine shaft horsepower
$T5_{max}$	- Maximum permissible power turbine inlet temperature
$W_{f_{max}}$	- Maximum permissible fuel flow
$N_{1_{max}}$	- Maximum permissible engine gas generator speed

the maximum $ESHP_A$ is calculated for a given airspeed and for standard temperature conditions or hot day conditions from sea level to 20,000 ft H_P as follows:

$$ESHP_A = \left(\frac{ESHP}{\delta_{T_2}\sqrt{\theta_{T_2}}}\right)_{max} \left(\delta_{T_2}\sqrt{\theta_{T_2}}\right) \qquad \textit{eq 4.32}$$

Students input data into the PWRAVL computer program as the curve fit coefficients and airspeed of interest (up to four different airspeeds). The observed engine performance data were entered previously into the ENGINE program. The power available data are output as a tabular listing of ESHP up to 20,000 ft H_P for the selected airspeeds. Data are also output as a plot of power available as a function of H_P for both standard day and hot day conditions (Figure 4.10).

In summary, the logic of the PWRAVL program is such that the program first determines the maximum power available consistent with N_1 limits, ESHP limits, T_5 limits, and W_f limits. For each H_P from sea level to 20,000 ft, the program cycles through the logic twice, once for standard day conditions and once for hot day conditions. USNTPS data reduction programs use MIL STD 210A definitions for standard day and hot day conditions. The process is repeated for up to four airspeeds.

4.6 DATA ANALYSIS

Inlet performance, as measured by the inlet pressure and temperature ratios, is used to determine corrected engine performance as described by $N_{1_{corr}}$, $ESHP_{corr}$, $T_{5_{corr}}$, and $W_{f_{corr}}$. Specific fuel consumption, as a measure of engine efficiency, is determined from W_f and ESHP. Both inlet performance and corrected engine performance are used to determine power available as a function of altitude for a given airspeed, or as a function of airspeed for a given altitude.

The importance of engine performance (power available) is the impact on aircraft performance parameters when combined with airframe power required. Subsequent chapters deal with power required. Engine power available as developed in this chapter is combined with airframe power required to determine aircraft performance parameters.

4.7 MISSION SUITABILITY

As with data analysis, mission suitability aspects of engine performance alone is not considered. Engine performance (power available) when combined with airframe performance (power required) will produce performance parameters for the complete aircraft.

4.8 SPECIFICATION COMPLIANCE

There are no general MilSpecs for engine performance. Consult each helicopter or engine specification for applicability. Often, maximum specific fuel consumption and minimum power available are specification requirements.

4.9 GLOSSARY

4.9.1 Notations

a_{ssl}	Standard sea level speed of sound	661.23 kn
A_0, A_1, A_2, A_3	Constants	
B_0, B_1, B_2, B_3	Constants	
C_0, C_1, C_2, C_3	Constants	
D_0, D_1, D_2, D_3	Constants	
E_0, E_1, E_2, E_3	Constants	
ESHP	Engine shaft horsepower	hp
$ESHP_A$	Engine shaft horsepower available	hp
$ESHP_{corr}$	Corrected engine shaft horsepower	hp
$ESHP_{max}$	Maximum permissible engine shaft horsepower	hp
$(ESHP_{corr})_{max}$	Maximum corrected engine shaft horsepower	hp
FC	Fuel counts	cts
FU	Fuel used	lb
g_{ssl}	Standard sea level gravity	32.17 ft/ s^2
g_c	32.17 lb - mass / slug	
GW	Gross weight	lb
H_P	Pressure altitude	ft
H_{P_o}	Observed pressure altitude	ft
K_Q	Engine torque constant	
K_T	Temperature recovery factor	
M_2	Inlet Mach number	
MilSpec	Military specification	
n	Number of probes	
N_R	Main rotor speed	%, rpm
N_1	Engine gas generator speed	%, rpm
$N_{1_{corr}}$	Corrected engine gas generator speed	%, rpm

$N_{1_{max}}$	Maximum permissible engine gas generator speed	%, rpm
$(N_{1_{corr}})_{max}$	Maximum corrected engine gas generator speed	%, rpm
$(N_{1_{corr}})_{limit}$	Corrected engine gas generator speed limit	%, rpm
N_2	Engine power turbine speed	%, rpm
OAT	Outside air temperature (total temperature)	°C
P	Pressure	psi, in H_2O, psf
P_a	Ambient pressure	psi, psf
P_s	Static pressure	psi, psf
P_{ssl}	Standard sea level pressure	14.7 psi, 2116.2 psf
P_{s_2}	Inlet static pressure (at engine compressor face)	psi, psf
$\bar{P}_{s_2}$	Average inlet static pressure (at engine compressor face)	psi, psf
P_T	Total pressure	psi, psf
P_{T_2}	Inlet total pressure (at engine compressor face)	psi, psf
$\bar{P}_{T_2}$	Average inlet total pressure (at engine compressor face)	psi, psf
Q	Engine torque	%, psi, lb-in
R	Engineering gas constant for air	96.03 ft - lb/ lb_m- °K
R_n	Reynolds number	
SFC	Specific fuel consumption	lb/h/hp
T	Temperature	°C, °K
T_5	Power turbine inlet temperature	°C
$T_{5_{corr}}$	Corrected power turbine inlet temperature	°C
$T_{5_{max}}$	Maximum permissible power turbine inlet temperature	°C
$(T_{5_{corr}})_{max}$	Maximum corrected power turbine inlet temperature	°C
T_a	Ambient temperature	°C
T_o	Observed temperature	°C

T_s	Static temperature	°C
T_{s_2}	Inlet static temperature (at engine compressor face)	°C
T_{ssl}	Standard sea level temperature	15°C, 288.15°K
T_T	Total temperature	°C
T_{T_2}	Inlet total temperature (at engine compressor face)	°C
V_c	Calibrated airspeed	KCAS
V_o	Observed airspeed	KOAS
V_T	True airspeed	KTAS
ΔV_{ic}	Airspeed instrument correction	kn
ΔV_{pos}	Airspeed position error	kn
W_f	Fuel flow	lb/h
$W_{f_{corr}}$	Corrected fuel flow	lb/h
$W_{f_{max}}$	Maximum permissible fuel flow	lb/h
$(W_{f_{corr}})_{max}$	Maximum corrected fuel flow	lb/h

4.9.2 Greek Symbols

δ (Delta)	Pressure ratio	
δ_a	Ambient pressure ratio	
δ_{s_2}	Static pressure ratio	
δ_{T_2}	Inlet pressure ratio	
γ (Gamma)	Ratio of specific heats	
λ (Lambda)	Temperature lapse rate	°C, ∞K/ft
λ_{ssl}	Standard sea level lapse rate	0.00198 °K/ft
θ (Theta)	Temperature ratio	
θ_a	Ambient temperature ratio	
θ_{s_2}	Static temperature ratio	
θ_{T_2}	Inlet temperature ratio	

4.10 REFERENCES

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CHAPTER FIVE

HOVER PERFORMANCE

CHAPTER FIVE

FIGURES

CHAPTER FIVE

EQUATIONS

$$dR = dL + dD_0 \qquad \textit{eq 5.1}$$

$$dR = dT + dF \qquad \textit{eq 5.2}$$

$$dT = dL\cos\alpha_i - dD_0 \sin\alpha_i \qquad \textit{eq 5.3}$$

$$dF = dL\sin\alpha_i + dD_0 \cos\alpha_i \qquad \textit{eq 5.4}$$

$$dL = C_\ell q_e \, dS \qquad \textit{eq 5.5}$$

$$dD_0 = C_{d_0} q_e \, dS \qquad \textit{eq 5.6}$$

$$dT = \frac{\rho(\Omega r)^2}{2\cos^2\alpha_i} c\,dr \left(C_\ell \cos\alpha_i - C_{d_0} \sin\alpha_i\right) \qquad \textit{eq 5.7}$$

$$dT = \frac{1}{2} \rho (\Omega r)^2 c\,dr\,C_\ell \qquad \textit{eq 5.8}$$

$$T_{perblade} = \int dT = \int \frac{1}{2} \rho (\Omega r)^2 c\,dr\,C_\ell \qquad \textit{eq 5.9}$$

$$T_b = \frac{1}{6} \sigma_R \bar{C}_\ell \rho A_D (\Omega R)^2 \qquad \textit{eq 5.10}$$

$$dF = \frac{\rho(\Omega r)^2}{2\cos^2\alpha_i} c\,dr \left(C_\ell \sin\alpha_i + C_{d_0} \cos\alpha_i\right) \qquad \textit{eq 5.11}$$

$$dF = \frac{1}{2}\rho\,(\Omega r)^2\ c\ dr\ (C_{d_i} + C_{d_0})$$ *eq 5.12*

$$dP = dF\,(r)\,\Omega$$ *eq 5.13*

$$P_b = \frac{1}{8}\,\sigma_R\,(\overline{C}_{d_0} + \overline{C}_{d_i})\ \rho\ A_D\ (\Omega R)^3$$ *eq 5.14*

$$P_0 = \frac{1}{8}\,\sigma_R\ \overline{C}_{d_0}\ \rho\ A_D\ (\Omega R)^3$$ *eq 5.15*

$$P_{T_u} = P_a = C_1$$ *eq 5.16*

$$P_{T_d} = P_a + q_w = C_2$$ *eq 5.17*

$$\Delta P = P_{T_d} - P_{T_u} = q_w$$ *eq 5.18*

$$\frac{T}{A_D} = \Delta P = \frac{1}{2}\rho\, v_w^2$$ *eq 5.19*

$$F = \frac{dm}{dt}\ \Delta V$$ *eq 5.20*

$$(\Delta P)\ A_D = \rho\ A_D\ (v_w)\ v_i$$ *eq 5.21*

$$T = \frac{dm}{dt}\ \Delta D = 2\ \rho\ A_D\ v_i^2$$ *eq 5.22*

$$v_i = \sqrt{\frac{T}{2\,\rho\,A_D}} = \sqrt{\frac{W}{2\,\rho\,A_D}}$$ *eq 5.23*

$$P_i = T v_i = \sqrt{\frac{T^3}{2\,\rho\,A_D}} = \sqrt{\frac{W^3}{2\,\rho\,A_D}}$$ *eq 5.24*

$$\eta_m = \frac{RSHP}{ESHP} \qquad \textit{eq 5.25}$$

$$P_{MR} = P_i + P_0 \qquad \textit{eq 5.26}$$

$$P_{MR} = \sqrt{\frac{T^3}{2\,\rho\,A_D}} + \frac{1}{8}\,\sigma_R\,\overline{C}_{d_0}\,\rho\,A_D(\Omega R)^3 \qquad \textit{eq 5.27}$$

$$P_{TOTAL} = P_i + P_0 + P_{TR} + P_{Acc} + P_{loss} \qquad \textit{eq 5.28}$$

$$C_T = \frac{T}{\rho_a\,A_D\,(\Omega R)^2} = f\left(\frac{T}{\sigma}, N_R\right) \qquad \textit{eq 5.29}$$

$$C_P = \frac{P}{\rho_a\,A_D\,(\Omega R)^3} = f\left(\frac{P}{\sigma}, N_R\right) \qquad \textit{eq 5.30}$$

$$C_T = \frac{1}{6}\,\sigma_R\,\overline{C}_1 \qquad \textit{eq 5.31}$$

$$C_P = \frac{1}{8}\,\sigma_R\left(\overline{C}_{d_0} + \overline{C}_{d_i}\right) \qquad \textit{eq 5.32}$$

$$C_T = f\left(W,\ H_D,\ \frac{1}{N_R}\right) \qquad \textit{eq 5.33}$$

$$C_P = f\left(RSHP,\ H_D,\ \frac{1}{N_R}\right) \qquad \textit{eq 5.34}$$

$$C_P = \sqrt{\frac{C_T^3}{2}} = \frac{1}{8}\,\sigma_R\,\overline{C}_{d_0} \qquad \textit{eq 5.35}$$

$$M' = \frac{T\,v_i}{P} = \frac{\text{Minimum Power Required}}{\text{Actual Power Required}} \qquad \textit{eq 5.36}$$

$$M' = \frac{T^{\frac{3}{2}}}{P}\sqrt{\frac{1}{2\,\rho\,A_D}} \qquad \textit{eq 5.37}$$

$$M' = \frac{\dfrac{C_T^{\frac{3}{2}}}{\sqrt{2}}}{\dfrac{C_T^{\frac{3}{2}}}{\sqrt{2}} + \dfrac{1}{8}\,\sigma_R\,\overline{C}_{d_0}} \qquad \textit{eq 5.38}$$

$$M' = \frac{\left(\dfrac{C_T^{\frac{3}{2}}}{\sqrt{2}}\right)}{C_P} \qquad \textit{eq 5.39}$$

$$C_P = C_1\,C_T^{\frac{3}{2}} + C_2 \qquad \textit{eq 5.40}$$

$$W_S = C_T\,\rho_S\,A_D\,(\Omega R_S)^2 \qquad \textit{eq 5.41}$$

$$RSHP_S = C_P\,\rho_S\,A_D\,(\Omega R_S)^3 \qquad \textit{eq 5.42}$$

$$W_S = W_T\,\frac{\rho_S}{\rho_T}\left(\frac{\Omega R_S}{\Omega R_T}\right)^2 \qquad \textit{eq 5.43}$$

$$ESHP_S = ESHP_T\,\frac{\rho_S}{\rho_T}\left(\frac{\Omega R_S}{\Omega R_T}\right)^3 \qquad \textit{eq 5.44}$$

$$W_{ref} = \frac{W_T}{\sigma_T}\left(\frac{N_{R_S}}{N_{R_T}}\right)^2 \qquad \textit{eq 5.45}$$

$$ESHP_{ref} = \frac{ESHP_T}{\sigma_T}\left(\frac{N_{R_S}}{N_{R_T}}\right)^3 \qquad \textit{eq 5.46}$$

$$H_{P_c} = H_{P_o} + \Delta H_{P_{ic}} + \Delta H_{pos}$$ *eq 5.47*

$$T_a = T_o + \Delta T_{ic}$$ *eq 5.48*

$$\sigma_T = \frac{\delta}{\theta} = \frac{\delta}{\left(\frac{T_a}{T_{ssl}}\right)}$$ *eq 5.49*

$$ESHP_T = K_Q\,(Q)\,(N_{R_T})$$ *eq 5.50*

$$GW = ESGW - FU$$ *eq 5.51*

$$GW = ESGW - FU + \text{Cable Tension}$$ *eq 5.52*

$$\Omega R = K_{GR}\left(N_{R_T}\right)\left(\frac{2\pi}{60}\right)(R)$$ *eq 5.53*

$$ESHP_{ref} = f(W_{ref})$$ *eq 5.54*

$$ESHP_{ref} = A_0 + A_1\, W_{ref}^{\frac{3}{2}}$$ *eq 5.55*

$$\delta = \frac{P_a}{P_{ssl}} = \left(1 - \frac{\lambda_{ssl}\; H_P}{T_{ssl}}\right)^{\frac{g_{ssl}}{g_c\;\lambda_{ssl}\;R}}$$ *eq 5.56*

$$\theta = \frac{T_a}{T_{ssl}} = \frac{OAT + 273.15}{288.15}$$ *eq 5.57*

$$\sigma = \frac{\rho}{\rho_{ssl}} = \frac{\delta}{\theta}$$ *eq 5.58*

$$ESHP = \frac{ESHP_{ref}\ (\sigma)}{\left(\frac{N_{R_S}}{N_{R_T}}\right)^3}$$ *eq 5.59*

$$W_{ref} = \frac{W_S}{\sigma}$$ *eq 5.60*

$$ESHP = ESHP_{ref}(\sigma)$$ *eq 5.61*

CHAPTER FIVE

HOVER PERFORMANCE

5.1 INTRODUCTION

This chapter deals with determining helicopter hover performance. Where Chapter 4 was concerned with establishing engine power available, this chapter will discuss determining airframe power required to hover. Engine power available and airframe power required to hover will be combined to determine aircraft hover performance. The theory of the hovering rotor is presented and considers an aircraft of fixed configurations only. Different test methods commonly in use are examined. Pilot test techniques, scope of test, and the limitations associated with each test are discussed.

5.2 PURPOSE OF TEST

The purpose of this test is to evaluate aircraft hover performance characteristics. Airframe power required to hover will be determined and combined with engine power available to establish aircraft hover performance.

5.3 THEORY

5.3.1 General

The objective of hover performance tests is to determine the power required to hover in ground effect (IGE) and out of ground effect (OGE). The theory of this test is based on the basic aerodynamic theory of the hovering rotor. The hovering rotor is reviewed; however, detailed mathematical derivations and lengthy written explanations are omitted. The intent of the following discussion is to explain how theory is used to support data collection, reduction, and extrapolation.

The total hover power required for a single main rotor helicopter can be measured directly at the power plant output driveshaft (engine shaft horsepower). The total power required is the sum of the main rotor power (P_{MR}), tail rotor power (P_{TR}), accessory power (P_{Acc}), and the transmission (or gear box) losses (P_{loss}) involved in transmitting this power. Torque and shaft rotational speed are normally used to determine power requirements. Torque and shaft speed can be measured at the engine output, main rotor, and tail rotor. The power plant output shaft speed and gear box ratios are used to determine shaft rotational speeds at the torque measurement points. Additional instrumentation may be provided to determine tail rotor gear box efficiencies; however, the losses in this area are generally small.

When considering power requirements of a rotor, it is necessary to have a general understanding of the actual rotor operation. The aerodynamics of a rotor are very complex. First order approximations are used generally for analysis and to establish methods of extrapolating rotor power requirements. The analysis need not be completely rigorous because the intent is to show trends, important parameters, and major effects. If secondary effects could not be neglected or if first order approximations could not be made, it would be very difficult to establish the generalizations which allow many performance flight tests to be practical. Profile drag, tip losses, blade interference, etc. are targets of the first order approximation approach. In many cases, the simplifying assumptions are restrictive and although applicable to one flight regime, may not be reasonable in another. This is not a serious problem if the restriction of the assumption is understood and a correction is predictable.

5.3.2 Main Rotor Power Required

5.3.2.1 GENERAL

The power required to drive the main rotor is the sum of induced power (P_i) and profile power (P_0). Induced power is power required to develop thrust and will be estimated by the momentum theory. Profile power is power required to rotate the rotor blades against the viscous action of the air and will be estimated by the blade element theory.

Prior to discussing the blade element and momentum theory, it is necessary to analyze the flow of air relative to the rotor blade. In a three-dimensional situation, such as a rotor of finite span, the lifting surface creates a downward acceleration of the air passing the surface. If the air is assumed to have a velocity equal and opposite to the blade rotational speed, the flow is deflected as it passes the blade, as shown in Figure 5.1. The downward velocity imparted to the air as it passes the lifting blade is called induced velocity (v_i) and the angle through which the stream is deflected at the blade is the induced angle (α_i). Notice that the air is initially deflected upstream of the blade and continues to turn downstream of the blade, ultimately to an angle equal to twice that at the blade. It is also useful to visualize the same phenomenon considering the blade moving relative to the stationary air, as shown in Figure 5.2.

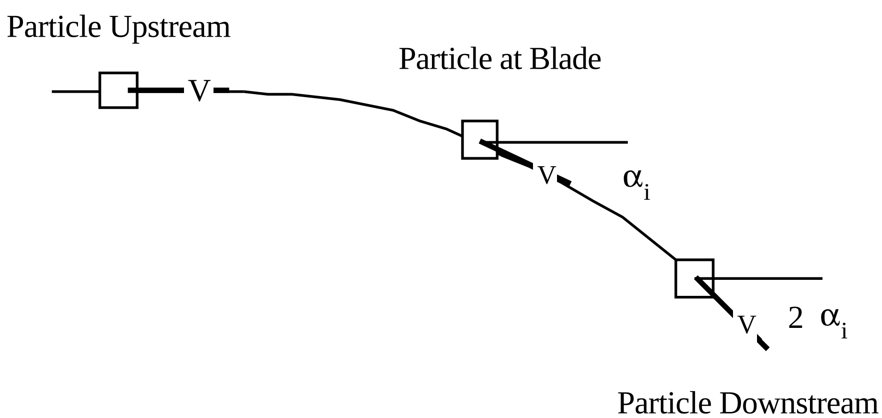


Figure 5.1
Motion of Air Particles Relative to Blade

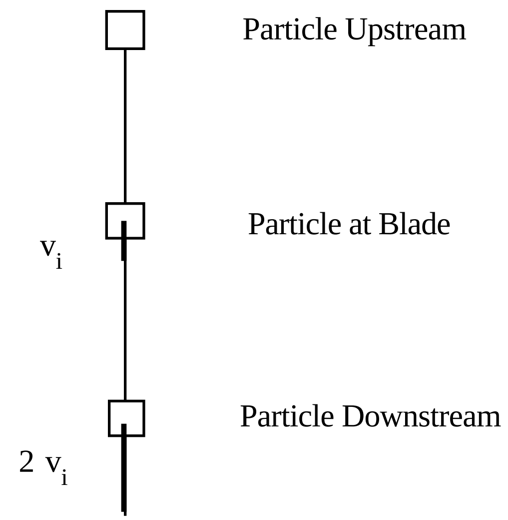


Figure 5.2
Motion of Air Particle Relative to Air Mass

5.3.2.2 BLADE ELEMENT ANALYSIS (PROFILE POWER)

To examine the effect of the induced velocity on the aerodynamic reactions at the blade, cut an element out of a hovering rotor at an arbitrary radius, r, from the center of rotation (Figure 5.3). The blade element resultant aerodynamic force (dR) acting on the blade element is composed of two components: the blade element lift, which is normal to the local resultant velocity through the rotor (V_R); and the blade element profile drag, which is parallel to the local velocity, or:

$$dR = dL + dD_0 \qquad \textit{eq 5.1}$$

Where:

dR	- Blade element resultant aerodynamic force
dL	- Blade element lift
dD_0	- Blade element profile drag.

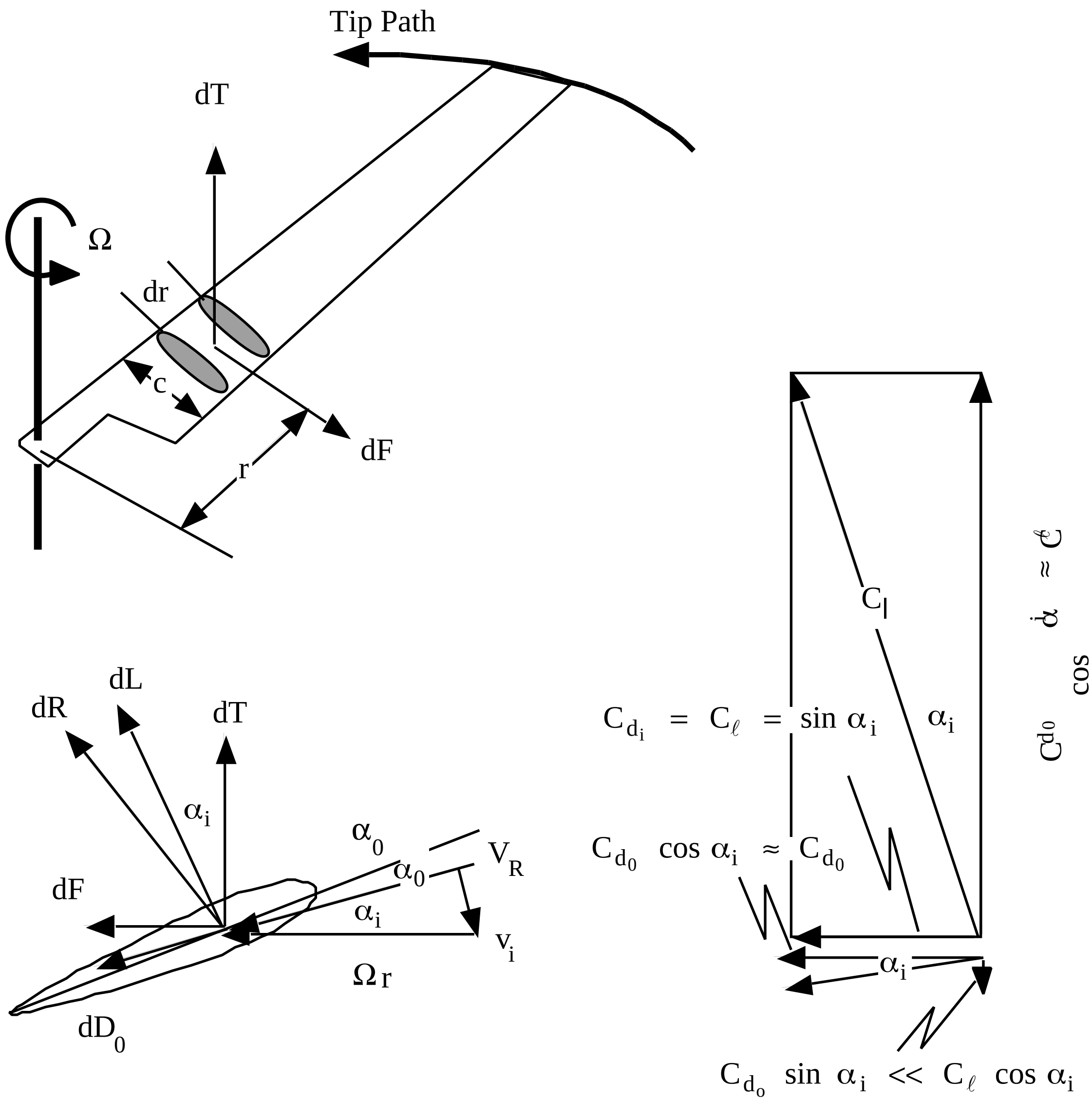


Figure 5.3
Rotor Blade Element in Hover

These components of the resultant force are interesting because they are predictable aerodynamically; however, the components of dR normal and parallel to the plane of rotation are more pertinent to performance analysis:

$$dR = dT + dF \qquad \textit{eq 5.2}$$

Where:

dR	- Blade element resultant aerodynamic force
dT	- Blade element thrust
dF	- Blade element torque force.

The component of the resultant aerodynamic force perpendicular to the plane of rotation, dT (thrust), is composed of a component of dL minus a component of dD_0 , and is the "useful" force being produced by the rotor. The other component of dR, parallel to the plane of rotation, is dF (torque force) and is composed of a component of dL (usually called induced drag) plus a component of dD_0 :

$$dT = dL\cos\alpha_i - dD_0 \sin\alpha_i \qquad \textit{eq 5.3}$$

And:

$$dF = dL \sin\alpha_i + dD_0 \cos\alpha_i \qquad \textit{eq 5.4}$$

Where:

dT	- Blade element thrust
dL	- Blade element lift
α_i	- Induced angle
dD_0	- Blade element profile drag
dF	- Blade element torque force.

The element lift and drag can be expressed as:

$$dL = C_\ell q_e \, dS \qquad \textit{eq 5.5}$$

And:

$$dD_0 = C_{d_0} q_e \, dS \qquad \textit{eq 5.6}$$

Where:

dL	- Blade element lift
C_ℓ	- Blade element lift coefficient

q_e	- Blade element dynamic pressure
dS	- Blade element area
dD_0	- Blade element profile drag
C_{d_0}	- Blade element profile drag coefficient.

First considering the thrust and substituting Equation 5.5 and Equation 5.6 into Equation 5.3:

$$dT = \frac{\rho(\Omega r)^2}{2\cos^2\alpha_i} c\,dr\,(C_\ell \cos\alpha_i - C_{d_0} \sin\alpha_i) \qquad \textit{eq 5.7}$$

Where:

dT	- Blade element thrust
ρ	- Density
Ωr	- Blade element rotational velocity
α_i	- Induced angle
c	- Blade chord
dr	- Blade element radius
$C\ell$	- Blade element lift coefficient
C_{d_0}	- Blade element profile drag coefficient.

Analysis of the induced velocity will show that $\Omega r >> v_i$; thus α_i is a small angle. Therefore, $\cos\alpha_i \cong 1.0$ and $\sin\alpha_i \cong \alpha_i$. Generally, $C\ell >> C_{d_0}$. Therefore, the second term in the differential thrust equation (Equation 5.7) can be neglected when compared to the first term:

$$dT = \frac{1}{2}\rho(\Omega r)^2 c\,dr\,C_\ell \qquad \textit{eq 5.8}$$

Where:

dT	- Blade element thrust
ρ	- Density
Ωr	- Blade element rotational velocity

c	- Blade chord
dr	- Blade element radius
C_ℓ	- Blade element lift coefficient.

The total thrust per blade can be determined by integration along the blade:

$$T_{perblade} = \int dT = \int \frac{1}{2} \rho (\Omega r)^2 \, c\,dr \, C_\ell \qquad \textit{eq 5.9}$$

Where:

T	- Thrust
dT	- Blade element thrust
ρ	- Density
Ωr	- Blade element rotational velocity
c	- Blade chord
dr	- Blade element radius
C_ℓ	- Blade element lift coefficient.

For a first approximation, the integration is simplified by specifying:

1. Density constant (incompressible).
2. Element chord constant (no taper).
3 The element lift coefficient replaced by the average value along the blade, ($\bar{C}_\ell$).

For “b” number of blades, the total thrust produced is:

$$T_b = \frac{1}{6} \sigma_R \bar{C}_\ell \rho A_D (\Omega R)^2 \qquad \textit{eq 5.10}$$

Where:

Tb	- Thrust of b blades
σ_R	- Rotor solidity ratio
$\bar{C}_\ell$	- Average blade element lift coefficient

ρ - Density

A_D - Rotor disc area

ΩR - Blade rotational velocity.

Analysis of the torque force is done in a similar manner:

$$dF = \frac{\rho\,(\Omega r)^2}{2\cos^2\alpha_i}\,c\,dr\,(C_\ell \sin\alpha_i + C_{d_0}\cos\alpha_i) \qquad \textit{eq 5.11}$$

Where:

dF - Blade element torque force

ρ - Density

Ωr - Blade element rotational velocity

α_i - Induced angle

c - Blade chord

dr - Blade element radius

$C\ell$ - Blade element lift coefficient

C_{d_0} - Blade element profile drag coefficient.

The small angle approximation is again applied but, in this case, it cannot be justified that either term be neglected (Figure 5.3). The equation can then be written as:

$$dF = \frac{1}{2}\rho\,(\Omega r)^2\,c\,dr\,(C_{d_i} + C_{d_0}) \qquad \textit{eq 5.12}$$

Where:

dF - Blade element torque force

ρ - Density

Ωr - Blade element rotational velocity

c - Blade chord

dr - Blade element radius

C_{d_i} - Blade element induced drag coefficient, $C_{d_i} = C\ell\ \alpha_i = C\ell \sin\alpha_i$

C_{d_0} - Blade element profile drag coefficient.

The power required, in ft-lb/s, to rotate the blade element about the shaft axis is:

$$dP = dF\,(r)\,\Omega \qquad \textit{eq 5.13}$$

Where:

dP	- Blade element power required
dF	- Blade element torque force
r	- Radius to blade element
Ω	- Rotor angular velocity.

The integration over b blades produces:

$$P_b = \frac{1}{8}\,\sigma_R\,(\overline{C}_{d_0} + \overline{C}_{d_i})\,\rho\,A_D\,(\Omega R)^3 \qquad \textit{eq 5.14}$$

Where:

P_b	- Power required for b blades
σ_R	- Rotor solidity ratio
$\overline{C}_{d_0}$	- Average blade element profile drag coefficient
$\overline{C}_{d_i}$	- Average blade element induced drag coefficient
ρ	- Density
A_D	- Rotor disc area
ΩR	- Blade rotational velocity.

To simplify the integration, C_{d_0} and C_{d_i} were taken to be constant along the blade. The blade element profile drag coefficient, C_{d_0}, varies little over the normal angle of attack and Mach number range of the operating rotor and, for these conditions, can be assumed constant (Figure 5.4). Either high angle of attack or high Mach number may cause large increases in C_{d_0} and, therefore, increases in power required. The blade element induced drag coefficient, C_{d_i}, varies considerably with changes of α or C_L. Thus, in general, the changes in rotor power required at constant N_R are due to changes in rotor induced power.

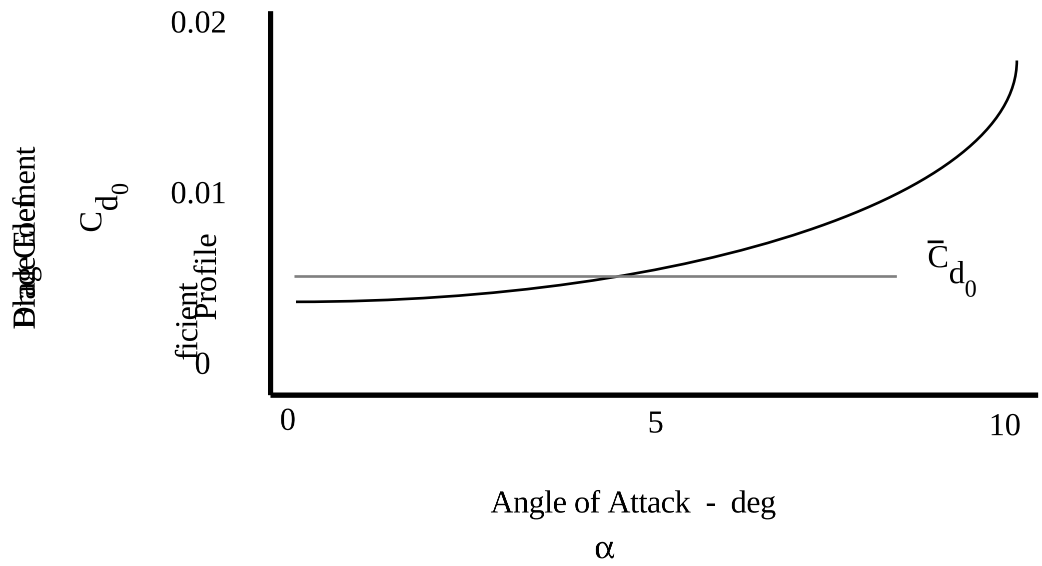


Figure 5.4
Variation of Section Profile Drag Coefficient with Angle of Attack (NACA 0012, M = 0.3, Ref: TN4357)

Analysis of power required is handled easily if the induced power term is predicted by the momentum theory. Blade element theory is used only to analyze the profile term. The above development shows the profile power to be:

$$P_0 = \frac{1}{8}\,\sigma_R\,\bar{C}_{d_0}\,\rho\,A_D\,(\Omega R)^3 \qquad \textit{eq 5.15}$$

Where:

P_0	- Profile power
σ_R	- Rotor solidity ratio
$\bar{C}_{d_0}$	- Average blade element profile drag coefficient
ρ	- Density
A_D	- Rotor disc area
ΩR	- Blade rotational velocity.

The induced power must be added to the above profile power to determine total main rotor power required.

5.3.2.3 MOMENTUM ANALYSIS (INDUCED POWER)

The momentum theory of rotor action visualizes the thrust created as a reaction to the force required to accelerate the air mass through an ideal actuator disc. The analysis affords a simple solution to the induced power required (power to overcome the induced drag of the rotor blades) by making many simplifying but restrictive assumptions. The results, then, are only approximations because of the many differences between the ideal actuator disc and the real rotor. However, many useful relationships are produced. Corrections can be applied to the results to account for the major discrepancies.

Listed below are the basic assumptions of the momentum theory, and parenthetically, the major corresponding consequences:

1. Inviscid, frictionless fluid (no profile drag).
2. Rotor acts as a disc with an infinite number of blades imparting a constant energy to the fluid (no periodicity of the wake).
3. Flow through the disc is uniform (optimum induced velocity and no tip losses).
4. Constant energy flow ahead of and behind the disc.

The static pressure in the flow acted on by the rotor changes in accordance with the Bernoulli equation and the disc sustains a pressure difference. However, the air far upstream of the hovering rotor is initially at zero velocity and at ambient pressure (P_a), and far downstream the ultimate wake must be at ambient pressure once again but moving with a velocity, v_w. Considering the flow to be incompressible, and examining the conditions far upstream and far downstream (Figure 5.5):

$$P_{T_u} = P_a = C_1 \qquad \textit{eq 5.16}$$

And:

$$P_{T_d} = P_a + q_w = C_2 \qquad \textit{eq 5.17}$$

Where:

P_{T_U}	- Upstream total pressure
P_a	- Ambient pressure
P_{T_D}	- Downstream total pressure
q_w	- Wake dynamic pressure
C_1, C_2	- Constant.

So that:

$$\Delta P = P_{T_d} - P_{T_u} = q_w$$

eq 5.18

Where:

ΔP	- Pressure change
P_{T_U}	- Upstream total pressure
P_{T_D}	- Downstream total pressure
q_w	- Wake dynamic pressure.

This pressure difference sustained by the actuator disc is the force produced per unit disc area, or:

$$\frac{T}{A_D} = \Delta P = \frac{1}{2} \rho v_w^2$$

eq 5.19

Where:

T	- Thrust
A_D	- Rotor disc area
ΔP	- Pressure change
ρ	- Density
v_w	- Ultimate wake velocity.

The above equation shows the dependence of the wake dynamic pressure (q_w) on the disc loading (T / A_D) and, as an approximation, describes the wake velocity (v_w) imposed on the surroundings under the lifting disc.

For steady flow, Newton's law states:

$$F = \frac{dm}{dt} \Delta V \qquad \textit{eq 5.20}$$

Where:

F	- Force
m	- Mass
ΔV	- Change in velocity
t	- Time.

The thrust produced by the rotor is the product of the mass flow rate through the rotor and the change in velocity of that mass. When describing the flow rate at the disc, Newton's law can be written:

$$(\Delta P)\ A_D = \rho\ A_D\ (v_w)\ v_i \qquad \textit{eq 5.21}$$

Where:

ΔP	- Pressure change
A_D	- Rotor disc area
ρ	- Density
v_w	- Ultimate wake velocity
v_i	- Induced velocity at hover.

Therefore:

$$v_w = 2v_i$$

From the above, it is seen that the pressure and velocity along the stream vary, as shown in Figure 5.5. The variations of stream tube size is given by the continuity equation, (AREA) (VELOCITY) = CONSTANT, for incompressible flow.

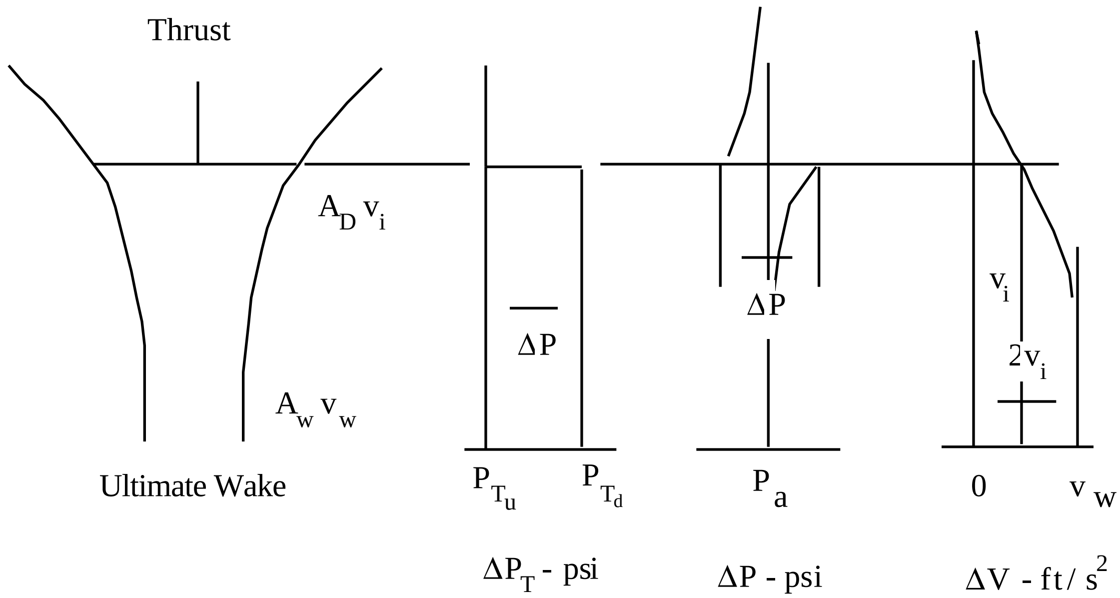


Figure 5.5
Actuator Disc in Hover

Using the relationship between wake velocity and induced velocity and substituting in Equation 5.21:

$$T = \frac{dm}{dt} \Delta D = 2 \rho A_D v_i^2 \qquad \textit{eq 5.22}$$

Where:

T	- Thrust
m	- Mass
ΔV	- Change in velocity
ρ	- Density
A_D	- Rotor disc area
v_i	- Induced velocity at hover
t	- Time.

Or, because of interest in the magnitude of the induced effects and assuming $T \cong W$ in hover:

$$v_i = \sqrt{\frac{T}{2\,\rho\,A_D}} = \sqrt{\frac{W}{2\,\rho\,A_D}} \qquad \textit{eq 5.23}$$

Where:

v_i	- Induced velocity at hover
T	- Thrust
ρ	- Density
A_D	- Rotor disc area
W	- Weight.

The power required to accelerate the air mass through the disc, induced power, assuming $T \cong W$ at a hover, is:

$$P_i = Tv_i = \sqrt{\frac{T^3}{2\,\rho\,A_D}} = \sqrt{\frac{W^3}{2\,\rho\,A_D}} \qquad \textit{eq 5.24}$$

Where:

P_i	- Induced power
v_i	- Induced velocity at hover
T	- Thrust
ρ	- Density
A_D	- Rotor disc area
W	- Weight.

It should be noted that the induced velocity, v_i, was assumed to be constant across the disc which is an optimum situation and, thus, the induced power indicated by this analysis is a minimum.

5.3.3 Total Power Required

The major portion of the total power being produced by the engine (ESHP) is absorbed by the main rotor shaft. The main rotor power (RSHP) is composed of the induced power and the profile power. The ratio of RSHP to ESHP of a single main rotor helicopter is defined as mechanical efficiency, η_m:

$$\eta_m = \frac{RSHP}{ESHP} \qquad \textit{eq 5.25}$$

Where:

η_m - Mechanical efficiency

RSHP - Rotor shaft horsepower

ESHP - Engine shaft horsepower.

η_m is typically about 0.85 in hover. The mechanical efficiency is a measure of the engine power required to overcome various mechanical losses and includes the power required to drive the tail rotor, which has its own components of induced and profile power. An analysis of the tail rotor power required is not presented in this manual. Remember the purpose of the tail rotor is to balance the torque reaction to the rotation of the main rotor. Thus, the tail rotor power requirement, in hover, is dependent on the power input to the main rotor. ESHP is typically divided 85% to RSHP and 15% to miscellaneous uses. Of the 85% that goes RSHP about 25% goes to P_0 and about 60% to P_i. The 15% miscellaneous power is split between P_{loss}, P_{Acc}, and P_{TR}.

P_{Acc} is the engine power supplied to pumps, generators, cooling fans, etc., necessary to run the many auxiliary systems in the aircraft and may vary widely depending on the loads imposed on the accessory systems. The transmission losses are a result of friction in the drive train and are primarily a function of N_R, thus remaining about constant for the helicopter. The tail rotor power is included in the miscellaneous 15% but is really not a constant value. Since it is a rotor and subject to the same power effects as the main rotor, the tail rotor power required varies as a function of rotor speed and density conditions.

The main rotor power required to hover, P_{MR}, can be written:

$$P_{MR} = P_i + P_0 \qquad \textit{eq 5.26}$$

Or:

$$P_{MR} = \sqrt{\frac{T^3}{2\,\rho\,A_D}} + \frac{1}{8}\,\sigma_R\,\bar{C}_{d_0}\,\rho\,A_D(\Omega R)^3 \qquad \textit{eq 5.27}$$

Where:

P_{MR} - Main rotor power required

P_i - Induced power

P_0 - Profile power

T - Thrust

σ_R - Rotor solidity ratio

$\bar{C}_{d_0}$ - Average blade element profile drag coefficient

ρ - Density

A_D - Rotor disc area

ΩR - Blade rotational velocity.

The total power required to hover, P_{TOTAL}, can be written:

$$P_{TOTAL} = P_i + P_0 + P_{TR} + P_{Acc} + P_{loss} \qquad \textit{eq 5.28}$$

Where:

P_{TOTAL} - Total power required to hover

P_i - Induced power

P_0 - Profile power

P_{TR} - Tail rotor power required

P_{Acc} - Accessory power required

P_{loss} - Power loss.

5.3.4 Nondimensional Coefficients

Examination of the main rotor power required equation reveals that the induced power depends on gross weight and density altitude, and the profile power depends on these two variables (through the average profile drag coefficient) and rotor speed. The effect of these variables on hover power can be determined and the results expressed as nondimensional power and weight coefficient.

$$C_T = \frac{T}{\rho_a \, A_D \, (\Omega R)^2} = f\left(\frac{T}{\sigma}, N_R\right) \qquad \textit{eq 5.29}$$

And:

$$C_P = \frac{P}{\rho_a \, A_D \, (\Omega R)^3} = f\left(\frac{P}{\sigma}, N_R\right) \qquad \textit{eq 5.30}$$

Where:

C_T	- Thrust coefficient
C_P	- Power coefficient
P	- Power
T	- Thrust
ρ_a	- Ambient air density
A_D	- Rotor disc area
ΩR	- Blade rotational velocity
σ	- Density ratio
N_R	- Main rotor speed.

Substituting Equation 5.10 for T in Equation 5.29 and substituting Equation 5.14 for P in Equation 5.30 gives:

$$C_T = \frac{1}{6} \sigma_R \overline{C}_l \qquad \textit{eq 5.31}$$

And:

$$C_P = \frac{1}{8}\sigma_R\left(\overline{C}_{d_0} + \overline{C}_{d_i}\right) \qquad \textit{eq 5.32}$$

Where:

C_T	- Thrust coefficient
C_P	- Power coefficient
σ_R	- Rotor solidity ratio
$\overline{C}\ell$	- Average blade element lift coefficient
$\overline{C}_{d_0}$	- Average blade element profile drag coefficient
$\overline{C}_{d_i}$	- Average blade element induced drag coefficient.

In the case of the hovering helicopter, where $T \cong W$, the thrust and power coefficients are variable only with gross weight, rotor speed, and density altitude:

$$C_T = f\left(W,\ H_D,\ \frac{1}{N_R}\right) \qquad \textit{eq 5.33}$$

And:

$$C_P = f\left(RSHP,\ H_D,\ \frac{1}{N_R}\right) \qquad \textit{eq 5.34}$$

Where:

C_T	- Thrust coefficient
C_P	- Power coefficient
W	- Weight (helicopter)
RSHP	- Rotor shaft horsepower
H_D	- Density altitude
N_R	- Main rotor speed.

The total main rotor power required to hover can now be expressed in terms of the nondimensional coefficients:

$$C_P = \sqrt{\frac{C_T^3}{2}} = \frac{1}{8}\sigma_R \, \overline{C}_{d_0} \qquad \textit{eq 5.35}$$

Where:

C_T - Power coefficient

C_P - Thrust coefficient

σ_R - Rotor solidity ratio

$\overline{C}_{d_0}$ - Average blade element profile drag coefficient.

5.3.5 Figure of Merit

The efficiency of a lifting rotor is determined by comparing the actual power required to produce a given amount of thrust with the minimum power required to produce that thrust. The minimum amount of power required to produce a given amount of thrust is obtained with an ideal rotor where the induced power is minimum (uniform flow), zero profile-drag exists, and there are no rotational or tip losses. Therefore, for an ideal rotor, the rotor efficiency would be unity and the figure of merit (M′) is 1.

In estimating the effectiveness of a lifting rotor at a hover, the relationship for figure of merit becomes:

$$M' = \frac{T\,v_i}{P} = \frac{\text{Minimum Power Required}}{\text{Actual Power Required}} \qquad \textit{eq 5.36}$$

Where:

M' - Figure of merit

T - Thrust (= weight)

v_i - Induced velocity at hover (Ideal)

P - Power required to hover.

Substituting Equation 5.23 for v_i:

$$M' = \frac{T^{\frac{3}{2}}}{P}\sqrt{\frac{1}{2\,\rho\,A_D}} \qquad \textit{eq 5.37}$$

Where:

M'	- Figure of merit
T	- Thrust
P	- Power require to hover
ρ	- Density
A_D	- Rotor disc area.

Thus, as the figure of merit approaches unity, the thrust per unit shaft horsepower is increased.

Uniform flow is associated with an ideal rotor, hence the figure of merit is 1. However, the figure of merit of a rotor having nonuniform flow and some profile drag is not a unique number but varies as a function of the thrust coefficient.

The ideal rotor power required to hover will always be less than actual power required for two reasons. First, profile power requirements are nonexistent for the ideal rotor and second, momentum analysis under-predicts the induced power requirements because of losses incurred by the actual rotor. The figure of merit expression, in coefficient form, of a rotor with profile power requirements and without induced power losses (uniform flow) is as follows:

$$M' = \frac{\dfrac{C_T^{\frac{3}{2}}}{\sqrt{2}}}{\dfrac{C_T^{\frac{3}{2}}}{\sqrt{2}} + \dfrac{1}{8}\sigma_R\,\overline{C}_{d_0}} \qquad \textit{eq 5.38}$$

Or:

$$M' = \frac{\left(\frac{C_T^{\frac{3}{2}}}{\sqrt{2}}\right)}{C_P} \qquad \textit{eq 5.39}$$

Where:

M'	- Figure of merit
C_T	- Thrust coefficient
C_P	- Power coefficient
σ_R	- Rotor solidity ratio
$\overline{C}_{d_0}$	- Average blade element profile drag coefficient.

Thus, for a particular value of rotor solidity and profile drag coefficient, the figure of merit increases with increasing thrust coefficient. As C_T increases and induced power increases, the profile power becomes a relatively smaller contribution to the efficiency of the rotor (figure of merit). However, when the profile drag coefficient is no longer constant but increasing, a further increase in thrust coefficient results in a reduction of figure of merit (Figure 5.6).

Since ground effect produces a significant change in the power required to hover, the same change is reflected in the figure of merit. The power required to produce a given amount of thrust (C_T) decreases with increasing ground effect (higher figure of merit). Figure 5.6 illustrates the effects of profile drag coefficient and IGE and OGE for an actual hovering rotor on the figure of merit.

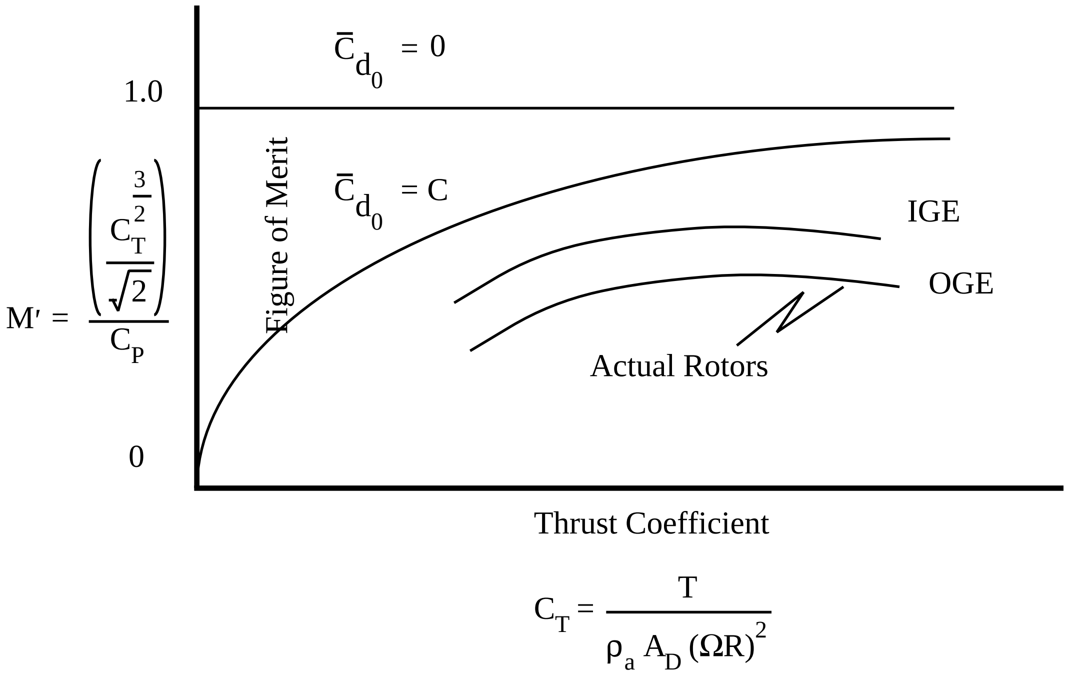


Figure 5.6
Figure of Merit

Figure 5.7 graphically presents a performance envelope for the hovering helicopter (T = W) in terms of disc loading (T/A_D) and power loading (T/P). Generally, a figure of merit of 0.75 is considered typical of a good rotor, while a value of 0.50 is representative of a poor rotor.

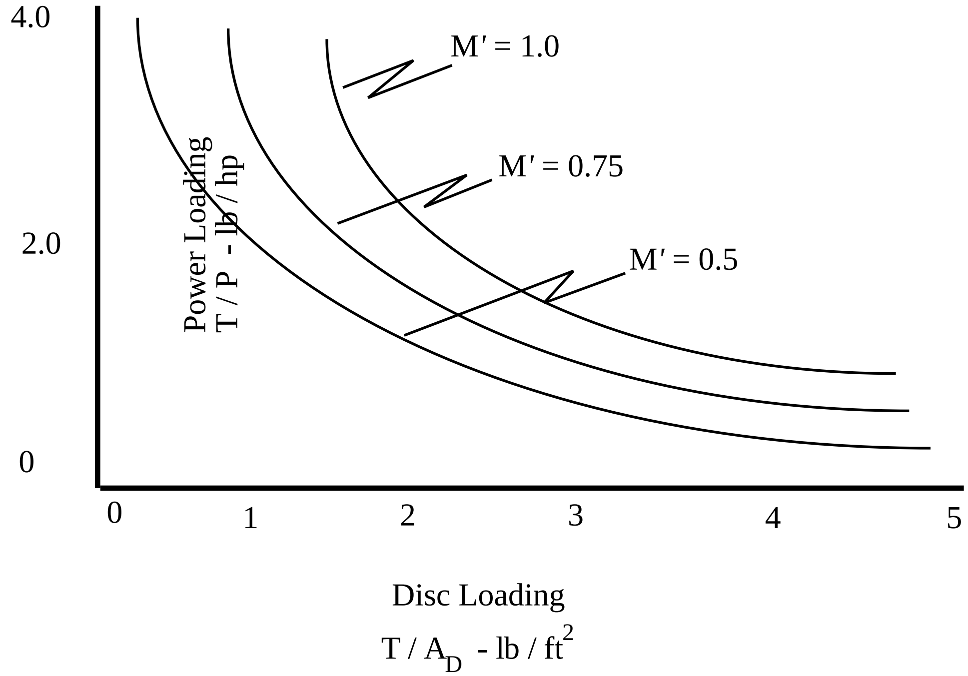


Figure 5.7
Rotor Power Loading Versus Disc Loading

5.3.6 Generalized Hover Performance

Using the nondimensional relationships for main rotor power required and figure of merit, a method of generalizing hover performance data can be established:

$$C_P = C_1\, C_T^{\frac{3}{2}} + C_2 \qquad \textit{eq 5.40}$$

Where:

$C_1 = 1/\sqrt{2}$

$C_2 = \frac{1}{8}\,\sigma_R\,\overline{C}_{d_0}$

And:

$$M' = \left(\left(C_T^{\frac{3}{2}}\right)/\sqrt{2}\right)/C_P\,.$$

The figure of merit relationship implies that at a constant figure of merit, $C_T^{\frac{3}{2}}$ will vary linearly with C_P. The main rotor power coefficient relationship further implies that $C_T^{\frac{3}{2}}$ will vary linearly with C_P as long as $\overline{C}_{d_0}$ remains constant. This assumption $\overline{C}_{d_0}$ = Constant is made to generalize hover performance data.

Linearizing the data is relatively accurate until $\overline{C}_{d_0}$ begins to vary significantly. In the high C_T range, the faired line generalization can no longer be used and caution must be exercised when extrapolating data for values of C_T not obtained during flight test. Since $T \cong W$ in a hover, generalized hover performance is expressed either in terms of gross weight and horsepower or nondimensional terms.

5.3.7 Referred Hover Performance

The magnitudes of C_T and C_P are typically C_T = 0.005 and C_P = 0.0003 and are not very meaningful to the pilot. As a result, hover performance data is often presented in terms of a referred system.

When a given value of RSHP (C_P) is determined by actual measurements during a hover test, the information represents a data point corresponding to a given weight or C_T condition. The test is repeated for the desired C_T range.

The values of density (ρ) and tip speed (ΩR) obtained during the above tests do not generally correspond to a set of standards for which the data is desired. To obtain the desired performance data, numerous values of C_T and C_P are multiplied by the standard values of (ΩR) and (ρ). The value selected for ρ is the standard value and the value for (ΩR) is the tip speed corresponding to the rotor speed stated in the detail specification or the operator's manual for hover performance. When C_T (Equation 5.29) and C_P (Equation 5.30) are multiplied by the nondimensionalizing terms (standard values), the results are weight (standard) and rotor shaft horsepower (standard).

$$W_S = C_T \rho_S A_D (\Omega R_S)^2 \qquad \textit{eq 5.41}$$

And:

$$RSHP_S = C_P \rho_S A_D (\Omega R_S)^3 \qquad \textit{eq 5.42}$$

Where:

WS	- Standard weight
$RSHP_S$	- Standard rotor shaft horsepower
C_T	- Thrust coefficient
C_P	- Power coefficient
ρ_S	- Standard density
A_D	- Rotor disc area
ΩR_S	- Standard blade rotational velocity.

Substituting the nondimensional relationships for C_T and C_P:

$$W_S = W_T \frac{\rho_S}{\rho_T}\left(\frac{\Omega R_S}{\Omega R_T}\right)^2 \qquad \textit{eq 5.43}$$

And:

$$ESHP_S = ESHP_T \frac{\rho_S}{\rho_T}\left(\frac{\Omega R_S}{\Omega R_T}\right)^3 \qquad \textit{eq 5.44}$$

Where:

W_S	- Standard weight
$ESHP_S$	- Standard engine shaft horsepower
W_T	- Test weight
$ESHP_T$	- Test engine shaft horsepower
ρ_S	- Standard density
ρ_T	- Test density
ΩR_S	- Standard blade rotational velocity
ΩR_T	- Test blade rotational velocity.

The above equations are used to determine standard gross weight and standard rotor shaft horsepower. If the standards chosen for the desired performance data are sea level, standard day conditions, the above equations reduce to the following which is known as the referred system.

$$W_{ref} = \frac{W_T}{\sigma_T}\left(\frac{N_{R_S}}{N_{R_T}}\right)^2 \qquad \textit{eq 5.45}$$

And:

$$ESHP_{ref} = \frac{ESHP_T}{\sigma_T}\left(\frac{N_{R_S}}{N_{R_T}}\right)^3 \qquad \textit{eq 5.46}$$

Where:

W_{ref}	- Referred weight
$ESHP_{ref}$	- Referred engine shaft horsepower
W_T	- Test weight
$ESHP_T$	- Test engine shaft horsepower
σ_R	- Test density ratio
N_{R_S}	- Standard main rotor speed
N_{R_T}	- Test main rotor speed.

The referred system eliminates the need to determine C_T and C_P.

Referred power may be either engine or rotor shaft horsepower. Engine shaft horsepower is the total power required to hover and is the sum of main and tail rotor power and transmission losses. The relationship between RSHP and ESHP is called the mechanical efficiency and often includes the tail rotor power required. This approach is not entirely accurate as tail rotor power varies with density, just as the main rotor power varies. However, this assumption allows suitable extrapolated hover performance using referred engine shaft horsepower ($ESHP_{ref}$) and referred gross weight (W_{ref}).

5.3.8 Compressibility Effects (Mach Effects)

Compressibility is an aerodynamic phenomenon that causes an abrupt and large increase in drag as the velocity of an airfoil approaches the speed of sound. In a helicopter, compressibility effects occur at the tip of the advancing blade where the highest Mach numbers occur. Compressibility is generally associated with high speed forward flight but may be noticeable with combinations of high altitude, cold temperature, and high values of C_T. The influence of compressibility effects on performance is expressed in terms of increasing power required at a constant C_T (gross weight) and can be as high as 15% in extreme cases.

The determination of hover ceilings is generally accomplished by extrapolating sea level hover performance to the desired altitude. This method generally produces relatively accurate results which are, however, occasionally optimistic. This discussion is intended only to acquaint the reader with compressibility and the limitations associated with extrapolated hover performance.

5.3.9 Ground Effect

5.3.9.1 GENERAL

Ground effect for a hovering helicopter can be defined as the change in power required to hover as the distance between the rotor disk and the ground decreases. There may or may not be an in-ground effect (IGE) hover guarantee to evaluate but there should be a requirement to determine IGE hover performance. Three specific gross weights should be investigated as time permits and in the following order: normal mission takeoff gross weight, maximum overload gross weight, and minimum mission landing weight. When determining a hover height to evaluate hover performance, it is important to determine the mission heights that are planned for operational use and the detailed specification requirement height.

5.3.9.2 FLOW PATTERN EFFECTS

As the distance between the hovering rotor and the surface below it is decreased to a distance that is approximately less than twice the rotor diameter, the flow pattern through the rotor changes. Figure 5.8 illustrates the difference in flow patterns IGE and OGE.

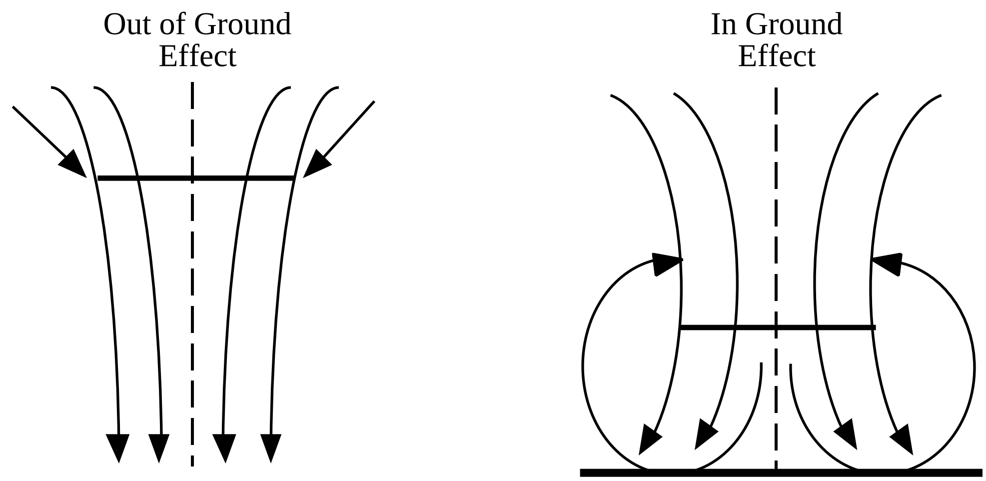


Figure 5.8
Flow Patterns In and Out-Of-Ground Effect

5.3.9.3 PRESSURE FIELD EFFECTS

The changes in performance near the ground that result from the pressure field can be favorable or unfavorable depending on the type of vehicle. As the rotor wake impinges on the surface, a region of greater than ambient pressure is created causing the deflection of the wake. The high pressure region is maintained by the centrifugal force of the air particles on the curved path as shown in Figure 5.9.

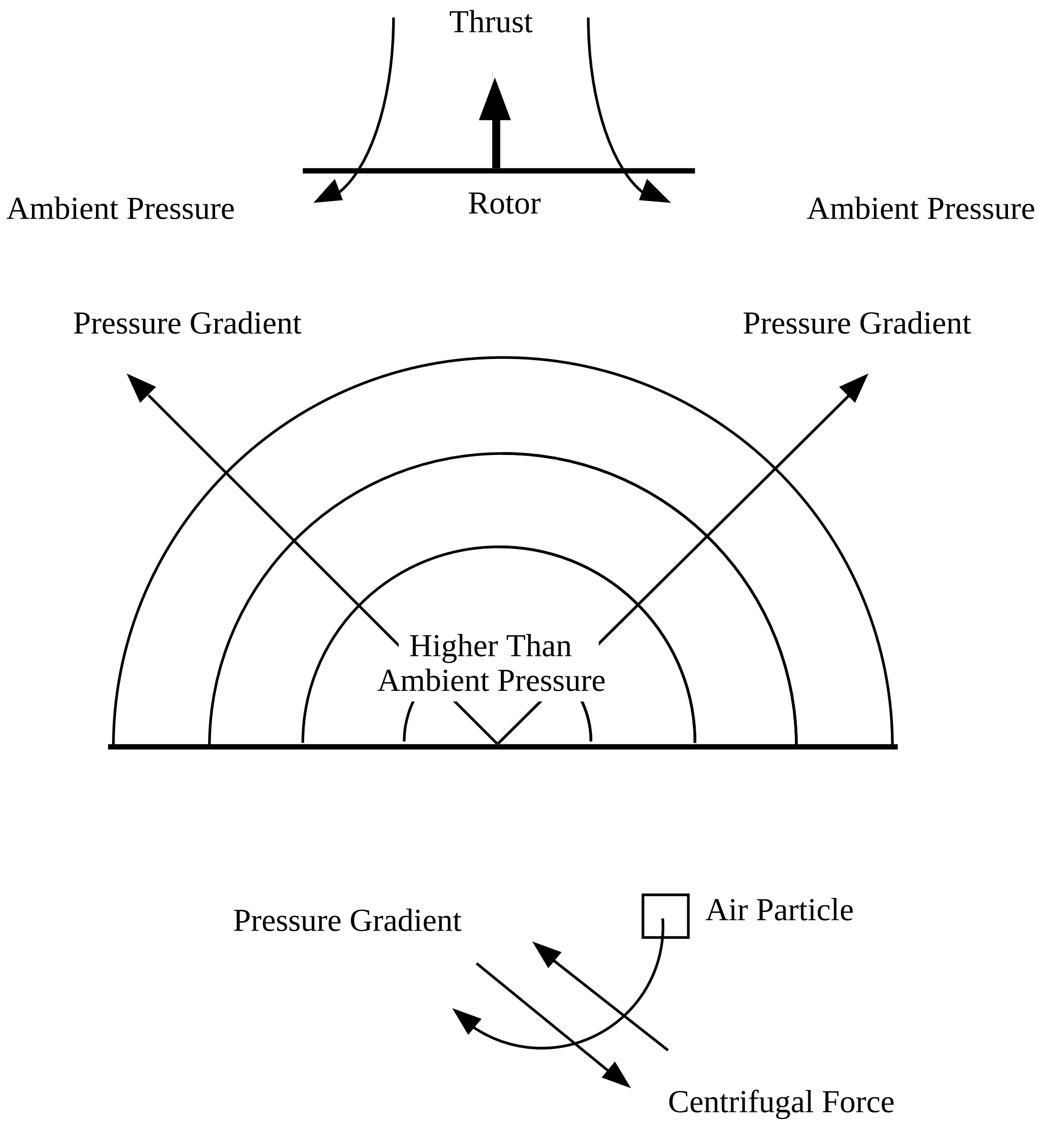


Figure 5.9
Ground Effect - Pressure Field

For momentum analysis, the thrust produced by an actuator disc is $T = A_D\ (\Delta P)$. As the rotor approaches the ground, the pressure field below the disc provides an additional DP on the disc which produces more thrust at a constant power, or reduces the power required at a constant thrust.

The effect becomes more pronounced as the rate of flow increases, that is, as the ratio of height to rotor diameter decreases. An approximate ratio of the power required to produce a certain thrust for a helicopter IGE to OGE and the effect on hover performance is shown in Figure 5.10. In rigorous analysis, this power correction involves only induced effects which influence the size of the correction and the height above which no correction is necessary (OGE).

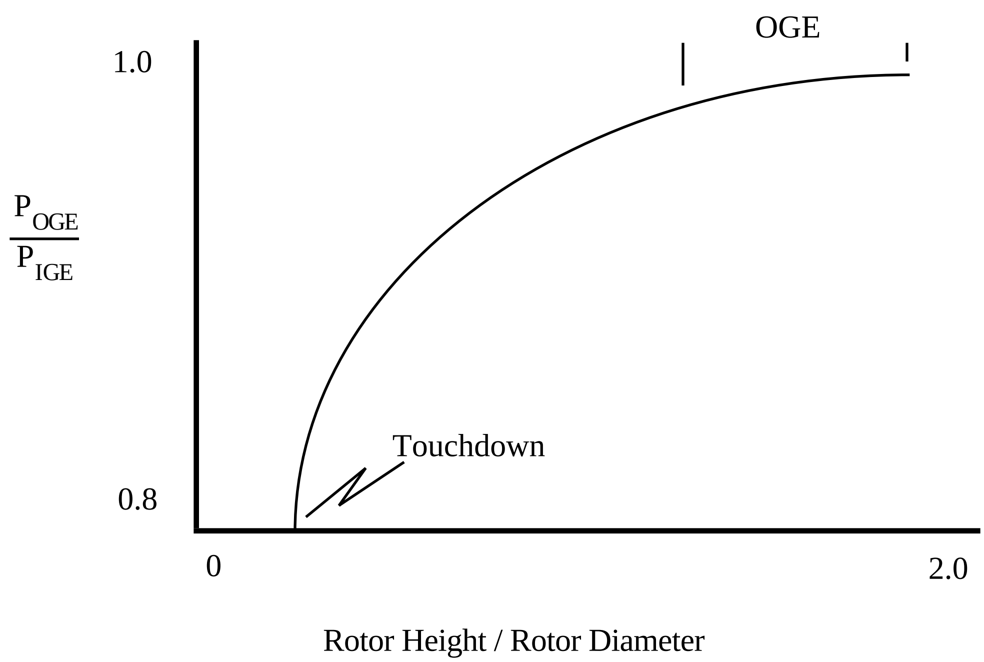


Figure 5.10
Ground Effect - Change in Power

5.4 TEST METHODS AND TECHNIQUES

5.4.1 General

There are two basic tests which provide hover performance data, free flight hover method and tethered hover method. These two techniques may be modified in several ways. The basic objective of all hover performance testing is the acquisition of power required data for the C_T range that describes the operating envelope of the test helicopter. The tests are based on the variation of at least one parameter found in the coefficient of thrust equation.

$$C_T = \frac{T}{\rho_a \, A_D \, (\Omega R)^2} = f\left(\frac{T}{\sigma}, N_R\right) \qquad \textit{eq 5.29}$$

It may be necessary to vary more than one parameter in an effort to extend the range of C_T obtained. The selection of a parameter to be varied and the significance of the parameter to the test results are very important.

5.4.1.1 WEIGHT

Since $T \cong W$ at a hover, varying gross weight is a relatively simple method of varying C_T. As weight is increased there is a proportionate increase in C_T. The weight of an aircraft will decrease normally by virtue of the fuel consumed, but this is a slow process. Weight may be changed in increments over a larger range by an incremental change in ballast or by varying tension on a tether cable (or both). Incremental ballasting by itself is not very practical because of manpower and time considerations but the tethered hover is used quite advantageously. Ballasting and tethered hover may require aircraft modification.

5.4.1.2 ALTITUDE

The variation of altitude to obtain variations of C_T at first appears to be an excellent way to determine altitude hover performance. This is not necessarily true for ground referenced tests as the cost of building, maintaining, and supplying high altitude sites, along with marginal wind and weather conditions found there, tends to prohibit large scale testing. Altitude testing at one high altitude site is conducted normally to determine service suitability and check hover performance data extrapolated from sea level tests.

Testing at the altitude of interest is important and is the only way to determine conclusively the high altitude operating characteristics of a rotor system. Mach effects found at altitude may introduce a significant increase in profile drag. OGE hover tests at altitude can be conducted using air referenced free flight methods.

5.4.1.3 ROTOR SPEED

Helicopters equipped with free turbine or reciprocating engines generally possess the ability to operate at various rotor speeds. This ability to vary rotor speed provides an extremely useful and rapid method of obtaining a number of different values of C_T. The big advantage of using rotor speed as a variable parameter is the ability to extend the C_T test range above and below those obtainable by other means. For example, if at sea level the maximum take-off gross weight does not provide a C_T corresponding to the anticipated hover ceiling, then the only rapid method to obtain a high C_T at the test site is a reduction in rotor speed (Ω). In many cases a sufficient amount of data can be obtained by using only two basic gross weights and four or five different rotor speeds.

5.4.2 Power Train Oscillations

Few helicopters will hover for an extended period of time without vertical or horizontal displacements; therefore, the power required to hover is an average value. Even if the helicopter is restrained by a tether, the aircraft will probably not become static. In fact, a helicopter may tend to bounce on a tether cable. The combination of the rotor system and free turbine engine constitutes a power train arrangement which allows rotor disturbances to produce lightly dampened engine/rotor speed oscillations. These oscillations may in turn provide a vertical aircraft oscillation which is seen as a continuous variation in hover height.

Data can be recorded in two ways to compensate for the dynamic oscillation just discussed. First, when the pilot is satisfied he has attained the most stable hover possible, he can momentarily fix the flight controls to prevent exciting the power train oscillation. This procedure will provide the minimum obtainable engine and rotor speed fluctuations but the aircraft may continue to oscillate in hover height. Again, the vertical oscillation will vary in height from a few inches to a foot depending on the test aircraft and its gross weight. This technique is used for the tethered hover method and the force recorded by the load cell varies instead of the hover height. The second pilot technique applied to hover performance tests requires the pilot to adjust the collective to maintain an exact hover height. This technique tends to excite oscillations in the power train but provides the minimum deviation in hover height.

The oscillations in fuel flow, torque, rotor speed, load cell force, hover height, etc., resulting from the lightly dampened oscillations of the power train must be recorded and averaged over a period of time. This reduces scatter to a point where the data is useful. The tethered hover method requires the data to be recorded and averaged for 30 seconds to obtain one data point. The free flight method usually allows relatively scatter-free data when one cycle of the power train oscillation is averaged (3 to 5 s).

5.4.3 Free Flight Hover

5.4.3.1 GENERAL

The free flight hover method is the oldest and simplest method of hover performance testing. The free flight hover method is used to check specific performance guarantees and to insure the tethered hover tests did not cause a performance loss. Free flight is the primary means when tethered hover is not possible.

Free flight tests can be performed with reference to the ground or air mass. The ground referenced tests are performed by establishing a hover height referenced to a weighted line suspended below the aircraft (Figure 5.11), by reference to a radar altimeter, or by reference to a hover height measuring device (HHMD). The air mass referenced tests are performed by reference to a weighted, tufted line; a hover meter; or other low airspeed measuring equipment. The power required to hover is recorded for various combinations of gross weight, density, and rotor operating conditions.

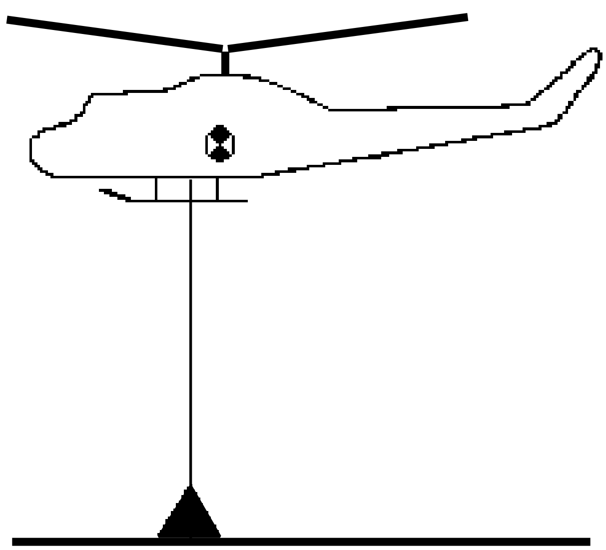


Figure 5.11
Free Flight Hover Method with Weighted Line

5.4.3.2 GROUND REFERENCED

The object of the ground referenced test is to determine the capability of the helicopter to hover at various thrust coefficient (C_T) values. If the test is to be completed at one test site, there are two ways to obtain various C_T values. One method is to start the test at the maximum gross weight permissible and reduce the ballast load after each data point. Ballasting is not recommended if time and money are considerations or unless one such flight is conducted for comparison against other methods. A second free flight hover method allows the test to be conducted at one, two, or three gross weights depending on the thrust range of the aircraft. The aircraft is stabilized at the desired height above the ground and power required data are recorded for three or four rotor speeds (N_R) within the prescribed variation. The altitude is varied to the next higher increment and the rotor speed sweep is repeated. This process is continued through the IGE range up to the OGE range. Normally, the hover height is varied incrementally up to twice the rotor diameter. Tests begin at low IGE altitudes and progress to OGE altitudes. Similarly, rotor speed variations begin at normal operating N_R and progress incrementally to the maximum and minimum N_R.

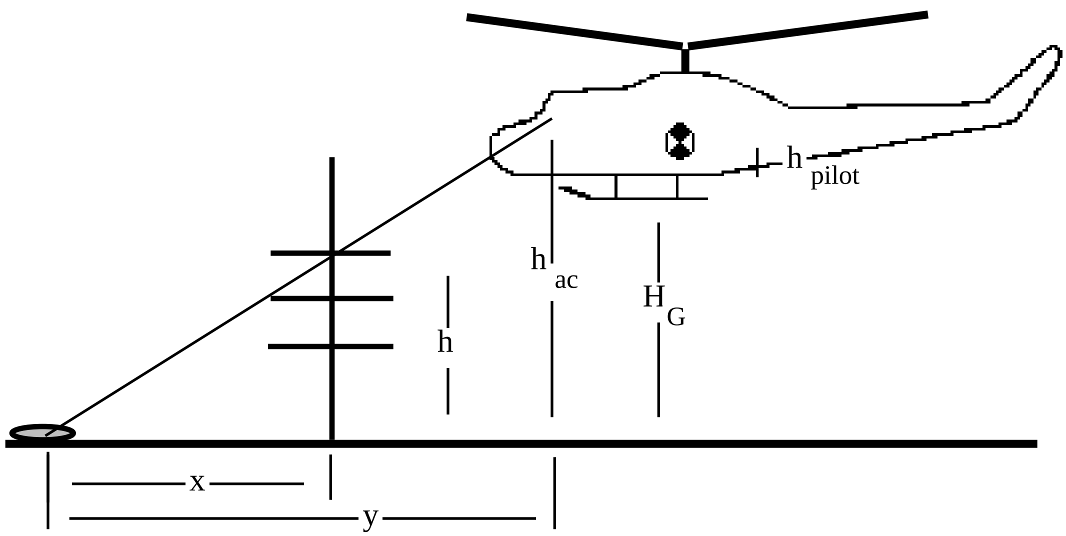
The limitations inherent in free flight hover arise from the basic simplicity of the test equipment, a rope and weight, radar altitude, or HHMD. The weighted line is complemented by an outside observer who indicates to the pilots by radio or hand signals errors in hover heights. It is evident that the lag in transfer of height information from the observer to the pilot produces less than optimum results. Station keeping (horizontal displacement) is difficult with all of the free flight methods.

In an OGE hover, the most important element of a stabilized hover is zero vertical/horizontal motion. The test height can be several feet in error and still not produce any error in the test results. As test hover height is decreased into IGE conditions, vertical/horizontal motion is easier to recognize and does not present a problem acquiring accurate data. The exact hover height is very important when obtaining IGE data as the power required to hover changes rapidly with variation in hover height.

5.4.3.3 HOVER HEIGHT MEASURING DEVICE (HHMD)

A Hover Height Measuring Device (HHMD) developed by USNTPS is illustrated in Figure 5.12. This device provides the pilot with a vertical reference for hover height.

The hover height is calculated using simple geometry. Horizontal reference is provided by positioning the helicopter over prominent ground markings such as a runway. This method allows the helicopter to be stabilized within feet horizontally and within inches of the desired hover height. The HHMD reference height is changed by varying the distance between the reference mark and the helicopter. As the helicopter develops vertical motion, the reference mark will vary in relation to the pole, providing the pilot with an instantaneous error signal.



$$h_{ac} = (h / x)\, y$$

$$H_G = h_{ac} - h_{pilot}$$

Figure 5.12
Free Flight Hover Method with HHMD

5.4.3.4 AIR REFERENCED

Air referenced free flight hover performance tests are an extension of the ground referenced tests. Testing at altitudes provides a readily available means of varying ρ. The tests are conducted by reference to a weighted, tufted line below the helicopter, by reference to a hover meter, or other low airspeed measuring device. The difficulty is achieving a true hover without reference to the ground. This is achieved by reference to a weighted, tufted line suspended from the helicopter. The string should be approximately

200 ft long with tufts attached at intervals to aid in airflow identification. The highest safe hover altitude is calculated and the tests begin at that altitude. The altitude is approached from below in a climb rather than approached from above in a descent as vortex ring state can develop. With the aircraft in a position slightly below the anticipated hover altitude, the pilot decreases airspeed slowly and approaches the hover. The pressure airspeed indicator is unreliable below approximately 20 kn and the pilot proceeds on aircraft feel, control positions, attitude, and low airspeed references. As airspeed is reduced further, the aircraft passes through translational lift and settles into the hover. If a weighted line is used, it is lowered at this time. The line bows opposite to the relative wind. Small control corrections are made to achieve a true hover. If a low airspeed system is used, control inputs are made to correct for any indicated translation.

After the hover is achieved, data for several rotor speeds are recorded. When changing N_R, be very careful to avoid vortex ring state. If vortex ring state is encountered, jettison the string if appropriate, lower power demand and fly out of the condition. The test is repeated for decreasing altitudes, while simultaneously reducing gross weight through fuel consumption. This sequence provides the greatest range of C_T during one flight. At each altitude, record the data for the permissible N_R range.

Air referenced free flight hover tests are difficult to perform from a pilot workload viewpoint due to the lack of visual cues to a true hover. In an OGE hover, the most important element of a stabilized hover is zero vertical/horizontal motion. The test height can be several feet in error and not produce any error in the test results.

5.4.3.5 DATA REQUIRED

Q, N_R, H_P, T_O, fuel used (FU, FC) or fuel remaining, hover height (gear or rotor height for ground referenced).

Note: Ground station (if available) record wind speed and direction.

5.4.3.6 TEST CRITERIA

1. Wind less than three kn (ground referenced).
2. Engine bleed air systems off or operated in normal flight mode.
3. Stabilized engine power demand.
4. Minimum power train oscillations.

5. Minimum vertical and horizontal translation.
6. Minimum cyclic control inputs.
7. No directional control inputs, orient into existing wind.

5.4.3.7 DATA REQUIREMENTS

1. Stabilize 15 s minimum before recording data.
2. Record stabilized data 30 s.
3. $N_R \pm 0.5$ rpm
4. $H_P \pm 0.5$ ft (air referenced).
5. Maximum drift ± 3 ft (ground reference)
6. Maximum hover height ± 1 ft IGE, ± 3 ft OGE (ground reference).

5.4.3.8 SAFETY CONSIDERATIONS/RISK MANAGEMENT

During all hover testing the most important safety consideration is aircraft response to an engine failure. All test locations should provide maximum opportunity to complete a successful landing following a power loss. Ground referenced tests must be conducted over a landing area clear of all obstructions. The air referenced tests must be conducted at an altitude and over terrain that will permit a successful landing. Operation in the avoid areas of the height velocity diagram can not be eliminated during all hover tests, but exposure must be minimized.

Thorough planning and crew briefing, including ground support personnel will minimize potential hazards. Procedures for weighted line jettison must be thoroughly understood. Tests should be conducted over unpopulated areas in the event of a line jettison. The weighted line must not be rigidly fixed to the helicopter or crewmember. Tail rotor clearance is a concern in the event of a jettison.

During all tests, vortex ring state should be avoided. If encountered, reduce power demand, increase airspeed, and fly out of the condition.

The crew should be alert to adverse handling qualities and unexpected performance degradation when changing N_R .

During the hover flight test various limits are approached. These limits are mechanical, structural, and aerodynamic. A mechanical limit restricts the aircraft operation because of a torque limit, rotational speed limit, temperature limit, etc. A time limit may involve any of the above mechanical limits and require pilot attention.

There are a number of structural limits imposed on hovering helicopters. Cargo deck loading, cargo hook limits, rotor blade strength, etc., may establish the maximum operating gross weight. The ability to load ballast or tether the helicopter by a cargo lift point may also establish the method used to determine hover performance.

The hover ceiling is a primary aerodynamic limit and depending upon the type of rotor system and configuration the CG location may be involved. In a tandem rotor helicopter, the aerodynamic limit of one rotor can be exceeded if the CG is significantly off-set from the mid rotor location. The resultant power loss is referred to as trim power. Although a wide range of CG locations is allowed in a tandem rotor helicopter configuration, be cautious when establishing test conditions and extrapolating mid CG hover performance to other CG loadings.

5.4.4 Tethered Hover

The tethered hover method is an exacting technique and produces excellent results. The aircraft is secured to the test surface by a cable (Figure 5.13). The tension in the cable and the weight of the cable is recorded and added to the weight of the aircraft. This test is conducted at two gross weights and a number of gear heights. A maximum take-off gross weight is selected to provide the highest allowable cable tension-gross weight combination, and provides high values of C_T. The lightest possible take-off weight allows the test aircraft to maintain cable tension down to very small values of C_T.

Some aircraft are configured specifically for this test by attaching a lifting harness to the main transmission-airframe mounting bolts. This eliminates loading of the airframe structure and provides higher values of cable tension than using a normal external cargo pick-up point. One tether hover configuration in use today, places the tension measurement and recording equipment on the ground, clear of the surface tether point. Another system uses a cockpit display so the pilot can see what the cable tension is, thus eliminating transmission of the data by radio from the ground.

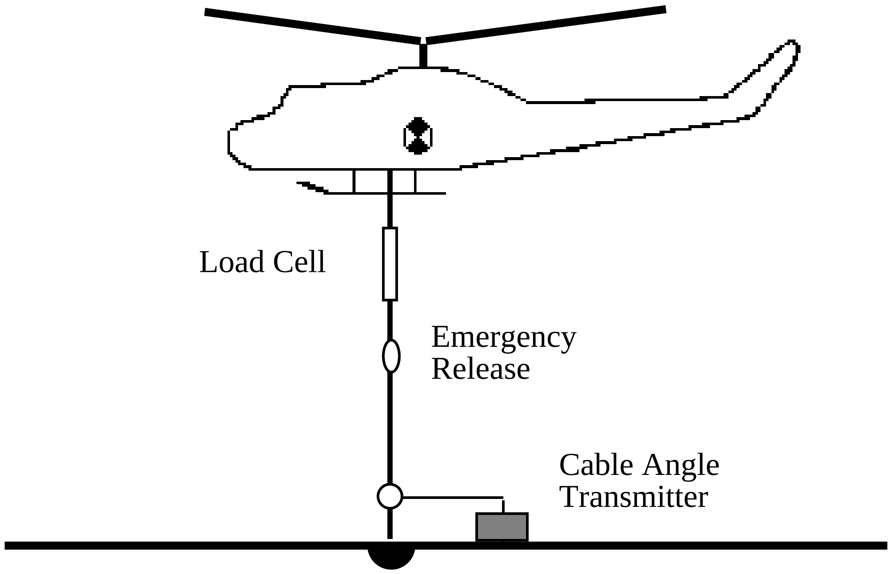


Figure 5.13
Tethered Hover Method

Some installations require one or two ground observers to give hand signals (or make radio transmissions) to advise the pilot of lateral or longitudinal displacements from vertical. A more desirable arrangement provides equipment which measures the angle of the cable to the airframe and transmits this information to a cockpit display of pitch and roll cross bars (ILS type instrument).

If only one rotor speed is investigated, the test aircraft may not achieve the scope of C_T values desired. In such cases, the rotor speed (N_R) is decreased in steps to the minimum allowable for the highest possible C_T (maximum gross-weight-tension) and increased to the highest N_R for the minimum C_T possible (minimum gross weight-tension). When changing N_R, reduce the tension in the cable to near zero to avoid harmonic oscillations in the total system and lessen control problems at high power settings and low rotor speeds.

The greatest limitation of the tethered hover method is the cost of test equipment and installation of a tether point on the aircraft. If cockpit displays of cable angle and cable tension are not available, the pilot will be at a definite disadvantage and will require information relayed from a crewmember or ground observer. Power train oscillations cause cable bounce and depending on the gross weight and rotor speed, can be an additional limitation.

The pilot should receive cable tension information in the cockpit by a visual instrument presentation indicating pounds of load. An instrument similar to antisubmarine warfare (ASW) helicopter hover displays or ILS approaches is used to present horizontal displacement information. The cable angle may be measured at the ground and relayed to the cockpit. This location gives a ground-stabilized indication and eliminates those due to aircraft pitch attitude changes. The emergency releases are arranged to retain the load cell with the aircraft (drop the tether) and have pull-away electrical connections for the cable angle measurement system at the airframe (drop the wire from the aircraft).

5.4.4.1 DATA REQUIRED

Q, N_R, Cable tension (T), cable angle, H_{P_O}, T_O, fuel used (FU, FC) or fuel remaining, hover height.

Note: Ground station (if available) record wind speed and direction.

5.4.4.2 TEST CRITERIA

1. Wind less than three kn.
2. Engine bleed air systems off or operated in normal flight mode.
3. Stabilized engine power demand.
4. Minimum power train oscillations.
5. Minimum cable angle deflection.
6. Minimum cyclic control inputs.
7. No directional control inputs, orient into existing wind.

5.4.4.3 DATA REQUIREMENTS

1. Stabilize 15 s minimum before recording data.
2. Record stabilized data 30 s.
3. $N_R \pm 0.5$ rpm.
4. Maximum cable angle 8 deg.

5.4.4.4 SAFETY CONSIDERATIONS/RISK MANAGEMENT

Additional safety considerations for tethered hover involve the use of the tether. The cable jettison must be checked prior to each hookup. Discharge electrical charge to ground from the aircraft prior to connecting the cable. Thorough coordination with all test team personnel is mandatory. One individual must be in charge of all ground support personnel. The aircraft crew chief is a logical choice. Exercise care when ballasting the aircraft or changing cables. The crew chief normally assists in establishing a vertical hover and when applying cable tension with radio transmissions of cable angle and cable length remaining. During the tests, he and all ground personnel must remain clear of possible landing areas in the event of an engine failure. In the event of a cable jettison, the ground support personnel must be well clear. A tethered hover cable released under tension can be deadly.

5.5 DATA REDUCTION

5.5.1 General

USNTPS uses a computer based data reduction routine for helicopter performance data reduction. This section covers a manual method of hover performance data reduction with an introduction to the computer based data reduction routine.

Whether free flight or tethered hover test methods are used, the data reduction required is very similar. The hover height (gear height) for IGE and low OGE, the gross weight, and power required must be calculated. Gross weight of the helicopter is either aircraft gross weight or aircraft gross weight plus cable tension for tethered hover. The power required calculations are the same regardless of method used.

All observed data must be corrected for instrument and position errors as appropriate.

5.5.2 Manual Data Reduction

The following equations are used in the manual hover performance data reduction:

$$H_{P_c} = H_{P_o} + \Delta H_{P_{ic}} + \Delta H_{pos} \qquad \textit{eq 5.47}$$

Where:

H_{P_C} - Calibrated pressure altitude

H_{P_O} - Observed pressure altitude

$\Delta H_{P_{ic}}$ - Altimeter instrument correction

ΔH_{pos} - Altimeter position error.

$$T_a = T_o + \Delta T_{ic} \qquad \textit{eq 5.48}$$

Where:

T_a - Ambient temperature

T_o - Observed temperature

ΔT_{ic} - Temperature instrument correction.

$$\sigma_T = \frac{\delta}{\theta} = \frac{\delta}{\left(\frac{T_a}{T_{ssl}}\right)} \qquad \textit{eq 5.49}$$

Where:

σ_T - Test density ratio

δ - Pressure ratio

θ - Temperature ratio

T_a - Ambient temperature

T_{ssl} - Standard sea level temperature.

$$ESHP_T = K_Q\,(Q)\,(N_{R_T})$$ *eq 5.50*

Where:

$ESHP_T$	- Test engine shaft horsepower
K_Q	- Engine torque constant (aircraft unique)
Q	- Engine torque
N_{R_T}	- Test main rotor speed.

$$ESHP_{ref} = \frac{ESHP_T}{\sigma_T}\left(\frac{N_{R_S}}{N_{R_T}}\right)^3$$ *eq 5.46*

$$GW = ESGW - FU$$ *eq 5.51*

$$GW = ESGW - FU + \text{Cable Tension}$$ *eq 5.52*

Where:

GW	- Gross weight
ESGW	- Engine start gross weight
FU	- Fuel used.

$$W_{ref} = \frac{W_T}{\sigma_T}\left(\frac{N_{R_S}}{N_{R_T}}\right)^2$$ *eq 5.45*

$$\Omega R = K_{GR}\left(N_{R_T}\right)\left(\frac{2\pi}{60}\right)(R)$$ *eq 5.53*

Where:

ΩR	- Rotational velocity
K_{GR}	- Gear ratio (Aircraft unique)
N_{R_T}	- Test main rotor speed
R	- Rotor radius.

$$C_P = \frac{P}{\rho_a\ A_D\ (\Omega R)^3} = f\left(\frac{P}{\sigma}, N_R\right) \qquad \textit{eq 5.30}$$

$$C_T = \frac{T}{\rho_a\ A_D\ (\Omega R)^2} = f\left(\frac{T}{\sigma}, N_R\right) \qquad \textit{eq 5.29}$$

$$M' = \frac{\left(\frac{C_T^{\frac{3}{2}}}{\sqrt{2}}\right)}{C_P} \qquad \textit{eq 5.39}$$

From observed hover performance data, compute $ESHP_{ref}$, W_{ref}, C_P, C_T, and M' as follows:

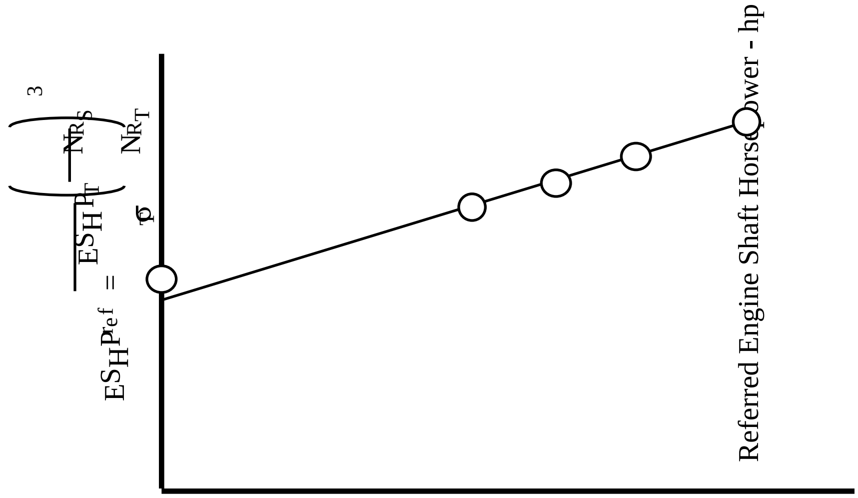
Parameter	Notation	Formula	Units	Remarks
(1) Observed pressure altitude	H_{P_O}		ft	
(2) Calibrated pressure altitude	H_{P_C}	eq 5.47	ft	Instrument & position corrections
(3) Observed temperature	T_O		°C	
(4) Ambient temperature	T_a	eq 5.48	°C	Instrument correction, ignore recovery and Mach correction
(5) Pressure ratio	δ			From tables at step (2)
(6) Test density ratio	σ_T	eq 5.49		
(7) Test main rotor speed	N_{R_T}		%	
(8) Engine torque	Q		%, psi	Total torque
(9) Test ESHP	$ESHP_T$	eq 5.50	hp	

(10) Referred ESHP	$ESHP_{ref}$	eq 5.46	hp	
(11) Fuel counts	FC		cts	
(12) Fuel used	FU	FC (gal/ct) (lb/gal)	lb	gal/ct - Lab calibration lb/gal - fuel density
(13) Gross weight	GW	eq 5.51	lb	
(13a) Gross weight	GW	eq 5.52	lb	Tethered hover
(14) Referred weight	W_{ref}	eq 5.45	lb	
(15) Rotational velocity	ΩR	eq 5.53	ft/s	
(16) Power coefficient	C_P	eq 5.30		
(17) Thrust coefficient	C_T	eq 5.29		
(18) Figure of merit	M'	eq 5.39		

Plot $ESHP_{ref}$ as a function of $W_{ref}^{\frac{3}{2}}$ (referred hover performance) for each hover height IGE and for OGE, (Figure 5.14).

Plot C_P as a function of C_T (nondimensional hover performance) for each hover height IGE and for OGE (Figure 5.15).

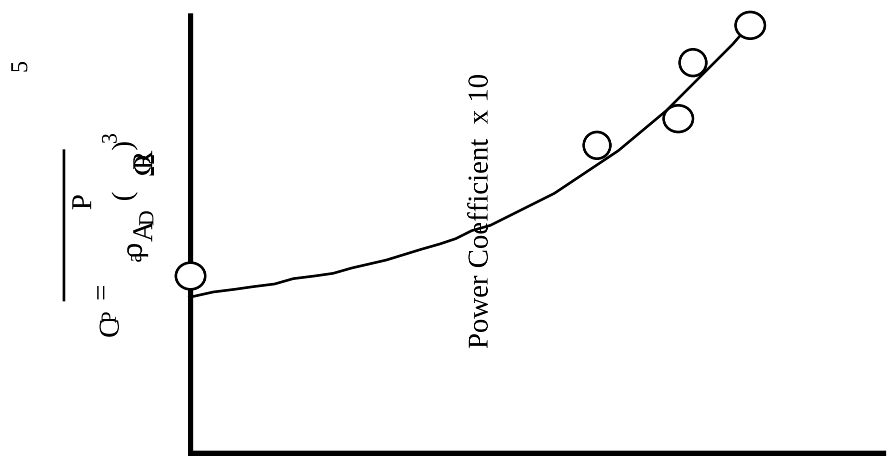
Plot M' as a function of C_T (hover efficiency) (Figure 5. 16).



Referred Weight $^{\frac{3}{2}}$ x 10^{-3} - lb

$$W_{ref} = \frac{W_T}{\sigma_T}\left(\frac{N_{R_S}}{N_{R_T}}\right)^2$$

Figure 5.14
Referred Hover Performance



Thrust Coefficient x 10^4

$$C_T = \frac{T}{\rho_a A_D (\Omega R)^2}$$

Figure 5.15
Nondimensional Hover Performance

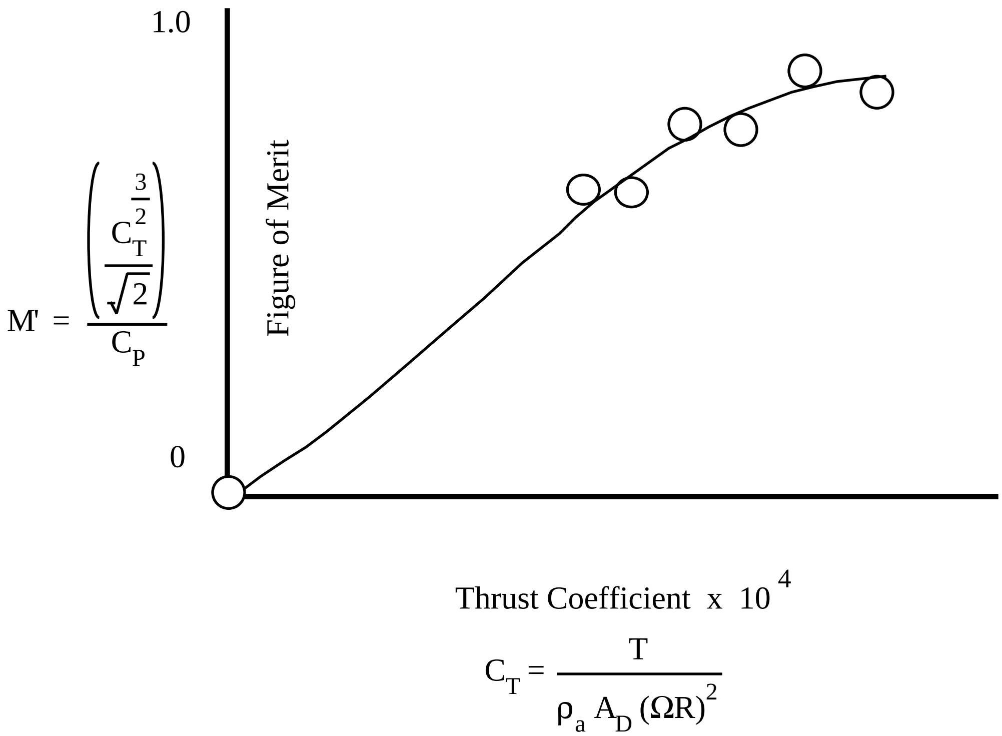


Figure 5.16
Hover Efficiency

5.5.3 USNTPS Computer Data Reduction

This section contains a brief summary of the assumptions and logic used in the USNTPS helicopter HOVER computer data reduction program.

Hover performance is assumed to be in the form:

$$ESHP_{ref} = f(W_{ref}) \qquad \textit{eq 5.54}$$

Where:

$ESHP_{ref}$	- Referred engine shaft horsepower
W_{ref}	- Referred weight.

Specifically $ESHP_{ref}$ is assumed to be a linear function of $W_{ref}^{\frac{3}{2}}$:

$$ESHP_{ref} = A_0 + A_1 W_{ref}^{\frac{3}{2}} \qquad \textit{eq 5.55}$$

Where:

$ESHP_{ref}$	- Referred engine shaft horsepower
A_0, A_1	- Constants
W_{ref}	- Referred weight.

The following equations are used in the calculations:

$$ESHP_{ref} = \frac{ESHP_T}{\sigma_T}\left(\frac{N_{R_S}}{N_{R_T}}\right)^3 \qquad \textit{eq 5.46}$$

$$W_{ref} = \frac{W_T}{\sigma_T}\left(\frac{N_{R_S}}{N_{R_T}}\right)^2 \qquad \textit{eq 5.45}$$

Where:

W_T - Test weight + load cell (if applicable).

$$ESHP_T = K_Q\,(Q)\,(N_{R_T}) \qquad \textit{eq 5.50}$$

$$\delta = \frac{P_a}{P_{ssl}} = \left(1 - \frac{\lambda_{ssl}\ H_P}{T_{ssl}}\right)^{\frac{g_{ssl}}{g_c\ \lambda_{ssl}\ R}} \qquad \textit{eq 5.56}$$

Where:

δ	- Pressure ratio
P_a	- Ambient pressure
P_{ssl}	- Standard sea level pressure, 2116.2 psf
H_P	- Pressure altitude
λ_{ssl}	- Standard sea level lapse rate, 0.00198 ∞K/ft
g_{ssl}	- Standard sea level gravity, 32.17 ft/ s2
g_c	- 32.17 lb - mass / slug
R	- Engineering gas constant for air, 96.03 ft - lb/ lbm- ∞K.

$$\theta = \frac{T_a}{T_{ssl}} = \frac{OAT + 273.15}{288.15}$$

eq 5.57

Where:

θ	- Temperature ratio
T_a	- Ambient temperature
T_{ssl}	- Standard sea level temperature, 288.15 ∞K
OAT	- Outside air temperature (total temperature).

$$\sigma = \frac{\rho}{\rho_{ssl}} = \frac{\delta}{\theta}$$

eq 5.58

Where:

σ	- Density ratio
ρ	- Density
ρ_{ssl}	- Standard sea level density, 0.0023769 slug/ ft3
δ	- Pressure ratio
θ	- Temperature ratio.

$$C_P = \frac{P}{\rho_a \, A_D \, (\Omega R)^3} = f\left(\frac{P}{\sigma}, N_R\right) \qquad \textit{eq 5.30}$$

$$C_T = \frac{T}{\rho_a \, A_D \, (\Omega R)^2} = f\left(\frac{T}{\sigma}, N_R\right) \qquad \textit{eq 5.29}$$

Where:

T - Thrust is assumed to equal weight in a hover (plus cable tension)

$$\Omega R = K_{GR} \left(N_{R_T}\right) \left(\frac{2\pi}{60}\right) (R) \qquad \textit{eq 5.53}$$

$$M' = \frac{\left(\frac{C_T^{\frac{3}{2}}}{\sqrt{2}}\right)}{C_P} \qquad \textit{eq 5.39}$$

$$ESHP = \frac{ESHP_{ref} \, (\sigma)}{\left(\frac{N_{R_S}}{N_{R_T}}\right)^3} \qquad \textit{eq 5.59}$$

Where:

ESHP	- Engine shaft horsepower
$ESHP_{ref}$	- Referred engine shaft horsepower
σ	- Density ratio
N_{R_S}	- Standard main rotor speed
N_{R_T}	- Test main rotor speed.

$ESHP_{ref}$, W_{ref}, $W_{ref}^{\frac{3}{2}}$, $C_{T,}$ C_P, M', and blade tip Mach number are output as tabular data and graphic data in the form of Figures 5.14, 5.15, 5.16.

A further calculation of power required to hover as a function of pressure altitude for 4 user input gross weights is calculated as follows:

(1) Select N_R .

(2) Select weight.

(3) Select H_P .

(4) Calculate δ using Equation 5.56.

(5) Choose standard day or hot day temperatures (°K)

$T_{a_S} = 288.15 - \lambda_{ssl} \ H_P$.

$T_{a_H} = 312.61 - \lambda_{ssl} \ H_P$.

(6) Calculate σ using Equation 5.58.

(7) Calculate W_{ref} using Equation 5.45.

(8) Calculate $ESHP_{ref}$ using Equation 5.55.

(9) Calculate ESHP using Equation 5.59.

Data are output in tabular and graphic form of pressure altitude as a function of power required to hover for both hot and standard day for the 4 user input weights. A sample for standard day is shown in Figure 5.17.

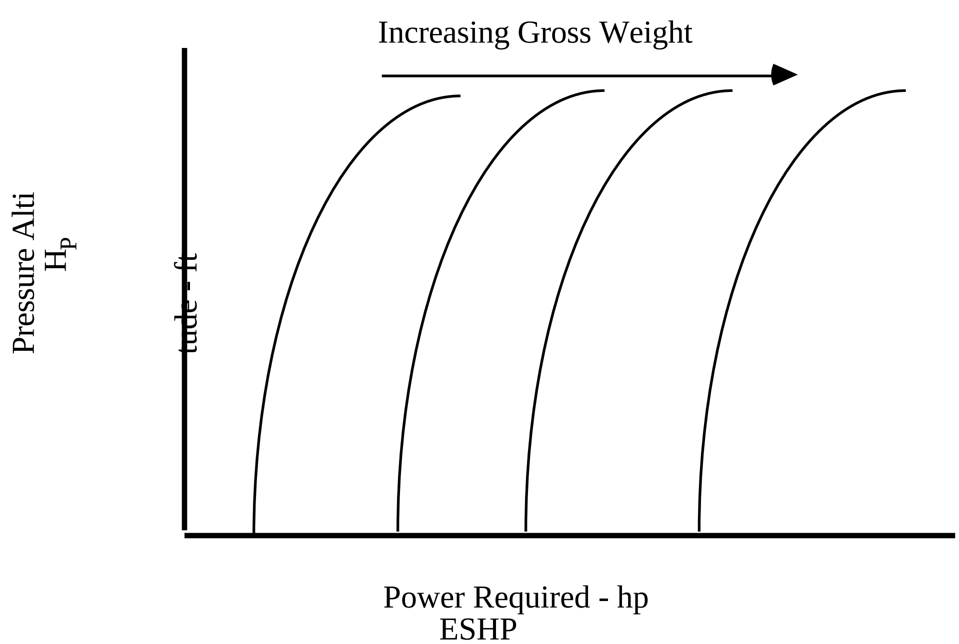


Figure 5.17
Hover Performance

5.6 DATA ANALYSIS

5.6.1 General

Observed hover performance data is reduced to nondimensional and generalized (also called referred) hover performance data. This data is analyzed to determine hover performance, hover ceiling, and IGE/OGE transition.

5.6.2 Hover Performance

Hover performance in the form of power required to hover as a function of calibrated pressure altitude can be extracted manually from referred hover performance in much the same manner as the USNTPS HOVER performance computer data reduction.

Use referred hover performance IGE or OGE as desired (Figure 5.14) and determine hover performance in the form of pressure altitude as a function of power required to hover (ESHP) using the following equations:

$$W_{ref} = \frac{W_S}{\sigma}$$

eq 5.60

Where:

W_{ref} - Referred weight

W_S - Standard weight

σ - Density ratio.

$$ESHP = ESHP_{ref}(\sigma)$$

eq 5.61

Where:

ESHP - Engine shaft horsepower

$ESHP_{ref}$ - Referred engine shaft horsepower

σ - Density ratio.

From referred hover performance (Figure 5.14) compute H_P and ESHP as follows:

Parameter	Notation	Formula	Units	Remarks
(1) Standard weight	W_S		lb	Select 3 representative weights
(2) Pressure altitude	H_P		ft	Select a series of H_P from sea level to > anticipated hover ceiling
(3) Density ratio	σ			From tables at step (2)
(4) Referred weight	W_{ref}	eq 5.60	lb	Corresponding to selected H_P & GW
(5) Referred ESHP	$ESHP_{ref}$		hp	From Figure 5.14 for corresponding W_{ref}
(6) ESHP	ESHP	eq 5.61	hp	Power required to hover at selected GW & H_P

(7) Repeat as necessary to cover selected GW and altitude range.

(8) Perform process for both standard day and hot day as required.

Plot hover performance, altitude (H_P) as a function of power required to hover (ESHP) at that altitude (Figure 5.17).

Additional plots of hover performance for conditions of interest are constructed in the same way. Conditions of interest include: IGE (various gear heights), OGE, and hot day conditions, as well as the specification conditions.

5.6.3 Hover Ceiling

Hover ceiling is the combination of airframe hover performance (power required) and engine performance(power available).

The hover ceiling in the form of pressure altitude (H_P) as a function of the maximum gross weight (GW) which can sustain a hover is calculated graphically from Figure 5.17 and engine power available.

Figure 5.17 is overlayed with power available as a function of pressure altitude to construct a working plot (Figure 5.18).

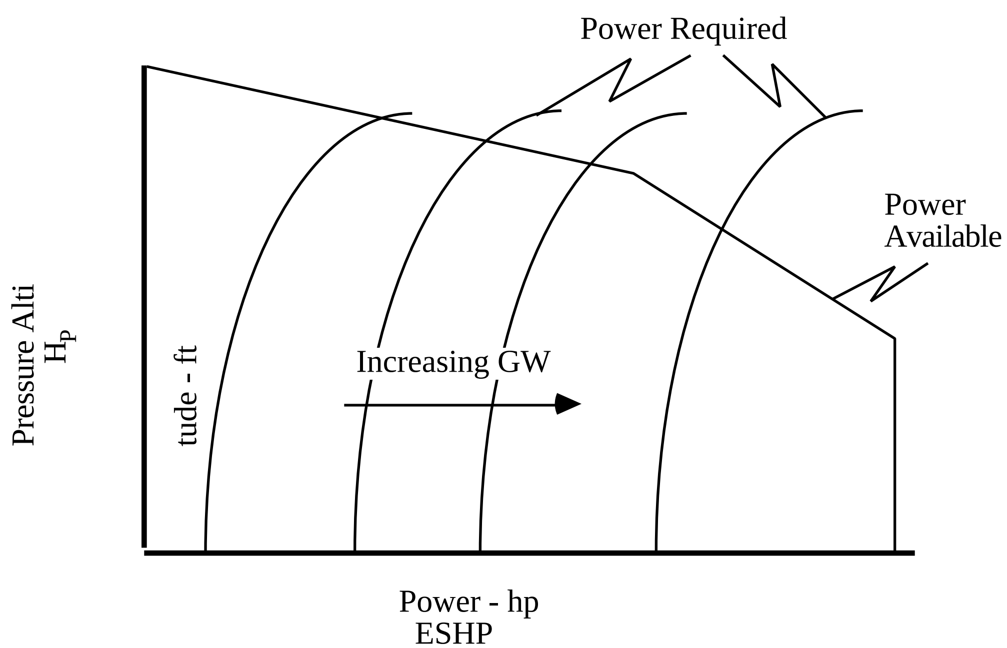


Figure 5.18
Hover Ceiling Working Plot

The intersection of the power required to hover and the power available, hover ceiling, is plotted as pressure altitude as a function of gross weight (Figure 5.19).

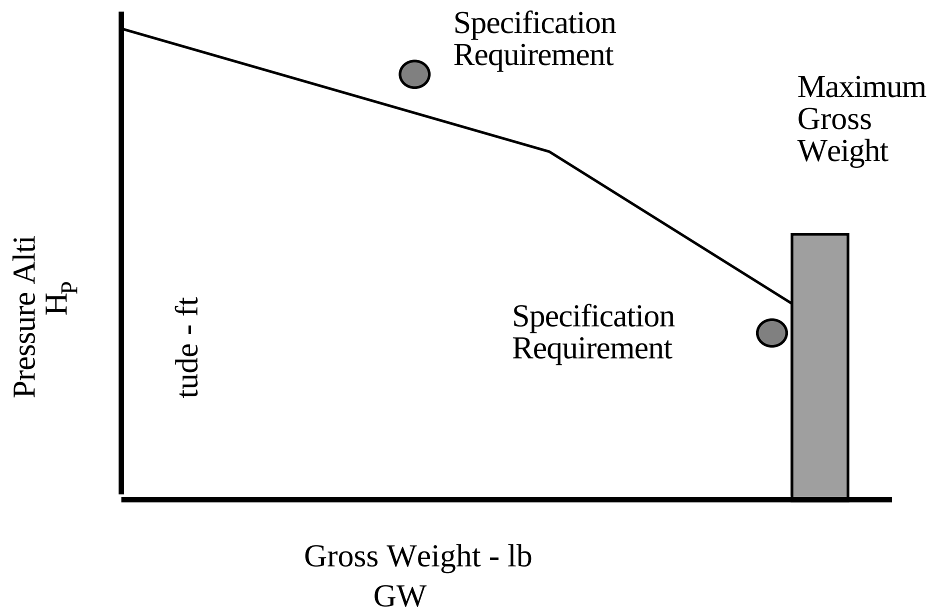


Figure 5.19
Hover Ceiling

Additional plots of hover ceiling for a hot day can be constructed in the same way.

5.6.3.1 POWER MARGIN

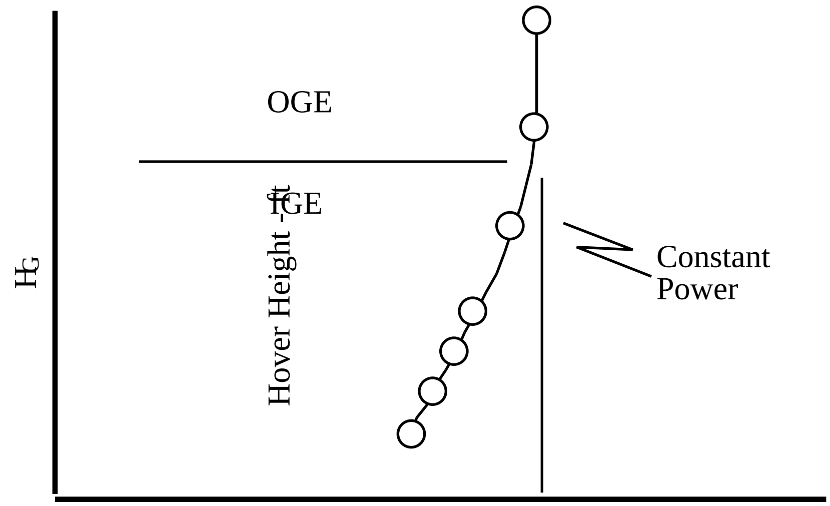
Hover ceiling plots are instructed to check specification compliance and mission suitability. For specification compliance, the specification hover ceiling should be compared to hover ceiling plots based on data taken under conditions as close to ideal as practicable. However, mission suitability must be checked for operations under conditions which are less than ideal. The most practical way to accomplish both tasks simultaneously is to determine an appropriate power margin during hover testing.

A power margin is defined as a deliberate reduction in excess power (power available minus power required) during the determination of a hover ceiling, to provide for a margin to maneuver the aircraft, counteract gusts, hover out of the windline, allow for engine degradation to fleet average, etc. Although typical mission relatable power margins average around 5%, actual power margins will vary between aircraft types due to differences in design, configuration, etc. Determining an appropriate power margin accounting for increases in power required can be accomplished during hover testing by noting any additional power requirements experienced while hovering in gusty conditions, out of the windline, or performing mission maneuvers representative of hovering under conditions of little to no excess power (moderate rate pedal turns, moderate collective movements for altitude maintenance, etc.). Determining an appropriate power margin accounting for decreases in power available due to engine degradation involves an engineering judgment of how much more powerful the engines under test are than engines which are fleet representative, and is normally a process performed in conjunction with the aircraft material manager. In the USNTPS syllabus, engines are assumed to be fleet representative for the purpose of calculating power margins. Therefore, at USNTPS power margins are calculated solely on the basis of power required changes noted during testing.

Once determined, a power margin should be applied during the calculation of mission relatable hover ceilings. Since a power margin is a reduction in excess power, it is appropriate to apply it as either an increase in power required or as a decrease in power available. However, care must be taken when comparing a percentage reduction in power available to an equal percentage increase in power required, since in general, these will not equal the same horsepower. Although other methods may be used, USNTPS applies power margins by converting any power required increases noted during testing into an excess power decrease in horsepower, then applying the excess power decrease uniformly as an effective reduction in power available.

5.6.4 IGE / OGE Transition

Data from free flight ground referenced hover tests and tethered hover can be used to determine the transition from IGE to OGE for a particular helicopter. Hover height, gear height (HG), is plotted as a function of referred ESHP ($ESHP_{ref}$) (Figure 5.20). As the hover height increases, the power required to hover increases until the helicopter is OGE. For all OGE heights (constant HD) the power required to hover is constant, reflected as the vertical portion of the curve in Figure 5.20. Once this altitude is determined, further OGE hover tests can proceed confidently.



$$ESHP_{ref} = \frac{ESHP_T}{\sigma_T}\left(\frac{N_{R_S}}{N_{R_T}}\right)^3$$

Figure 5.20
Incremental Hover Performance

5.7 MISSION SUITABILITY

The great tactical advantage of the helicopter is its ability to operate in the low speed environment and to hover. From a performance perspective then, hover performance takes on great importance. This importance is expressed in relation to the payload that can be carried, the useful fuel load, or the areas of the world where the helicopter can operate. In determining the mission suitability of a helicopter's hover performance, the hover ceiling should be determined for all applicable mission gross weights, for standard day and hot day conditions, in ground effect and out of ground effect, and applying a mission relatable power margin as previously discussed. If mission suitability is deficient in any areas, mission relation may be made in terms of amount and percentage of fuel or payload reduction required to hover at a given altitude, or, if a given payload and fuel load is required for the mission, the areas of the world and altitudes where the helicopter cannot operate.

5.8 SPECIFICATION COMPLIANCE

5.8.1 General

The detail specification for the model helicopter typically lists one or two hover ceiling guarantees. One guarantee is for a particular gross weight, at military rated power, standard day conditions and is stated in terms of OGE hover altitude. Depending on the type of helicopter, a second guarantee may or may not be listed. Some helicopters have an OGE hover ceiling guarantee for other than standard day conditions (hot day conditions). All guarantees are given for 3 knots of wind or less. Occasionally, a guarantee is stated as a percent of military power at a particular altitude rather than an altitude at a given power rating.

5.8.2 Power Available

No matter how the guarantees are stated for hover ceilings, the variation of engine shaft horsepower available with pressure altitude must be determined prior to computing the hover ceiling. If a specification engine is required by the detail specification for use in determining the hover ceiling, then the power available curve for the specification engine must be used. However, if the specification engine is not required, the actual installed power available should be used in the computations. Installed power available should be determined as discussed in Chapter 4.

5.9 GLOSSARY

5.9.1 Notations

$A_0,\ A_1$	Constants	
ASW	Antisubmarine warfare	
A_D	Rotor disc area	ft^2
b	Number of blades	
c	Blade chord	ft
$C_1,\ C_2$	Constants	
CG	Center of gravity	in
C_{d_i}	Blade element induced drag coefficient	

$\overline{C}_{d_i}$	Average blade element induced drag coefficient	
C_{d_0}	Blade element profile drag coefficient	
$\overline{C}_{d_0}$	Average blade element profile drag coefficient	
C_{ℓ}	Blade element lift coefficient	
$\overline{C}_{\ell}$	Average blade element lift coefficient	
C_L	Lift coefficient	
C_P	Power coefficient	
C_T	Thrust coefficient	
dD_0	Blade element profile drag	lb
dF	Blade element torque force	lb
dL	Blade element lift	lb
dP	Blade element power required	lb
dr	Blade element radius	ft
dR	Blade element resultant aerodynamic force	lb
dS	Blade element area	ft^2
dT	Blade element thrust	lb
ESGW	Engine start gross weight	lb
ESHP	Engine shaft horsepower	hp
$ESHP_{ref}$	Referred engine shaft horsepower	hp
$ESHP_S$	Standard engine shaft horsepower	hp
$ESHP_T$	Test engine shaft horsepower	hp
F	Force	lb
FC	Fuel counts	cts
FU	Fuel used	lb
g_c	32.17 lb - mass / slug	
g_{ssl}	Standard sea level gravity	$32.17\ ft/s^2$
GW	Gross weight	lb
h	Height	ft
HHMD	Hover height measuring device	
H_D	Density altitude	ft
H_G	Gear height	ft
H_P	Pressure altitude	ft

H_{P_C}	Calibrated pressure altitude	ft
H_{P_O}	Observed pressure altitude	ft
$\Delta H_{P_{ic}}$	Altimeter instrument correction	ft
ΔH_{pos}	Altimeter position error	ft
IGE	In-ground effect	
ILS	Instrument landing system	
K_{GR}	Gear ratio (aircraft unique)	
K_Q	Engine torque constant (aircraft unique)	
m	Mass	
M	Mach number	
M′	Figure of merit	
N_R	Main rotor speed	%, rpm
N_{R_S}	Standard main rotor speed	%, rpm
N_{R_T}	Test main rotor speed	%, rpm
OAT	Outside air temperature (total temperature)	°C
OGE	Out-of-ground effect	
P	Power (required to hover)	hp
P_{Acc}	Accessory power required	hp
P_b	Power required for b blades	hp
P_i	Induced power	hp
P_{loss}	Power loss	hp
P_{MR}	Main rotor power required	hp
P_0	Profile power	hp
P_{TOTAL}	Total power (required to hover)	hp
P_{TR}	Tail rotor power required	hp
P_a	Ambient pressure	psi, psf
P_{ssl}	Standard sea level pressure	2116.2 psf
P_T	Free stream total pressure	psi, psf
$P_{T\,d}$	Downstream total pressure	psi, psf
$P_{T\,u}$	Upstream total pressure	psi, psf
ΔP	Pressure change	psi, psf

q_e	Blade element dynamic pressure	psi, psf
q_r	Wake dynamic pressure	psi, psf
Q	Engine torque	%, psi
r	Radius to blade element	ft
R	Engineering gas constant for air	96.03 ft - lb/ lb_m- °K
	Rotor radius	ft
RSHP	Rotor shaft horsepower	hp
$RSHP_S$	Standard rotor shaft horsepower	hp
t	Time	
T	Temperature	°C, °K
T_a	Ambient temperature	°C
T_{a_H}	Hot day ambient temperature	°C
T_{a_S}	Standard day ambient temperature	°C
T_o	Observed temperature	°C
T_{ssl}	Standard sea level temperature	15°C, 288.15°K
ΔT_{ic}	Temperature instrument correction	°C
T	Thrust	lb
	Cable tension	lb
T_b	Thrust of b blades	lb
v_i	Induced velocity at hover	ft/s
v_w	Ultimate wake velocity	ft/s
V_R	Resultant velocity through rotor	ft/s
ΔV	Change in velocity	fl/s
W	Weight	lb
W_{ref}	Referred weight	lb
W_S	Standard weight	lb
W_T	Test weight	lb

5.9.2 Greek Symbols

α (Alpha)	Angle of attack	deg
α_i	Induced angle	deg
δ (Delta)	Pressure ratio	
η_m (Eta)	Mechanical efficiency	
λ_{ssl} (Lambda)	Standard sea level lapse rate	0.00198 °K/ft
θ (Theta)	Temperature ratio	
ρ (Rho)	Density	slug/ft^3
ρ_a	Ambient air density	slug/ft^3
ρ_S	Standard density	slug/ft^3
ρ_T	Test density	slug/ft^3
ρ_{ssl}	Standard sea level air density	0.0023769 slug/ft^3
σ (Sigma)	Density ratio	
σ_T	Test density ratio	
σ_R	Rotor solidity ratio	
Ω (Omega)	Rotor angular velocity	rad/s
Ωr	Blade element rotational velocity	ft/s
ΩR	Blade rotational velocity	ft/s
ΩR_S	Standard blade rotational velocity	ft/s
ΩR_T	Test blade rotational velocity	ft/s

5.10 REFERENCES

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CHAPTER SIX

VERTICAL CLIMB PERFORMANCE

VERTICAL CLIMB PERFORMANCE

CHAPTER SIX

FIGURES

CHAPTER SIX

EQUATIONS

$$\Delta RSHP = RSHP_A - RSHP_{req}$$ *eq 6.1*

$$P_{\Delta PE} = W\left(\frac{dh}{dt}\right)$$ *eq 6.2*

$$V' = \frac{\Delta RSHP(33{,}000)}{W}$$ *eq 6.3*

$$V' = \frac{\Delta ESHP\left(\eta_m\right)(33{,}000)}{W}$$ *eq 6.4*

$$F = \frac{dm}{dt}\Delta V$$ *eq 6.5*

$$T = \rho\, A_D\,(V_V + v_{i_v})\,(2v_{i_v})$$ *eq 6.6*

$$\frac{T}{2\,\rho\, A_D} = v_{i_v}\,(V_V + v_{i_v})$$ *eq 6.7*

$$v_i^2 = v_{i_v}^2 + v_{i_v} V_V$$ *eq 6.8*

$$v_{i_v} = \sqrt{\left(\frac{V_V}{2}\right)^2 + v_i^2} - \frac{V_V}{2}$$ *eq 6.9*

$$P_{A_h} = P_{A_v}$$ *eq 6.10*

$$P_{p_h} + P_{0_h} + P_{i_h} + \Delta P_h = P_{p_v} + P_{0_v} + P_{i_v} + \Delta P_v$$ *eq 6.11*

$$W\left(v_i + V'\right) = W\left(v_{i_v} + V_v\right)$$ *eq 6.12*

$$\frac{V_v}{V'} = 1 + \frac{1}{\frac{V'}{v_i} + 1}$$ *eq 6.13*

$$v_i = \sqrt{\frac{W}{2\rho A_D}}$$ *eq 6.14*

$$H_{P_c} = H_{P_o} + \Delta H_{P_{ic}} + \Delta H_{pos}$$ *eq 6.15*

$$T_a = T_o + \Delta T_{ic}$$ *eq 6.16*

$$\sigma_T = \frac{\delta}{\theta} = \frac{\delta}{\left(\frac{T_a}{T_{ssl}}\right)}$$ *eq 6.17*

$$ESHP_h = K_Q\,(Q_h)\,(N_{R_h})$$ *eq 6.18*

$$ESHP_{climb} = K_Q(Q_{climb})\,(N_{R_{climb}})$$ *eq 6.19*

$$\Delta ESHP = ESHP_{climb} - ESHP_h$$ *eq 6.20*

$$GW = ESGW - FU$$ *eq 6.21*

$$V' = \frac{(33{,}000)(\Delta ESHP)}{GW}$$ *eq 6.22*

$$K_{H_P} = \frac{T_a + 273.15}{T_S + 273.15}$$ *eq 6.23*

$$V_v = K_{Hp}\left(\frac{dh}{dt}\right)$$ *eq 6.24*

$$v_i = 60 \sqrt{\frac{GW}{2\,\rho\,A_D}}$$ *eq 6.25*

$$V' = \frac{\left(ESHP_A - ESHP_h\right)(33{,}000)}{W}$$ *eq 6.26*

$$V_V = \left(\frac{V_V}{V'}\right) V'$$ *eq 6.27*

$$ESHP_{ref} = A_0 + A_1\, W_{ref}^{\frac{3}{2}}$$ *eq 6.28*

$$\theta = \frac{T_a}{T_{ssl}} = \frac{OAT + 273.15}{288.15}$$ *eq 6.29*

$$\delta = \left[1 - \left(\frac{\lambda_{ssl}\ H_P}{T_{ssl}}\right)\right]^{\frac{g_{ssl}}{g_c\,\lambda_{ssl}\,R}}$$ *eq 6.30*

$$\theta = \frac{\delta}{\theta}$$ *eq 6.31*

$$V' = \frac{\left(ESHP_{climb} - ESHP_h\right)(33{,}000)}{W_T}$$ *eq 6.32*

$$\frac{T_T}{T_S} = \frac{OAT + 273.15}{T_{ssl} - \lambda_{ssl}\, H_{P_h}}$$ *eq 6.33*

$$V_V = 60 \left(\frac{T_T}{T_S}\right)\left(\frac{\Delta H_P}{\Delta t}\right)$$ *eq 6.34*

$$v_i = 60\sqrt{\frac{W_T}{2\rho_{ssl}\,\sigma\,A_D}}$$ *eq 6.35*

$$\theta = 1 - \left(\frac{\lambda_{ssl}}{T_{ssl}}\right) H_P$$

eq 6.36

$$\theta = \frac{312.61 - \lambda_{ssl} H_P}{T_{ssl}}$$

eq 6.37

$$W_{ref} = \frac{W_T}{\sigma_T}\left(\frac{N_{R_S}}{N_{R_T}}\right)^2$$

eq 6.38

$$ESHP_h = ESHP_{ref}\sigma_S\left(\frac{N_R}{N_{R_S}}\right)^3$$

eq 6.39

$$ESHP_{A_2} = ESHP_{A_1}\left(\frac{100 - \text{margin}}{100}\right)$$

eq 6.40

CHAPTER SIX

VERTICAL CLIMB PERFORMANCE

6.1 INTRODUCTION

This chapter deals with the general aspects of determining helicopter vertical climb performance. The theory of a rotor in axial flight is discussed briefly in terms of momentum and blade element analysis. The test method discussed here is relatively simple yet produces a very thorough evaluation of the vertical climb performance. Test day vertical climb data are used to determine the vertical climb correction factor and standard day vertical climb performance. Additionally, a generalized data reduction technique is presented. First order approximations are used where applicable to reduce the data reduction workload. Service suitability and requirements of the military specifications are examined where applicable.

6.2 PURPOSE OF TEST

The purpose of this test is to investigate aircraft vertical climb performance characteristics to achieve the following objectives:

1. Determine the vertical climb correction factor for use in computing vertical climb performance.
2. Evaluate the pertinent requirements of the detailed model specification.
3. Define mission suitability problems.

6.3 THEORY

6.3.1 General

The analysis of the helicopter in a vertical climb is a simple extension of hover analysis. Although directed primarily at the main rotor, there may be significant effects on mechanical efficiency, η_m, and parasite drag to consider. The value of η_m changes as airspeed increases from zero due to changes in tail rotor power required.

In hover, as in a stabilized climb, the main rotor thrust is equal to the aircraft weight plus airframe parasite drag resulting from rotor downwash. For small rates of climb, the parasite drag is essentially equal to that in hover, but may increase significantly at high rates of climb.

For simplicity, mechanical efficiency is assumed constant and parasite drag is assumed zero for vertical climb analysis.

6.3.2 Energy Analysis

If the helicopter has power available in excess of that required to hover, this excess power may be used to produce climbing flight and change the potential energy of the helicopter. The excess power that is available to the main rotor in hover, $\Delta RSHP$, is determined from hover tests as:

$$\Delta RSHP = RSHP_A - RSHP_{req} \qquad \text{eq 6.1}$$

Where:

$\Delta RSHP$	-	Excess rotor shaft horsepower
$RSHP_A$	-	Rotor shaft horsepower available
$RSHP_{req}$	-	Rotor shaft horsepower required.

The power required to change the potential energy of a given weight, W, at a certain rate is:

$$P_{\Delta PE} = W\left(\frac{dh}{dt}\right) \qquad \text{eq 6.2}$$

Where:

$P_{\Delta PE}$	-	Power required to change potential energy
W	-	Weight
h	-	Height
t	-	Time.

This power is the power required to hoist the weight at a given rate. The specific climb rate corresponding to the excess rotor shaft horsepower in a hover can be calculated:

$$V' = \frac{\Delta RSHP(33{,}000)}{W} \qquad \textit{eq 6.3}$$

Where:

V'	- Predicted rate of climb
ΔRSHP	- Excess rotor shaft horsepower
W	- Weight.

Or, in terms of engine shaft horsepower:

$$V' = \frac{\Delta ESHP\left(\eta_m\right)(33{,}000)}{W} \qquad \textit{eq 6.4}$$

Where:

V'	- Predicted rate of climb (ft/min)
ΔESHP	- Excess engine shaft horsepower
W	- Weight
η_m	- Mechanical efficiency.

The predicted rate of climb, V', is the rate produced under the following conditions:

1. If the main rotor induced power is the same in the climb as in hover.
2. If the main rotor profile power is the same in the climb as in hover.
3. If there is no loss in transmitting the excess rotor shaft horsepower to the air.

None of the above conditions exist, and as a consequence the predicted rate of climb, V', is not a realistic estimation of actual climb performance. However, corrections can be determined and applied to the predicted rate of climb, allowing closer estimation of true climb performance. To understand better what effects climb performance, the rotor in axial flight is examined by momentum and blade element analyses.

6.3.3 Momentum Analysis

To determine the approximate effect of vertical velocity on the induced power of the main rotor, simple momentum analysis is used (recall the restrictive simplifying assumptions discussed in Chapter 5). An expression for the thrust produced by the actuator disc, T, in the presence of vertical velocity is:

$$F = \frac{dm}{dt} \Delta V \qquad \textit{eq 6.5}$$

Where:

F	- Force
m	- mass
ΔV	- Change in velocity
t	- Time.

And:

$$T = \rho\, A_D \left(V_V + v_{i_v}\right)\left(2 v_{i_v}\right) \qquad \textit{eq 6.6}$$

Where:

T	- Thrust (produced by the actuator disc)
ρ	- Density
A_D	- Rotor disc area
V_V	- Actual rate of climb
v_{i_v}	- Induced velocity in vertical climb.

Where $(V_V + v_{i_v})$ is the resultant flow rate through A_D or, rearranged:

$$\frac{T}{2\,\rho\, A_D} = v_{i_v}\left(V_V + v_{i_v}\right) \qquad \textit{eq 6.7}$$

Where:

T	- Thrust produced by the actuator disc
ρ	- Density

A_D - Rotor disc area

V_V - Actual rate of climb

v_{i_V} - Induced velocity in vertical climb.

Once the rate of climb is stabilized (vertical acceleration is zero), and assuming parasite drag to be zero (a restrictive assumption), T = W and $T/(2\rho A_D) = W/(2\rho A_D) = v_i^2$, where v_i is the induced velocity in hover. The above shows that the induced velocities in hover and climb are related by the rate of climb:

$$v_i^2 = v_{i_v}^2 + v_{i_v} V_V \qquad \textit{eq 6.8}$$

Or:

$$v_{i_V} = \sqrt{\left(\frac{V_V}{2}\right)^2 + v_i^2} - \frac{V_V}{2} \qquad \textit{eq 6.9}$$

Where:

v_i - Induced velocity at hover

v_{i_V} - Induced velocity in vertical climb

V_V - Actual rate of climb.

The resulting decrease in induced velocity in the presence of vertical velocity is obvious from the basic momentum equation $F = (dm/dt)(\Delta V)$. Because a natural flow rate through the disc is established due to the vertical motion of the disc, less change in velocity is imparted to the air to produce a given thrust (Figures 6.1 and 6.2).

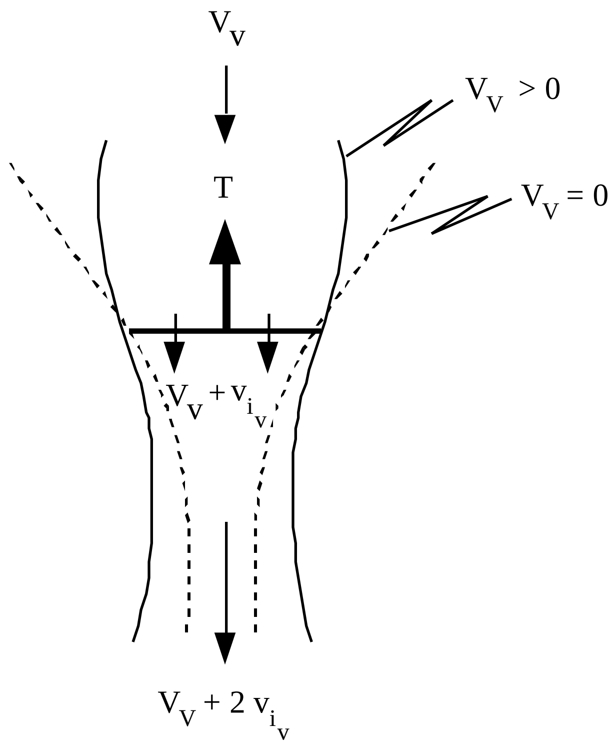

Figure 6.1
The Actuator Disc in Axial Flight

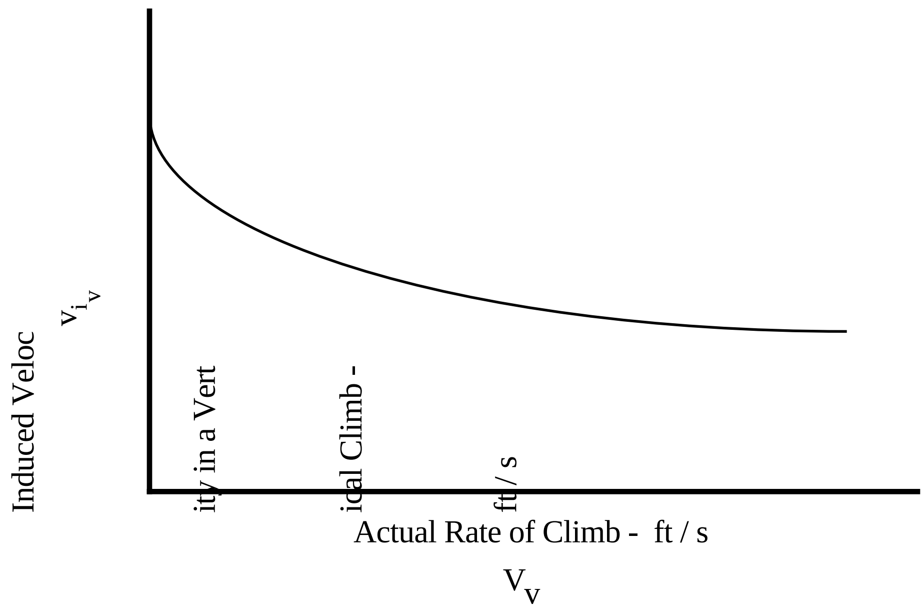

Figure 6.2
Variation of Induced Velocity with Vertical Velocity

6.3.4 Vertical Climb Correction Factor

The power available to the main rotor is selected by the pilot and can be expressed as: $P_A = P_{req} + \Delta P$ or $P_A = P_p + P_0 + P_i + \Delta P$. The time rate of change of potential energy is dictated by the excess power. Considering a helicopter with a certain amount of power available:

$$P_{A_h} = P_{A_v} \qquad \textit{eq 6.10}$$

$$P_{p_h} + P_{0_h} + P_{i_h} + \Delta P_h = P_{p_v} + P_{0_v} + P_{i_v} + \Delta P_v \qquad \textit{eq 6.11}$$

Assuming: $P_{p_h} = P_{p_v}$ and $P_{0_h} = P_{0_v}$

Then: $P_{i_h} + \Delta P_h = P_{i_v} + \Delta P_v$

Where:

P_{A_h}	- Power available at hover
P_{A_v}	- Power available in a vertical climb
P_{p_h}	- Parasite power at hover
P_{0_h}	- Profile power at hover
P_{i_h}	- Induced power at hover
ΔP_h	- Excess power at hover
P_{p_v}	- Parasite power in a vertical climb
P_{0_v}	- Profile power in a vertical climb
P_{i_v}	- Induced power in a vertical climb
ΔP_v	- Excess power in a vertical climb.

Expressing each excess power in terms of the vertical velocity that could be produced, and the induced power in terms of the induced velocity, then for a stabilized climb (T = W):

$$W\left(v_i + V'\right) = W\left(v_{i_v} + V_v\right) \qquad \textit{eq 6.12}$$

Where:

W	- Weight
v_i	- Induced velocity at hover
V'	- Predicted rate of climb
v_{i_v}	- Induced velocity in vertical climb
V_V	- Actual rate of climb.

Recall that v_i, v_{i_V} and V_V are related:

$$v_i^2 = v_{i_v}^2 + v_{i_v} V_V \qquad \textit{eq 6.8}$$

Solving the two equations (6.8 and 6.12) to eliminate v_{i_v}, which is unknown, results in the climb correction factor:

$$\frac{V_v}{V'} = 1 + \frac{1}{\frac{V'}{v_i} + 1} \qquad \textit{eq 6.13}$$

Where:

V_V	- Actual rate of climb
V'	- Predicted rate of climb
v_i	- Induced velocity at a hover.

The quantities on the right side of the equation are given by:

$$v_i = \sqrt{\frac{W}{2\rho A_D}} \qquad \textit{eq 6.14}$$

Where:

v_i	- Induced velocity in hover
ρ	- Density
A_D	- Rotor disc area
W	- Weight.

And:

$$V' = \frac{\Delta ESHP(\eta_m)(33{,}000)}{W} \qquad \textit{eq 6.4}$$

The desired value is V_V, the actual rate of climb. A plot of the climb correction factor is shown in Figure 6.3.

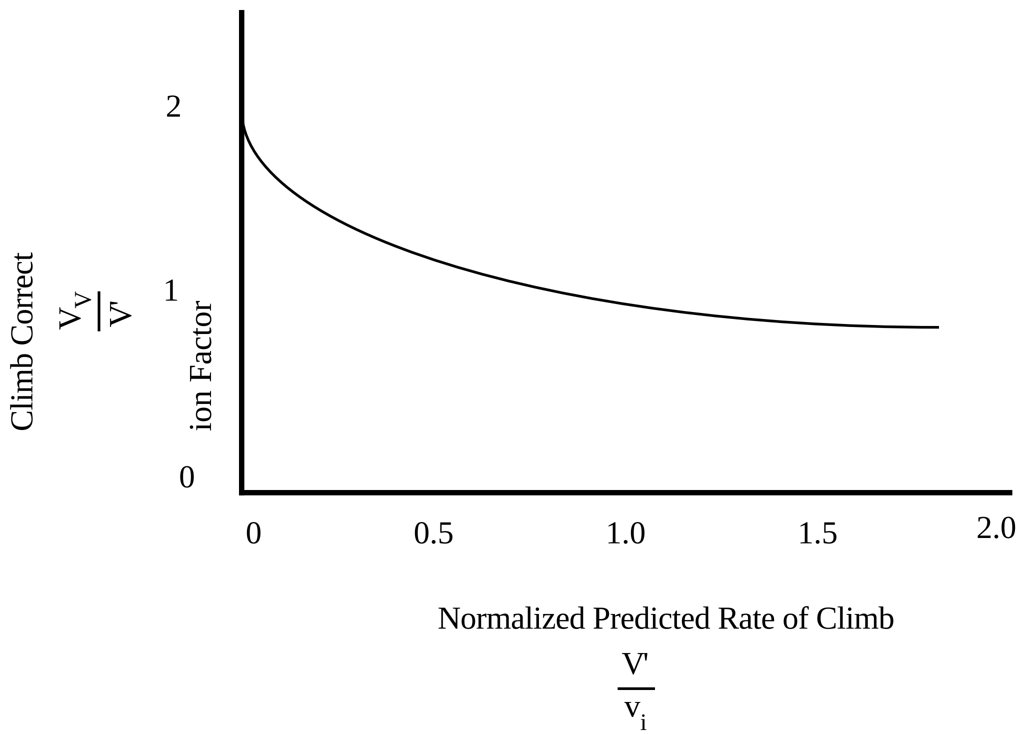

Figure 6.3
Vertical Climb Correction Factor

The vertical climb correction factor shown in Figure 6.3 is computed from ideal momentum theory and is therefore the “ideal” curve. The “ideal” curve does not reflect the effects of parasite drag, nonuniform induced velocity, and losses. Because of the unpredictable parasite drag effects at the high rates of climb, extrapolation beyond the maximum observed rate of climb is not recommended. The effects of parasite drag on the climb correction factor is shown in Figure 6.4.

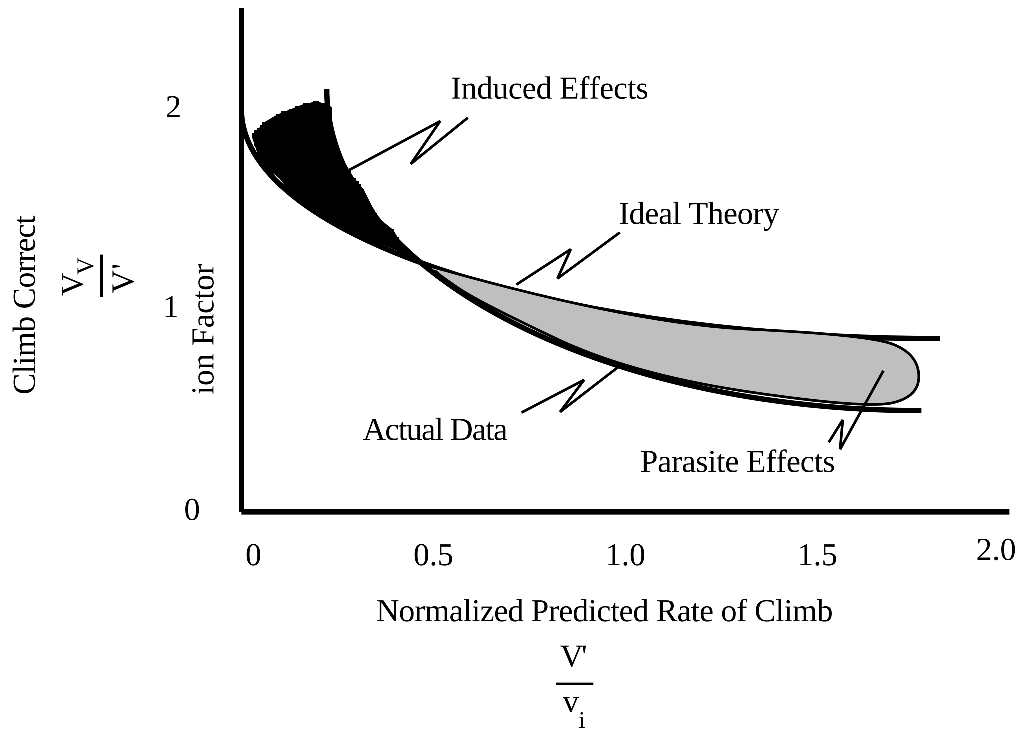

Figure 6.4
Parasite Drag Effects on Vertical Climb Correction Factor

The induced power required to produce a certain thrust is less in climbing flight than in hover. Therefore, for a given engine power there is more excess power to produce the change in potential energy in a vertical climb than in a hover. The rate of climb calculated from hover excess power (the predicted rate of climb, V') is corrected for the change in induced power by multiplying V' by the climb correction factor, V_V/V', obtained from Figure 6.3. Typical variations of both the actual and predicted rates of climb are shown versus ΔESHP in Figure 6.5. V' is less than V_V and is essentially a linear function of ΔESHP if the variation of η_m is small.

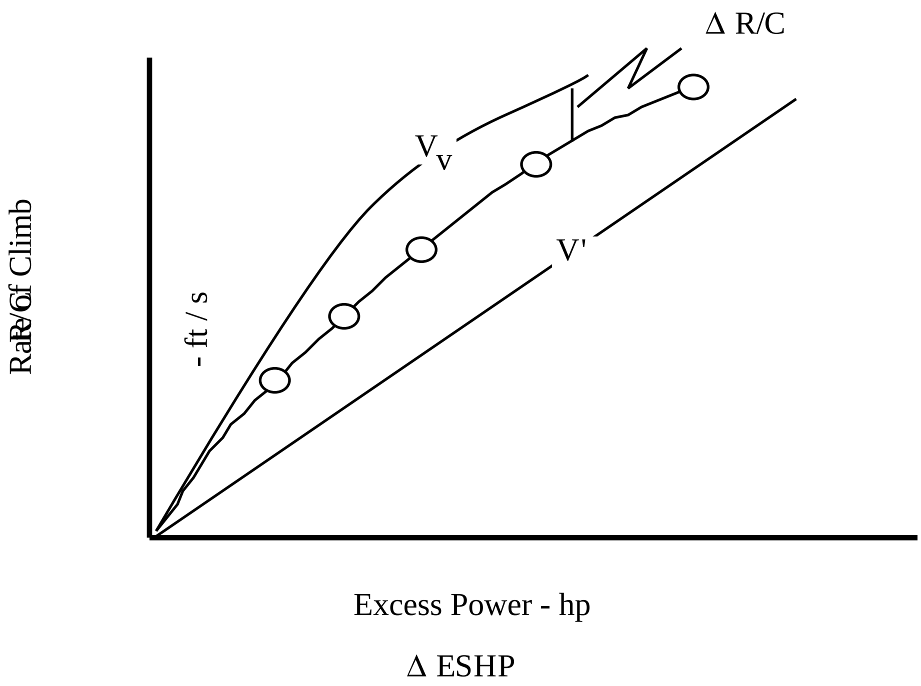

Figure 6.5
Variation of Rate of Climb with Excess Power

6.3.5 Parasite Drag

The data points shown in Figure 6.5 illustrate the effects of parasite drag on vertical climb performance. Parasite drag and Mach effects cause a degradation in climb performance compared to the corrected estimate obtained by using the energy method and the computed vertical climb correction factor. The difference in rate of climb shown in Figure 6.5 is an indication of the unpredictable parasite drag present at relatively high rates of climb. The computed vertical climb correction factor is theoretically valid for any vertically rising aircraft but in practice it is overly optimistic at rates of climb where parasite drag is a factor.

6.3.6 Blade Element Analysis

A typical blade element in hover is shown in Figure 6.6(a). If the blade pitch is increased from that required to hover, the blade section angle of attack is increased resulting in a larger aerodynamic reaction, increased lift, and increased induced velocity (Figure 6.6(b)). The unbalanced vertical force will cause the helicopter to accelerate upward, producing a vertical velocity which is added vectorially to the velocity due to rotation and the induced velocity to determine the element angle of attack. Thus, as the vertical velocity increases, the angle of attack decreases, decreasing the excess thrust and the acceleration. The vertical velocity will continue to increase, decreasing the angle of attack, until the excess thrust becomes zero and the rotor is in a stabilized climb, as shown in Figure 6.6(c). Momentum analysis shows that an increase in velocity of flow through the disc results in a given lift being produced with a smaller induced velocity. Although the decrease in the induced velocity results in a reduction in the induced power required, the effect of the climb velocity is to tip the lift vector. The tilt of the lift vector increases the force component in the plane of rotation thus increasing the total power required in the climb.

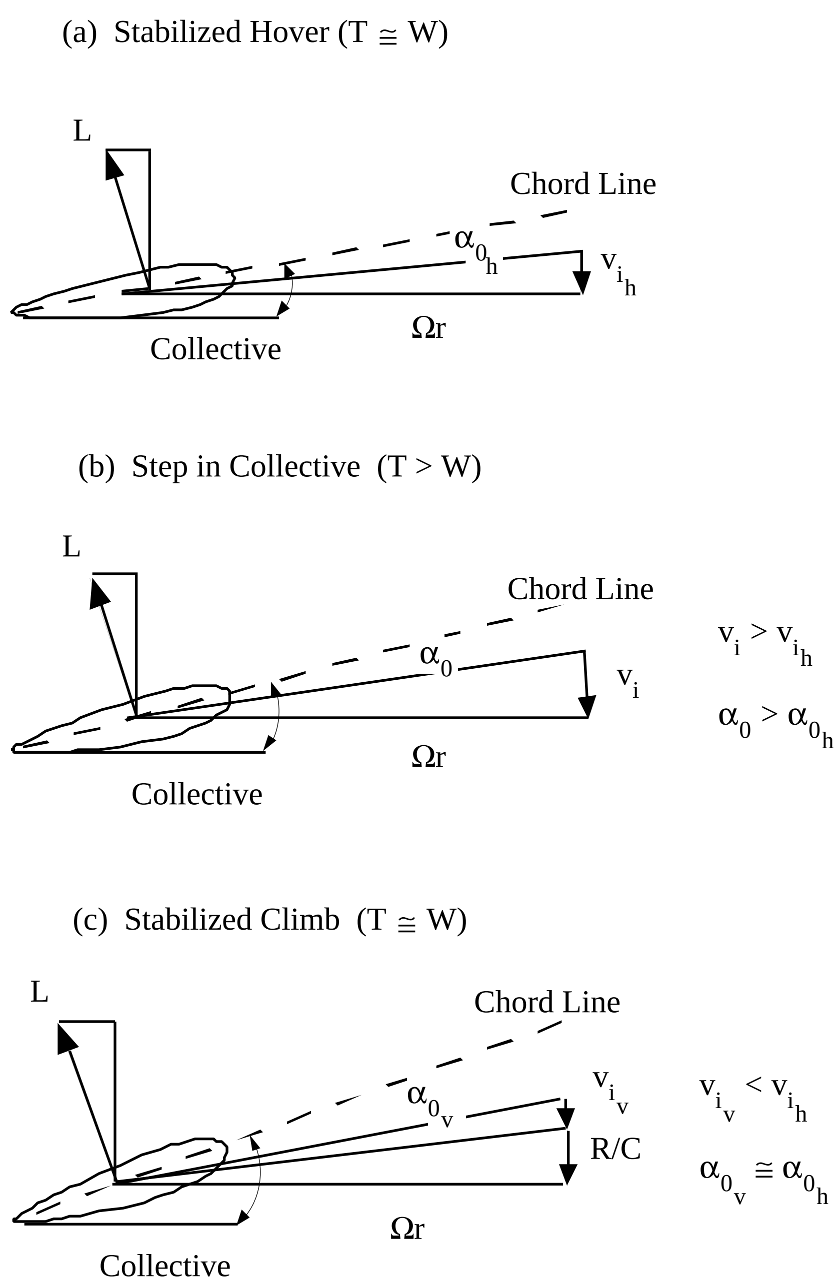

Figure 6.6
Blade Element in Hover and Vertical Climb

6.4 TEST METHODS AND TECHNIQUES

6.4.1 General

The objectives of vertical climb testing are to determine test day vertical climb performance and to compute a vertical climb correction factor. Vertical climbs are conducted using several flight test techniques discussed below. The data are presented as a variation of climb correction factor with the variation of normalized predicted rate of climb (Figure 6.3). After obtaining this climb correction factor relationship from flight test, vertical climb rates can be predicted for any combination of aircraft gross weight, rotor speed, atmospheric parameters or power margins. Vertical climb rate predictions are valid for the range of flight test conditions evaluated in building the climb correction factor model. Aircraft configuration is considered when grouping sets of data.

6.4.2 Ground Referenced

The most common method of accomplishing vertical climb performance tests is the ground referenced method. This method requires good ground references which are visible from the cockpit even when climbing rapidly and at altitudes greater than 1000 feet above ground level. In most situations a set of intersecting runways is adequate. Other long straight ground features can be used, such as roads.

The aircraft is stabilized first at an OGE hover (height as determined in hover performance tests) and data are collected. The power margin is determined (power available less the power required to hover OGE). Several vertical climbs are accomplished at increasingly larger excess power values until the limit power available is reached. Following the power application, the pilot must use all available attitude and ground reference cues to ensure the quality of the vertical climb. Small variations in horizontal velocity (as little as 3 knots) contaminate the data. Heading control is required. Once the vertical climb has stabilized, as determined by reference to a sensitive vertical accelerometer or a vertical speed indicator, data are recorded. Time history altitude information is used generally, but time to climb through an altitude band can be substituted when an automatic data collection system is not available. Once the data has been collected, the pilot increases forward speed and flies the aircraft back to the initial hover point for succeeding data points. Some aircraft may require ballasting between data points to compensate for fuel burned or to expand the gross weight data band.

A true vertical climb is required to obtain useful data. Small horizontal velocity will cause data scatter sufficient to complicate the data analysis and optimistically bias the predicated vertical climb capability of the helicopter. Good vertical climbs require intense concentration and often high pilot workload. For these reasons, don't expect the pilot to accomplish other data collection tasks during these maneuvers. Ground observers and space positioning systems are used to verify the quality of the vertical climb.

Wind variations at different altitudes above the ground are difficult to monitor. Often a steady wind is encountered just above the thermal inversion layer (a few hundred feet) which can contaminate the data of otherwise good vertical climbs. Streamers have been installed at approximately 50 ft. spacing along a balloon cable to monitor these winds. Another method involves releasing a balloon every few minutes during the tests and observing the ascent. Prior to increased environmental concerns, automobile tires were burned to produce a vertical column of smoke. Regardless of the method, consider wind effects. Generally, it is agreed that winds less than 3 knots are required for contract/specification compliance data.

6.4.2.1 DATA REQUIRED

OGE Hover:

T_o, H_{P_o}, N_R, Q, FC, configuration, wind data

Vertical Climb:

T_o, N_R, Q, FC, configuration, wind data, time history of climb

Where:

T_o	- Observed temperature
H_{P_o}	- Observed pressure altitude
N_R	- Main rotor speed
Q	- Engine torque
FC	- Fuel counts.

6.4.2.2 TEST CRITERIA

1. Vertical climb.
2. Calm winds.
3. Constant power setting.
4. Constant rotor speed.
5. Stabilized climb rate.

6.4.2.3 DATA REQUIREMENTS

OGE Hover:

1. Stabilize 15 seconds prior to recording data.
2. $N_R \pm 0.5$ rpm.
3. OGE hover height ± 3 ft.

Vertical Climb:

1. Stable vertical climb rate or zero vertical acceleration.
2. $N_R \pm 1$ rpm.
3. Drift plus wind speed ± 3 knots.
4. Heading tolerance ± 2 degrees.

6.4.2.4 SAFETY CONSIDERATIONS/RISK MANAGEMENT

The execution of these tests is more hazardous than most performance tests. OGE hover conditions present the pilot with a difficult recovery if the engine or engines should fail. During the hover as well as the climb, the aircraft is operated in the critical region of the height velocity diagram. Large, rapid power changes can cause compressor surge and stall. The pilot must be aware of the best possible forced landing area at all times. Since these tests are conducted in and around airport traffic areas, be aware of the position of other aircraft to ensure safe separation. Always fly out of the climb as opposed to descending vertically back to the start point to preclude entering vortex ring state.

6.4.3 Low Airspeed System

Another method of conducting vertical climb performance tests involves the use of a low airspeed system. Many types of devices exist to provide the pilot with longitudinal and lateral speed cues to conduct air mass referenced vertical climbs. Air referenced tests can be conducted when the surface wind is over 3 knots and at a safer altitude. The capability to conduct vertical climbs at altitude also allows expansion of the generalized data range without ballasting. Gross weight change through ballasting and the expanded atmospheric conditions available at altitude provides a climb correction factor determination for most mission related situations.

The air referenced test is conducted much like the ground reference test sequence. An OGE hover is established using the air mass referenced low airspeed system, data are collected and the available power variation is determined. Several vertical climbs are conducted using increasingly larger power settings, until maximum allowable power is reached. Following each climb, make a descent back to the original hover altitude to restart the sequence.

The quality of the data is dependent on the pilot's ability to produce consistent vertical climb conditions. In these tests, the low airspeed system must be installed and calibrated properly to provide true horizontal velocity. It is difficult to install a system which provides accurate data at various power settings. In this test the pilot integrates outside attitude references with the low airspeed system indications. It may require several practice data points to learn the proper division of attention and cross-check. Poor data points must be aborted and reflown to reduce data scatter. The sequence may be repeated at several test altitudes to enlarge the test day band.

6.4.3.1 DATA REQUIRED

OGE Hover:

T_o, H_{P_o}, N_R, Q, FC, configuration.

Vertical Climb:

T_o, N_R, Q, FC, configuration, time history of climb.

6.4.3.2 TEST CRITERIA

1. Vertical climb.
2. Smooth air.
3. Constant power setting.
4. Constant rotor speed.
5. Stabilized climb rate.

6.4.3.3 DATA REQUIREMENTS

OGE Hover:

1. Stabilize 15 seconds prior to recording data.
2. $N_R \pm 0.5$ rpm.
3. OGE hover height ± 100 ft of target altitude.

Vertical Climb:

1. Stable vertical climb rate or zero vertical acceleration.
2. $N_R \pm 1$ rpm.
3. Horizontal velocity ± 3 knots.
4. Heading tolerance ± 2 degrees.

6.4.3.4 SAFETY CONSIDERATIONS/RISK MANAGEMENT

Conducting the tests at altitude provides the pilot with additional options in the event of engine malfunctions. Conduct OGE hover and climbs over suitable forced landing areas and preferably oriented into the prevailing wind. Stay clear of other air traffic. Use an area chase aircraft to provide inflight separation from other aircraft. Review vortex ring state recovery procedures.

6.5 DATA REDUCTION

6.5.1 General

The objectives of vertical climb performance testing are to determine test day vertical climb performance and to compute a vertical climb correction factor. The test involves conducting a series of vertical climbs from an OGE hover using increments of power above that required to hover OGE. Time histories of altitude changes during stabilized climbs are recorded. Test day data are compared to mathematically predicted climb performance and a vertical climb correction factor is determined. This correction factor allows the prediction of vertical climb performance at any referred gross weight and altitude.

USNTPS uses a computer data reduction routine for helicopter vertical climb performance data reduction. This section presents a manual method of vertical climb performance data reduction as well as an introduction to the computer data reduction routine.

Whether vertical climb tests are performed using ground references or using an air mass reference system, the data reduction schemes are very similar. Gross weight and actual engine power must be calculated for each data point including the initial OGE hover condition.

All observed data must be corrected for instrument and position errors as appropriate.

6.5.2 Manual Data Reduction

The following equations are used in the manual reduction of vertical climb performance data:

$$H_{P_c} = H_{P_o} + \Delta H_{P_{ic}} + \Delta H_{pos} \qquad \textit{eq 6.15}$$

Where:

H_{P_c}	- Calibrated pressure altitude
H_{P_o}	- Observed pressure altitude
$\Delta H_{P_{ic}}$	- Altimeter instrument correction
Δh_{pos}	- Altimeter position error.

$$T_a = T_o + \Delta T_{ic} \qquad \textit{eq 6.16}$$

Where:

T_a	- Ambient temperature
T_o	- Observed temperature
ΔT_{ic}	- Temperature instrument correction.

$$\sigma_T = \frac{\delta}{\theta} = \frac{\delta}{\left(\frac{T_a}{T_{ssl}}\right)}$$ *eq 6.17*

Where:

σ_T	- Test density ratio
δ	- Pressure ratio
θ	- Temperature ratio
T_a	- Ambient temperature
T_{ssl}	- Standard sea level temperature, 288.15 °K.

$$ESHP_h = K_Q (Q_h)(N_{R_h})$$ *eq 6.18*

Where:

$ESHP_h$	- Engine shaft horsepower in a hover
K_Q	- Engine torque constant unique for each aircraft
Q_h	- Engine torque in a hover
N_{R_h}	- Main rotor speed in a hover.

$$ESHP_{climb} = K_Q(Q_{climb})(N_{R_{climb}})$$ *eq 6.19*

Where:

$ESHP_{climb}$	- Engine shaft horsepower in climb
K_Q	- Engine torque constant unique for each aircraft
Q_{climb}	- Engine torque in a climb
$N_{R_{climb}}$	- Main rotor speed in a climb.

$$\Delta ESHP = ESHP_{climb} - ESHP_h$$ *eq 6.20*

Where:

$\Delta ESHP$	- Excess engine shaft horsepower
$ESHP_{climb}$	- Engine shaft horsepower in climb
$ESHP_h$	- Engine shaft horsepower in a hover.

$$GW = ESGW - FU$$ *eq 6.21*

Where:

GW	- Gross weight
ESGW	- Engine start gross weight
FU	- Fuel used.

$$V' = \frac{(33{,}000)(\Delta ESHP)}{GW} \qquad \textit{eq 6.22}$$

Where:

V'	- Predicted rate of climb
$\Delta ESHP$	- Excess engine shaft horsepower
GW	- Gross weight.

$$K_{H_P} = \frac{T_a + 273.15}{T_S + 273.15} \qquad \textit{eq 6.23}$$

Where:

K_{Hp}	- Temperature correction factor
T_a	- Ambient temperature
T_S	- Standard temperature (at that altitude).

$$V_v = K_{Hp}\left(\frac{dh}{dt}\right) \qquad \textit{eq 6.24}$$

Where:

V_V	- Actual rate of climb
K_{Hp}	- Temperature correction factor
h	- Height
t	- Time.

$$v_i = 60 \sqrt{\frac{GW}{2\,\rho\,A_D}}$$ *eq 6.25*

Where:

v_i	- Induced velocity at hover
GW	- Gross weight
ρ	- Density
A_D	- Rotor disc area.

From observed hover and vertical climb performance data, compute V'/v_i and V_V/V' as follows:

Parameter	Notation	Formula	Units	Remarks
(1) Observed pressure altitude	H_{P_o}		ft	
(2) Calibrated pressure altitude	H_{P_c}	eq 6.15	ft	Instrument & position corrections
(3) Observed temperature	T_o		°C	
(4) Ambient temperature	T_a	eq 6.16	°C	Instrument correction, ignore recovery and Mach correction
(5) Pressure ratio	δ			From tables at step (2)
(6) Test density ratio	σ_T	eq 6.17		
(7) Hover main rotor speed	N_{R_h}		%	(or appropriate units)
(8) Standard main rotor speed	N_{R_S}		%	(or appropriate units)
(9) Climb main rotor speed	$N_{R_{climb}}$		%	(or appropriate units)

(10) Engine torque at a hover	Q_h		%, psi	(or appropriate units)
(11) Hover ESHP	$ESHP_h$	eq 6.18	hp	
(12) Engine torque in a climb	Q_{climb}		%, psi	(or appropriate units)
(13) Climb ESHP	$ESHP_{climb}$	eq 6.19	hp	
(14) Engine power for climb	$\Delta ESHP$	eq 6.20	hp	
(15) Fuel counts	FC		cts	
(16) Fuel used	FU	FC (gal/ct) (lb/gal)	lb	ct/gal - lab calibration lb/gal - fuel density
(17) Gross weight	GW	eq 6.21	lb	
(18) Predicted rate of climb	V'	eq 6.22	ft/min	
(19) Slope of test day time history	dh/dt		ft/min	
(20) Standard temperature	T_S		°C	From tables or calculated for (2)
(21) Temperature correction factor	K_{Hp}	eq 6.23		
(22) Actual rate of climb	V_V	eq 6.24	ft/min	
(23) Induced velocity in hover	v_i	eq 6.25	ft/min	
(24) Normalized predicted rate of climb	V'/v_i	(18)/(23)		
(25) Climb correction factor	V_V/V'	(22)/(18)		

Plot V_V/V' (climb correction factor) as a function of V'/v_i (normalized predicted rate of climb) (Figure 6.7).

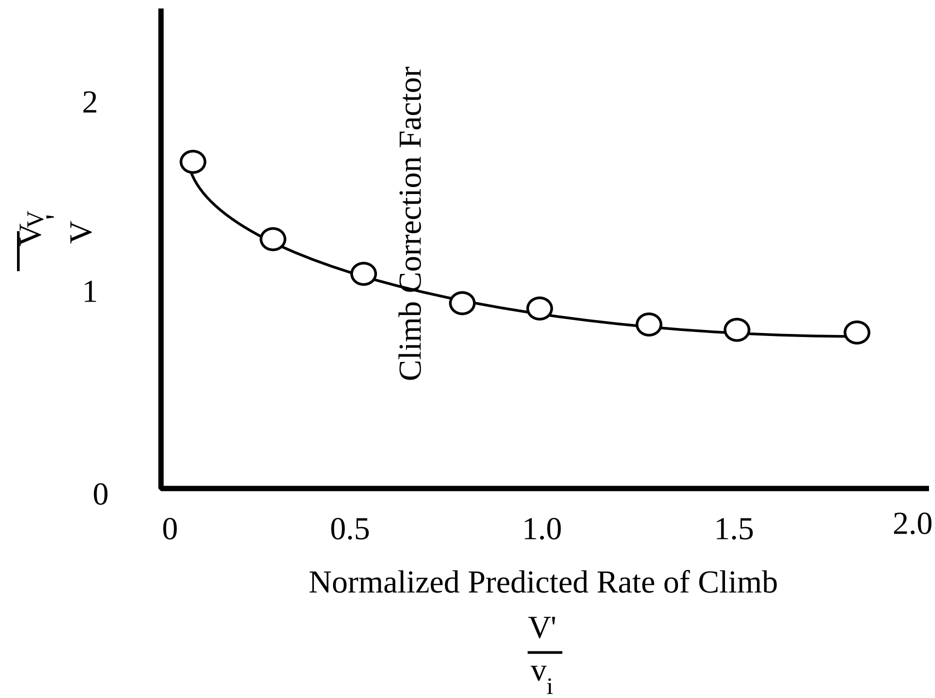

Figure 6.7
Vertical Climb Correction Factor Data

6.5.3 USNTPS Computer Data Reduction

This section contains a brief summary of the assumptions and logic used in the USNTPS helicopter vertical climb computer data reduction program.

Vertical climb performance can be estimated using the following equation:

$$V' = \frac{\left(ESHP_A - ESHP_h\right)(33{,}000)}{W} \qquad \textit{eq 6.26}$$

Where:

V'	- Predicted rate of climb
$ESHP_A$	- Engine shaft horsepower available
$ESHP_h$	- Engine shaft horsepower required to hover
W	- Weight.

Flight test data provides the correction factor and is expressed as V_V/V'.

Apply the climb correction factor to any estimated (predicted) climb rate:

$$V_V = \left(\frac{V_V}{V'}\right) V' \qquad \textit{eq 6.27}$$

Where:

V_V	- Actual rate of climb
V'	- Predicted rate of climb
V_V/V'	- Climb correction factor.

The results of the HOVER performance program are input to the vertical climb program in the form:

$$ESHP_{ref} = A_0 + A_1 W_{ref}^{\frac{3}{2}} \qquad \textit{eq 6.28}$$

Where:

$ESHP_{ref}$	- Referred engine shaft horsepower
A_0, A_1	- Constants
W_{ref}	- Referred weight.

The following equations are used in the computerized calculations:

$$\theta = \frac{T_a}{T_{ssl}} = \frac{OAT + 273.15}{288.15} \qquad \textit{eq 6.29}$$

Where:

θ	- Temperature ratio
T_a	- Ambient temperature
T_{ssl}	- Standard sea level temperature, 288.15°K
OAT	- Outside air temperature (total temperature).

$$\delta = \left[1 - \left(\frac{\lambda_{ssl} \; H_P}{T_{ssl}}\right)\right]^{\frac{g_{ssl}}{g_c \, \lambda_{ssl} \; R}} \qquad \textit{eq 6.30}$$

Where:

δ	- Pressure ratio
λ_{ssl}	- Standard sea level lapse rate, 0.00198 °K/ft.
T_{ssl}	- Standard sea level temperature, 288.15 °K
H_P	- Pressure altitude
g_{ssl}	- Standard sea level gravity, 32.17 ft/s^2
g_c	- 32.17 lbm/slug
R	- Engineering gas constant for air, 96.03 ft - lb/lbm- °K.

$$\theta = \frac{\delta}{\theta} \qquad \textit{eq 6.31}$$

Where:

σ	- Density ratio
δ	- Pressure ratio
θ	- Temperature ratio.

$$ESHP_{climb} = K_Q \, (Q_{climb}) \, (N_{R_{climb}}) \qquad \textit{eq 6.19}$$

$$ESHP_h = K_Q \, (Q_h) \, (N_{R_h}) \qquad \textit{eq 6.18}$$

$$V' = \frac{(ESHP_{climb} - ESHP_h)\ (33{,}000)}{W_T}$$

eq 6.32

Where:

V'	- Predicted rate of climb
$ESHP_{climb}$	- Engine shaft horsepower in a climb
$ESHP_h$	- Engine shaft horsepower at a hover
W_T	- Test weight.

$$\frac{T_T}{T_S} = \frac{OAT + 273.15}{T_{ssl} - \lambda_{ssl}\ H_{P_h}}$$

eq 6.33

Where:

T_T	- Test temperature
T_S	- Standard temperature (at that altitude)
OAT	- Outside air temperature (total temperature), °C
T_{ssl}	- Standard sea level temperature, 288.15 °K
λ_{ssl}	- Standard sea level lapse rate, 0.00198 °K/ft
H_{P_h}	- Pressure altitude at a hover.

$$V_V = 60 \left(\frac{T_T}{T_S}\right) \left(\frac{\Delta H_P}{\Delta t}\right)$$

eq 6.34

Where:

V_V	- Actual rate of climb
T_T	- Test temperature
T_S	- Standard temperature (at that altitude)
$\Delta H_P/\Delta t$	- Flight test rate of climb, ft/s.

$$v_i = 60\sqrt{\frac{W_T}{2\rho_{ssl}\,\sigma\,A_D}} \qquad \textit{eq 6.35}$$

Where:

v_i	- Induced velocity in hover
W_T	- Test weight
ρ_{ssl}	- Standard sea level density, 0.0023769 slug/ft^3
σ	- Density ratio
A_D	- Rotor disc area.

The user is prompted to provide fuel, outside air temperature, altitude, torque, rotor speed and climb rate data for each data point. Hover performance data (coefficients of referred format) are entered. Weighted least squares curve fits are applied to determine the coefficients of the theoretical equation form. Graphic representations are displayed for the generalized vertical climb performance data set (V_V/v_i versus V'/v_i) and climb correction factor (V_V/V').

The next portion of the computer program allows the user to select the desired rotor speed, weights (usually four) and power margins. Calculation can be based on standard or hot day atmospheric conditions. Power available data are input for the atmospheric conditions specified as a function of pressure altitude.

The final data are presented in a tabular form. If a printout is not provided for a given weight, power margin, or power available combination, no rate of climb exists for these conditions.

6.6 DATA ANALYSIS

6.6.1 General

Flight test vertical climb performance data are reduced to normalized, generalized terms. Test day data are then compared to mathematically predicted vertical climb performance and a climb correction factor is determined. Climb correction factor information is used to correct predicted vertical climb performance for any aircraft gross weight and atmospheric condition of interest. Do not extrapolate data. Valid performance predictions can be made only for the range of test day data used in the construction of the climb correction factor model.

6.6.2 Specific Conditions

To analyze vertical climb performance data, specific aircraft gross weight and atmospheric conditions must be related to a task or mission. The following equations are used to determine vertical climb rate predictions for aircraft gross weight, rotor speed, and atmospheric conditions of interest.

Standard day:

$$\theta = 1 - \left(\frac{\lambda_{ssl}}{T_{ssl}}\right) H_P \qquad \textit{eq 6.36}$$

Hot day:

$$\theta = \frac{312.61 - \lambda_{ssl} H_P}{T_{ssl}} \qquad \textit{eq 6.37}$$

Where:

θ	- Temperature ratio
λ_{ssl}	- Standard sea level temperature lapse rate, 0.00196 °K/ft
T_{ssl}	- Standard sea level temperature, 288.15°K
H_P	- Pressure altitude.

$$\delta = \left[1 - \left(\frac{\lambda_{ssl} \; H_P}{T_{ssl}}\right)\right]^{\frac{g_{ssl}}{g_c \lambda_{ssl} R}} \qquad \textit{eq 6.30}$$

$$\theta = \frac{\delta}{\theta} \qquad \textit{eq 6.31}$$

$$W_{ref} = \frac{W_T}{\sigma_T}\left(\frac{N_{R_S}}{N_{R_T}}\right)^2 \qquad \textit{eq 6.38}$$

Where:

W_{ref}	- Referred weight
W_T	- Test weight
σ_T	- Test density ratio
N_{R_S}	- Standard main rotor speed
N_{R_T}	- Test main rotor speed.

$$ESHP_{ref} = A_0 + A_1\, W_{ref}^{\frac{3}{2}} \qquad \textit{eq 6. 27}$$

$$ESHP_h = ESHP_{ref}\,\sigma_S\left(\frac{N_R}{N_{R_S}}\right)^3 \qquad \textit{eq 6.39}$$

Where:

$ESHP_h$	- Engine shaft horsepower required to hover
$ESHP_{ref}$	- Referred engine shaft horsepower
σ_S	- Standard density ratio
N_R	- Main rotor speed
N_{R_S}	- Standard main rotor speed.

$$V' = \frac{\left(ESHP_A - ESHP_h\right)(33{,}000)}{W} \qquad \textit{eq 6.26}$$

$$v_i = 60\sqrt{\frac{W_T}{2\rho_{ssl}\,\sigma\,A_D}} \qquad \textit{eq 6.35}$$

$$V_V = \left(\frac{V_V}{V'}\right) V' \qquad \textit{eq 6.27}$$

Use normalized vertical climb correction factor (Figure 6.7) and engine power available (Figure 4.10) to calculate specific vertical climb performance capabilities as a function of altitude as follows:

Parameter	Notation	Formula	Units	Remarks
(1) Weight	W		lb	Select representative weights as desired
(2) Pressure altitude	H_P		ft	Select a series of H_P from sea level to > anticipated hover ceiling
(3) Temperature ratio	θ	eq 6.36 or eq 6.37	ft	Compute for each H_P and atmospheric model
(4) Pressure ratio	δ	eq 6.30		Compute for each H_P and atmospheric model
(5) Density ratio	σ	eq 6.31		Compute for each H_P and atmospheric model
(6) Rotor Speed	N_R		%	Select a main rotor speed of interest
(7) Referred Weight	W_{ref}	eq 6.38	lb	Compute for each H_P and atmospheric model

(8) Referred ESHP	$ESHP_{ref}$	eq 6.28	hp	Compute referred ESHP using constants obtained in Figure 5.14
(9) Power required to hover	$ESHP_h$	eq 6.39	hp	Compute for each H_P and atmospheric model
(10) Power available	$ESHP_A$		hp	Determine the engine power available at each H_P and atmospheric model (Figure 4.10)
(11) Predicted rate of climb	V'	eq 6.26	ft/min	Compute for each H_P and atmospheric model
(12) Induced velocity in hover	v_i	eq 6.35	ft/min	Compute for each H_P and atmospheric model
(13) Normalized predicted rate of climb	V'/v_i	(11)/(12)		Compute for each H_P and atmospheric model
(14) Climb correction factor	V_V/V'			Determine for each normalized predicted rate of climb (13) from Figure 6.7
(15) Actual rate of climb	V_V	eq 6.27	ft/min	Compute for each H_P and atmospheric model

Plot vertical rate of climb for each gross weight at the pressure altitudes computed above. Each gross weight will define a unique line as shown in Figure 6.8 below.

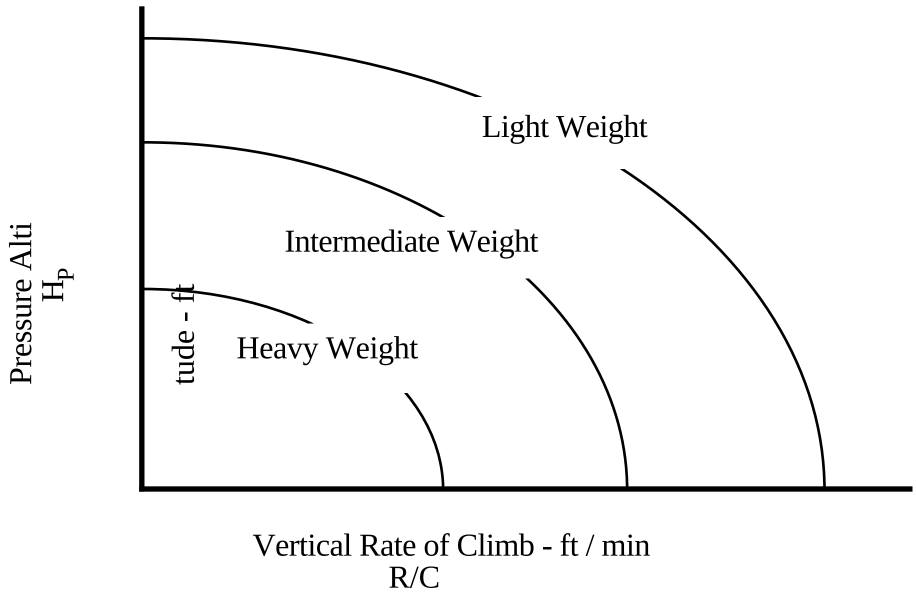

Figure 6.8
Vertical Climb Performance for Several Gross Weights

Plot vertical rate of climb for each atmospheric condition (standard day/hot day) computed above. An example is shown in Figure 6.9. This figure illustrates the effects of atmospheric conditions on the vertical climb performance of the helicopter.

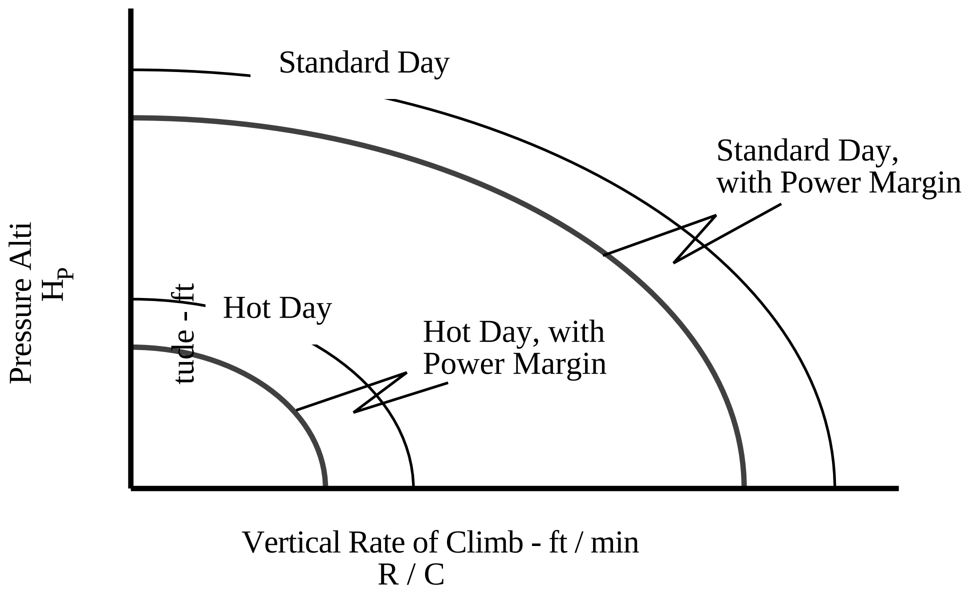

Figure 6.9
Vertical Climb Performance for Different Atmospheric Conditions

Figures 6.8 and 6.9 will be modified if a power margin is used in the data reduction. A power margin, for instance 5%, can be applied to flight test results to account for engine performance degradation encountered in field conditions. The reductions in power available can be caused by dirty compressor sections, worn compressor blades or many other reasons. To show the effects of nonspecification power available, apply the following equation to item 10 of the data analysis scheme depicted above. The remaining steps of the process are unchanged.

$$ESHP_{A_2} = ESHP_{A_1}\left(\frac{100 - \text{margin}}{100}\right)$$

eq 6.40

Where:

$ESHP_{A_2}$	- Engine shaft horsepower available after the power margin is applied
$ESHP_{A_1}$	- Engine shaft horsepower available before the power margin is applied
margin	- Arbitrary power margin (in percent).

The effects of power margins is presented in Figure 6.10:

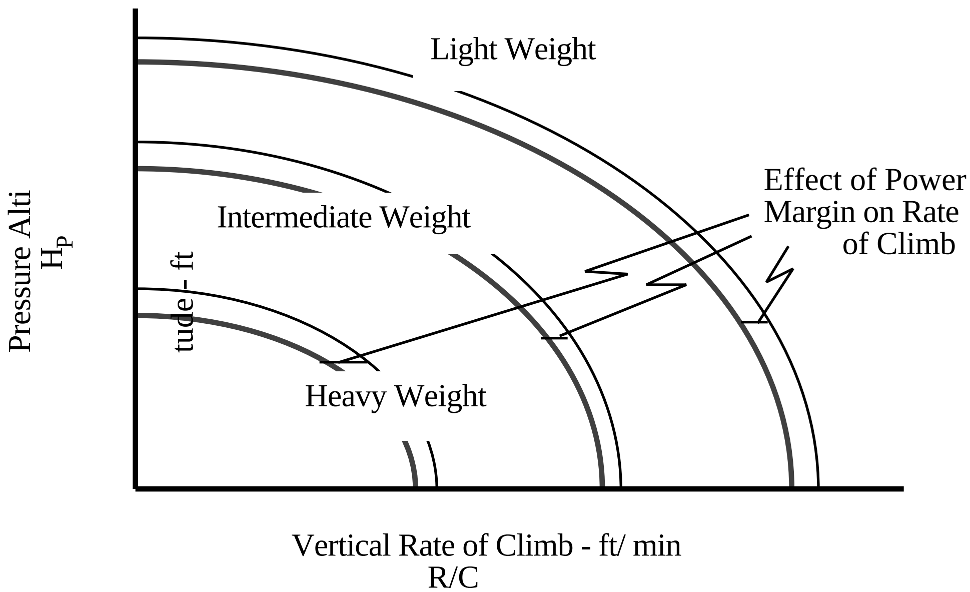

Figure 6.10
Vertical Climb Performance with a Power Margin

The effects of gross weight, atmospheric conditions and power margins are available to establish contract and/or specification compliance.

6.7 MISSION SUITABILITY

6.7.1 General

The requirement for establishing the vertical rate of climb capabilities of a helicopter is becoming more and more important during performance evaluations. Satisfactory use of the helicopter, whether military or commercial, requires some form of a vertical climb. The employment of the helicopter in a tactical situation requiring confined area takeoffs or immediate "waveoffs" from a hover are examples of mission suitability requirements which dictate a satisfactory degree of vertical climb performance.

6.7.2 Pilot Effort

The flight test determination of the vertical climb performance is simple, yet very demanding. The accuracy of the results depends on the pilot's familiarity with the test aircraft. The pilot must sense horizontal displacement through changes in aircraft heading and attitude. The pilot effort required to conduct a “pure” vertical climb must be at maximum to minimize scatter in the data. The overall service suitability determination of the vertical climb performance characteristics reflect the pilot effort required during the maneuver.

6.8 SPECIFICATION COMPLIANCE

6.8.1 General

The detail specification for the model helicopter normally states a vertical climb performance guarantee in terms of a specified rate of climb. The guarantee is defined as the rate of climb obtainable at standard day sea level conditions (or a specific non-standard condition), military or takeoff power, and at a standard or mission gross weight. Some helicopters do not have a guarantee for vertical rate of climb, and if this is the case, the vertical climb performance should be evaluated in terms of service suitability. More than one gross weight is evaluated for service suitability considerations. Generally three gross weights are sufficient to describe the vertical climb performance capabilities of the aircraft: minimum operating weight, mission weight, and maximum overload weight.

6.8.2 Specification Deviation

The vertical climb guarantee is specified in zero winds conditions; however, winds of 3 knots or less are satisfactory for determining specification compliance. Wind problems are complicated by wind shear in the test altitude band. To obtain accurate results, the wind must be constant from the ground to the peak altitude used during the test.

6.9 GLOSSARY

6.9.1 Notations

A_D	Rotor disc area	ft^2
A_0, A_1	Constants	
ESGW	Engine start gross weight	lb
$ESHP_A$	Engine shaft horsepower available	hp
$ESHP_{climb}$	Engine shaft horsepower in climb	hp
$ESHP_h$	Engine shaft horsepower in a hover	hp
$ESHP_{ref}$	Referred engine shaft horsepower	hp
$\Delta ESHP$	Excess engine shaft horsepower	hp
F	Force	lb
FC	Fuel counts	cts
FU	Fuel used	lb
GW	Gross weight	lb
g_c	32.17 lbm/ slug	
g_{ssl}	Standard sea level gravity	32.17 ft/s^2
h	Height	ft
H_P	Pressure altitude	ft
H_{P_c}	Calibrated pressure altitude	ft
H_{P_o}	Observed pressure altitude	ft
H_{P_h}	Pressure altitude at a hover	ft
$\Delta H_{P_{ic}}$	Altimeter instrument correction	ft
$\Delta H_P/\Delta t$	Flight test rate of climb	ft/min
ΔH_{pos}	Altimeter position error	ft
K_{Hp}	Temperature correction factor	
K_Q	Engine torque constant (aircraft unique)	
L	Lift	
m	Mass	lbm

N_R	Main rotor speed	%, rpm
$N_{R_{climb}}$	Main rotor speed in a climb	%, rpm
N_{R_h}	Main rotor speed in a hover	%, rpm
N_{R_S}	Standard main rotor speed	%, rpm
N_{R_T}	Test main rotor speed	%, rpm
OAT	Outside air temperature (total temperature)	°C
P_A	Power available	hp
P_{A_h}	Power available at hover	hp
P_{A_v}	Power available in a vertical climb	hp
$P_{\Delta PE}$	Power required to change potential energy	hp
ΔP	Excess power	hp
ΔP_h	Excess power at hover	hp
ΔP_v	Excess power in a vertical climb	hp
P_i	Induced power	hp
P_{i_h}	Induced power at hover	hp
P_{i_v}	Induced power in a vertical climb	hp
P_0	Profile power	hp
P_{0_h}	Profile power at hover	hp
P_{0_v}	Profile power in a vertical climb	hp
P_p	Parasite power	hp
P_{p_h}	Parasite power at hover	hp
P_{p_v}	Parasite power in a vertical climb	hp
P_{req}	Power required	hp
Q	Engine torque	%, psi
Q_{climb}	Engine torque in a climb	%, psi
Q_h	Engine torque in a hover	%, psi

R	Engineering gas constant for air	96.03 ft-lb/lb_m-°K
R/C	Rate of climb	ft/min
$RSHP_A$	Rotor shaft horsepower available	hp
$RSHP_{req}$	Rotor shaft horsepower required	hp
$\Delta RSHP$	Excess rotor shaft horsepower	hp
t	Time	min, s
T	Thrust (produced by the actuator disc)	lb
T_a	Ambient temperature	°C
T_o	Observed temperature	°C
T_{ssl}	Standard sea level temperature	288.15 °K
T_S	Standard temperature (at that altitude)	°C
T_T	Test temperature	°C
ΔT_{ic}	Temperature instrument correction	°C
v_i	Induced velocity at hover	ft/min
v_{i_V}	Induced velocity in vertical climb	ft/min
V'	Predicted rate of climb	ft/min
V'/v_i	Normalized predicted rate of climb	
V_V	Actual rate of climb	ft/min
V_V/V'	Climb correction factor	
V_V/v_i	Normalized actual rate of climb	
ΔV	Change in velocity	ft/s
W	Weight	lb
W_{ref}	Referred weight	lb
W_T	Test weight	lb

6.9.2 Greek Symbols

α_0 (Alpha)	Initial angle of attack	deg
α_{0_h}	Initial angle of attack in hover	deg
α_{0_v}	Initial angle of attack in vertical climb	deg
δ (Delta)	Pressure ratio	
λ_{ssl} (Lambda)	Standard sea level lapse rate	0.00198 °K/ft
η_m (Eta)	Mechanical efficiency	
ρ (Rho)	Density	slug/ft^3
ρ_{ssl}	Standard sea level air density	0.0023769 slug/ft^3
σ (Sigma)	Density ratio	
σ_T	Test density ratio	
σ_S	Standard density ratio	
θ (Theta)	Temperature ratio	
Ωr	Blade element rotational velocity	ft/s

6.10 REFERENCES

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CHAPTER SEVEN

LEVEL FLIGHT PERFORMANCE

CHAPTER SEVEN

FIGURES

CHAPTER SEVEN

TABLES

CHAPTER SEVEN

EQUATIONS

$$T = \frac{dm}{dt}\,\Delta V$$ *eq 7.1*

$$T_h = 2\,\rho\,A_D\,v_i^2$$ *eq 7.2*

$$T_v = \rho\,A_D\,(2\,v_{i_v})\,(v_{i_v} + V_V)$$ *eq 7.3*

$$T_f = \rho\,A_D\,V_R\,(2\,v_{i_f})$$ *eq 7.4*

$$V_R^2 = V_f^2 + v_{i_f}^2 + \left[(2V_f)\left(v_{i_f}\right)(\sin a)\right]$$ *eq 7.5*

$$V_R = V_f$$ *eq 7.6*

$$P_{i_f} = \frac{T^2}{2\,\rho\,A_D\,V_R} = \frac{W^2}{2\,\rho\,A_D\,V_f}$$ *eq 7.7*

$$dP_0 = dD_0\,v_e = \tfrac{1}{2}C_{d_0}\,\rho\left(\Omega r + V_f\,\sin\,\psi\right)^3\,c\,dr$$ *eq 7.8*

$$P_0 = \frac{b}{2\pi}\iint \tfrac{1}{2}\,C_{d_0}\,\rho\left(\Omega r + V_f\,\sin\,\psi\right)^3\,c\,dr\,d\psi$$ *eq 7.9*

$$P_0 = \frac{1}{8}\,\sigma_R\,\overline{C}_{d_0}\,\rho\,A_D\,(\Omega R)^3\left(1 + 3\mu^2\right)$$ *eq 7.10*

$$P_{0_{H2}} = \frac{1}{8}\,\sigma_R\,\overline{C}_{d_0}\,\rho\,A_D\,(\Omega R)^3\left(1 + 1.65\mu^2\right)$$ *eq 7.11*

$$P_0 = \frac{1}{8}\,\sigma_R\,\overline{C}_{d_0}\,\rho\,A_D\,(\Omega R)^3\left(1 + 4.65\mu^2\right)$$ *eq 7.12*

$$D_p = \frac{1}{2} \rho \, C_{D_p} \, A_f \, V^2 \qquad \textit{eq 7.13}$$

$$P_p = \frac{1}{2} \rho \, C_{D_p} \, A_f \, V_f^3 = \frac{1}{2} \rho \, f \, V_f^3 \qquad \textit{eq 7.14}$$

$$P_p = \frac{1}{2} \rho \, f \, (\Omega R)^3 \, \mu^3 \qquad \textit{eq 7.15}$$

$$T_{TR} = \frac{Q_{MR}}{\ell_{TR}} \qquad \textit{eq 7.16}$$

$$P_{TR} = \left(\frac{T_{TR}^2}{2 \, \rho \, A_D \, V_f} \right)_{TR} + \left[\frac{1}{8} \sigma_R \, \bar{C}_{d_0} \, \rho \, A_D \, (\Omega R)^3 \left(1 + 4.65 \mu^2 \right) \right]_{TR} \qquad \textit{eq 7.17}$$

$$P_{TOTAL} = P_i + P_0 + P_p + P_m \qquad \textit{eq 7.18}$$

$$P_{MR} = \left(\frac{W^2}{2 \, \rho \, A_D \, V_f} \right)_{MR} + \left[\frac{1}{8} \sigma_R \, \bar{C}_{d_0} \, \rho \, A_D \, (\Omega R)^3 \left(1 + 4.65 \mu^2 \right) \right]_{MR} + \left(\frac{1}{2} \rho \, V_f^3 \, f \right) \qquad \textit{eq 7.19}$$

$$C_P = \frac{C_T^2}{2\mu} + \left[\frac{1}{8} \sigma_R \, \bar{C}_{d_0} \left(1 + 4.65 \mu^2 \right) \right] + \left(\frac{1}{2} C_{D_p} \, \mu^3 \right) \qquad \textit{eq 7.20}$$

$$C_T = \frac{\left(\frac{W}{\sigma_T} \right)}{\rho_{ssl} \, A_D \, (\Omega R)^2} \qquad \textit{eq 7.21}$$

$$r = -\mu \, R \, \sin \psi \qquad \textit{eq 7.22}$$

$$P_0 = \frac{1}{8} \sigma_R \, \bar{C}_{d_0} \, \rho \, A_D \, (\Omega R)^3 \left(1 + 4.65 \mu^2 - \frac{3}{8} \mu^4 \right) \qquad \textit{eq 7.23}$$

$$RSHP_{ref} = \frac{C_P \, \rho_{ssl} \, A_D \left(\Omega R_S \right)^3}{550} = \left(\frac{RSHP_T}{\sigma_T} \right) \left(\frac{N_{R_S}}{N_{R_T}} \right)^3 \qquad \textit{eq 7.24}$$

$$W_{ref} = C_T \rho_{ssl} A_D (\Omega R_S)^2 = \frac{W_T}{\sigma_T}\left(\frac{N_{R_S}}{N_{R_T}}\right)^2 \qquad \textit{eq 7.25}$$

$$V_{T_{ref}} = \mu (\Omega R_T) = V_T\left(\frac{N_{R_S}}{N_{R_T}}\right) \qquad \textit{eq 7.26}$$

$$ESHP_{ref} = \frac{ESHP_T}{\sigma_T}\left(\frac{N_{R_S}}{N_{R_T}}\right)^3 \qquad \textit{eq 7.27}$$

$$P = D_{eff} V_f \qquad \textit{eq 7.28}$$

$$H_{P_c} = H_{P_o} + \Delta H_{P_{ic}} + \Delta H_{pos} \qquad \textit{eq 7.29}$$

$$T_a = T_o + \Delta T_{ic} \qquad \textit{eq 7.30}$$

$$T_a(^\circ K) = T_a(^\circ C) + 273.15 \qquad \textit{eq 7.31}$$

$$\sigma_T = \frac{\delta}{\theta} = \frac{\delta}{\left(\frac{T_a}{T_{ssl}}\right)} \qquad \textit{eq 7.32}$$

$$ESHP_T = K_Q (Q) (N_{R_T}) \qquad \textit{eq 7.33}$$

$$W_T = ESGW - FU \qquad \textit{eq 7.34}$$

$$W_{ref} = \frac{W_T}{\sigma_T}\left(\frac{N_{R_S}}{N_{R_T}}\right)^2 \qquad \textit{eq 7.35}$$

$$\Omega R = K_{GR}\left(N_{R_T}\right)\left(\frac{2\pi}{60}\right)(R)$$ *eq 7.36*

$$C_P = \frac{P}{\rho_a\,(A_D)\,(\Omega R)^3}$$ *eq 7.37*

$$C_T = \frac{T}{\rho_a\,(A_D)\,(\Omega R)^2}$$ *eq 7.38*

$$V_c = V_o + \Delta V_{ic} + \Delta V_{pos}$$ *eq 7.39*

$$V_T = \frac{V_c}{\sqrt{\sigma}}$$ *eq 7.40*

$$V_{T_{ref}} = V_T\left(\frac{N_{R_S}}{N_{R_T}}\right)$$ *eq 7.41*

$$\mu = 1.69\left(\frac{V_T}{\Omega R}\right)$$ *eq 7.42*

$$a = 38.967\sqrt{T_a\,(°K)}$$ *eq 7.43*

$$V_{TIP} = \left(\frac{\Omega R}{1.69}\right) + V_T$$ *eq 7.44*

$$M_{TIP} = \frac{V_{TIP}}{a}$$ *eq 7.45*

$$ESHP_{ref} = f\left(W_{ref},\ V_{T_{ref}}\right)$$ *eq 7.46*

$$ESHP_{ref} = E_0 + E_1\,V_{T_{ref}} + E_2\,V_{T_{ref}}^2 + E_3\,V_{T_{ref}}^3 + E_4\,V_{T_{ref}}^4$$ *eq 7.47*

$$\delta = \frac{P_a}{P_{ssl}} = \left[1 - \left(\frac{\lambda_{ssl}\ H_P}{T_{ssl}}\right)\right]^{\frac{g_{ssl}}{g_c\ \lambda_{ssl}\ R}}$$ *eq 7.48*

$$\theta = \frac{T_a}{T_{ssl}} = \frac{OAT + 273.15}{288.15}$$ *eq 7.49*

$$\sigma = \frac{\rho}{\rho_{ssl}} = \frac{\delta}{\theta}$$ *eq 7.50*

$$V_T = \frac{V_c}{\sqrt{\sigma_T}}$$ *eq 7.51*

$$M_{TIP} = 1.68781V_T + \left(\frac{\Omega R_T}{\sqrt{\gamma\ g_c\ R\ T_{ssl}\ \theta}}\right)$$ *eq 7.52*

$$ESHP_{ref} = A_0 + A_1\ W_{ref}^{\frac{3}{2}}$$ *eq 7.53*

$$W_{f_{corr}} = B_0 + B_1\ ESHP_{corr} + B_2\ ESHP^2_{corr} + B_3\ ESHP^3_{corr}$$ *eq 7.54*

$$\frac{P_{T_2}}{P_a} = C_0 + C_1\ V_c + C_2\ V_c^2 + C_3\ V_c^3$$ *eq 7.55*

$$T_{T_2} - T_a = D_0 + D_1\ V_c + D_2\ V_c^2 + D_3\ V_c^3$$ *eq 7.56*

$$SR = \frac{V_T}{1.05\ W_f}$$ *eq 7.57*

$$W = \frac{W_{ref}\ (\sigma)}{\left(\frac{N_{R_S}}{N_R}\right)^2}$$ *eq 7.58*

$$V_T = \frac{V_{T_{ref}}}{\left(\frac{N_{R_S}}{N_R}\right)}$$ *eq 7.59*

$$ESHP = \frac{ESHP_{ref}\,(\sigma)}{\left(\frac{N_{R_S}}{N_R}\right)}$$ *eq 7.60*

$$V_c = V_T\sqrt{\sigma}$$ *eq 7.61*

$$ESHP_{corr} = \frac{ESHP}{\delta_{T_2}\sqrt{\theta_{T_2}}}$$ *eq 7.62*

$$W_f = W_{f_{corr}}\left(\delta_{T_2}\right)\sqrt{\theta_{T_2}}$$ *eq 7.63*

$$\text{Endurance} = \frac{\text{Fuel available}}{1.05\ W_f}$$ *eq 7.64*

CHAPTER SEVEN

LEVEL FLIGHT PERFORMANCE

7.1 INTRODUCTION

This chapter examines how aircraft level flight performance is determined. Airframe level flight performance (power required) is combined with engine performance (power available) to determine aircraft level flight performance. The simplified theory of a rotor in level flight is presented with an analysis of power required.

7.2 PURPOSE OF TEST

The purpose of this test is to investigate aircraft performance characteristics in level flight to achieve the following objectives:

1. Determine significant performance parameters: maximum level flight airspeed (V_H), maximum range airspeed ($V_{max\ range}$), cruise airspeed (V_{cruise}), maximum endurance airspeed($V_{max\ end}$), combat radius, maximum endurance, and maximum range.
2. Define mission suitability problems.
3. Evaluate the pertinent requirements of the detailed specification.

7.3 THEORY

7.3.1 General

The concepts associated with the analysis of level flight power required are understood better when the individual contributions are examined separately. The total power required in forward flight (Figure 7.1) is defined as the summation of the following power requirements:

1. Induced power (P_i) : Power required to produce induced flow, affects main and tail rotor, discussed in terms of the momentum theory.

2. Profile power (P_0) : Power required to drag the rotor blade through a viscous fluid, affects main and tail rotor, discussed in terms of the blade element theory.
3. Parasite power (P_p) : Power required to drag fuselage through a viscous fluid, affected by equivalent flat plate area, discussed in terms of simple aerodynamic theory.
4. Miscellaneous power (P_m) : Power required to overcome cooling losses, transmission losses, accessory power (hydraulic and electrical) and tail rotor power required, generally presented in terms of mechanical efficiency, η_m, as a function of forward speed.

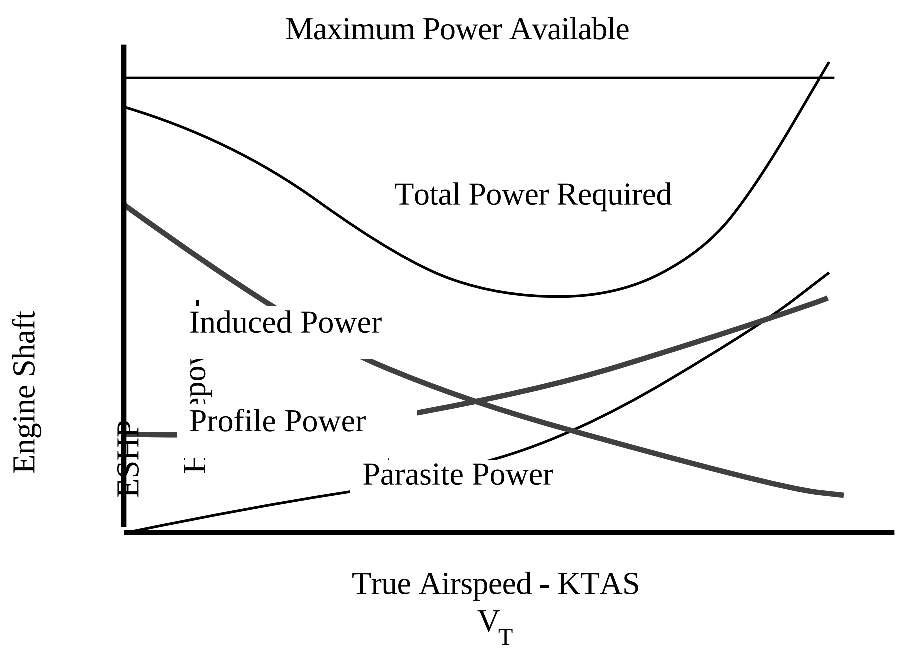

Figure 7.1
Analysis of Power Required in Level Flight

7.3.2 Induced Power

In the hover performance chapter, the momentum theory was used to explain the dependence of thrust and induced power on rotor disc area, air density, rate of flow through the disc, and the magnitude of the induced velocity. The vertical climb analysis discussed the variation of induced velocity (v_{i_v}) with vertical flight speed (V_V) and stated that the mass flow rate was dependent upon the sum of v_{i_v} and V_V:

Error! Not a valid link. *eq 7.1*

$$T_h = 2 \rho A_D v_i^2 \qquad \textit{eq 7.2}$$

$$T_v = \rho A_D (2 v_{i_v}) (v_{i_v} + V_V) \qquad \textit{eq 7.3}$$

Where:

T	- Thrust
t	- Time
m	- Mass
ΔV	- Change in velocity
T_h	- Thrust at hover
ρ	- Density
A_D	- Rotor disc area
v_i	- Induced velocity at hover
T_v	- Thrust in vertical climb
v_{i_v}	- Induced velocity in a vertical climb
V_V	- Actual rate of climb.

Induced power in forward flight can be analyzed by the momentum theory to explain an additional increase in mass flow rate through the disc with forward speed. The mass flow rate through the disc is a function of the vectorial sum of the component velocities at the rotor (Figure 7.2). This mass flow rate is multiplied by the induced velocity in forward flight to obtain thrust:

$$T_f = \rho A_D V_R (2 v_{i_f}) \qquad \textit{eq 7.4}$$

Where:

T_f - Thrust in forward flight

ρ - Density

A_D - Rotor disc area

V_R - Resultant velocity

v_{i_f} - Induced velocity in forward flight.

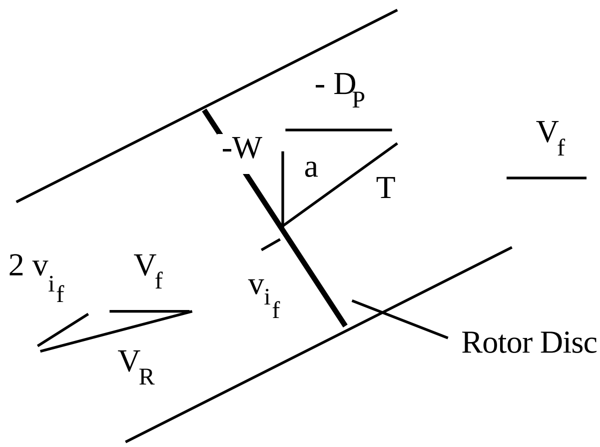

Figure 7.2
Velocities at the Rotor in Forward Flight

The mass of air in the mass flow rate expression is that air which flows through a circle whose diameter is assumed equal to the rotor disc diameter.

As shown in Figure 7.2, the thrust vector is assumed perpendicular to the rotor tip path plane and, for small angles of tilt, can be considered equal to the gross weight (15 degree tilt results in a 3 percent error). One additional assumption must be made before deriving an expression for induced power in forward flight. At relatively high forward speeds, the resultant velocity (V_R) is assumed equal to the forward velocity of the aircraft (V_f); in other words, the induced velocity in forward flight becomes small.

Using the law of cosines and Figure 7.2, the expression for V_R becomes:

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Where:

V_R	- Resultant velocity
V_f	- Forward flight velocity
v_{i_f}	- Induced velocity in forward flight
a	- Tilt angle between T and weight.

Then for small tilt angles and relatively high forward speeds:

$$V_R = V_f \qquad \text{eq 7.6}$$

Where:

V_R	- Resultant velocity
V_f	- Forward flight velocity.

The induced power in forward flight then becomes:

$$P_{i_f} = \frac{T^2}{2\,\rho\,A_D\,V_R} = \frac{W^2}{2\,\rho\,A_D\,V_f} \qquad \text{eq 7.7}$$

Where:

P_{i_f}	- Induced power in forward flight
T	- Thrust
ρ	- Density
A_D	- Rotor disc area
V_R	- Resultant velocity
W	- Weight
V_f	- Forward flight velocity.

7.3.3 Profile Power

Similar to hovering flight, the blade element theory is used to analyze the rotor blade profile power in forward flight. One obvious difference is the fact that the resultant velocity at the rotor blade is a function of azimuth angle (ψ). Lift and drag are calculated from the total velocity acting on each blade element. Blade element analysis in forward flight is shown in Figure 7.3.

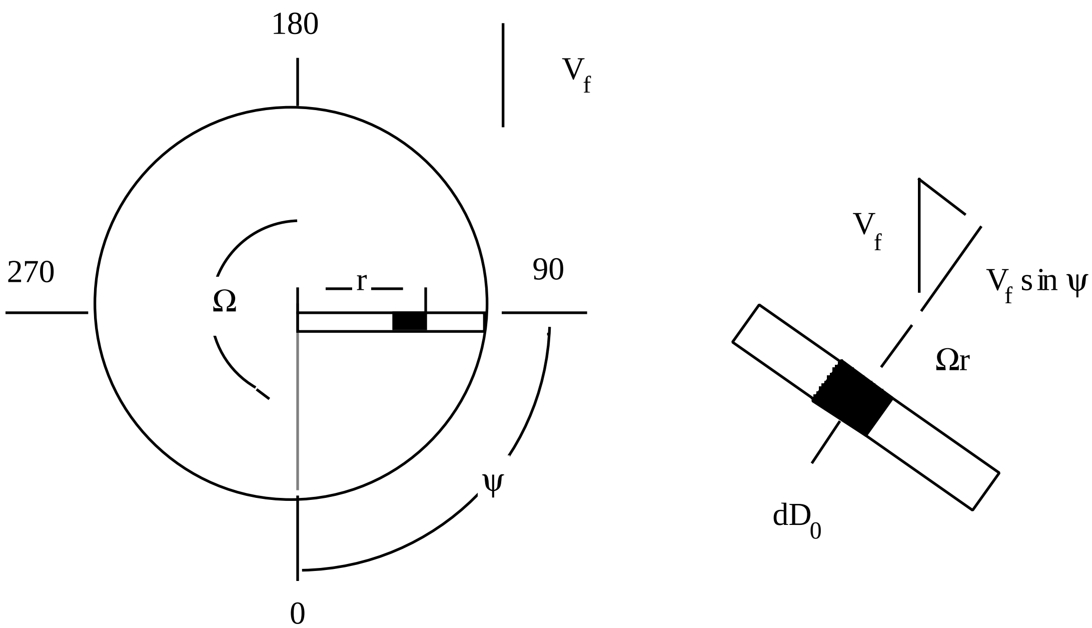


Figure 7.3
Blade Element Analysis in Forward Flight

The derivation for the profile power expression is similar to the hovering rotor except that the velocity of the blade element is now: $\Omega r + V_f \sin \psi$. Blade element profile power is represented by:

Error! Not a valid link. *eq 7.8*

Where:

dP_0	- Blade element profile power
dD_0	- Blade element profile drag
v_e	- Element velocity
C_{d_0}	- Blade element profile drag coefficient
ρ	- Density
Ωr	- Blade element rotational velocity
V_f	- Forward flight velocity
ψ	- Blade azimuth angle
c	- Blade chord
dr	- Blade element radius.

Since the profile power is a function of azimuth angle (ψ), it is cyclic in magnitude. The profile power is generally expressed as an average value (Figure 7.4).

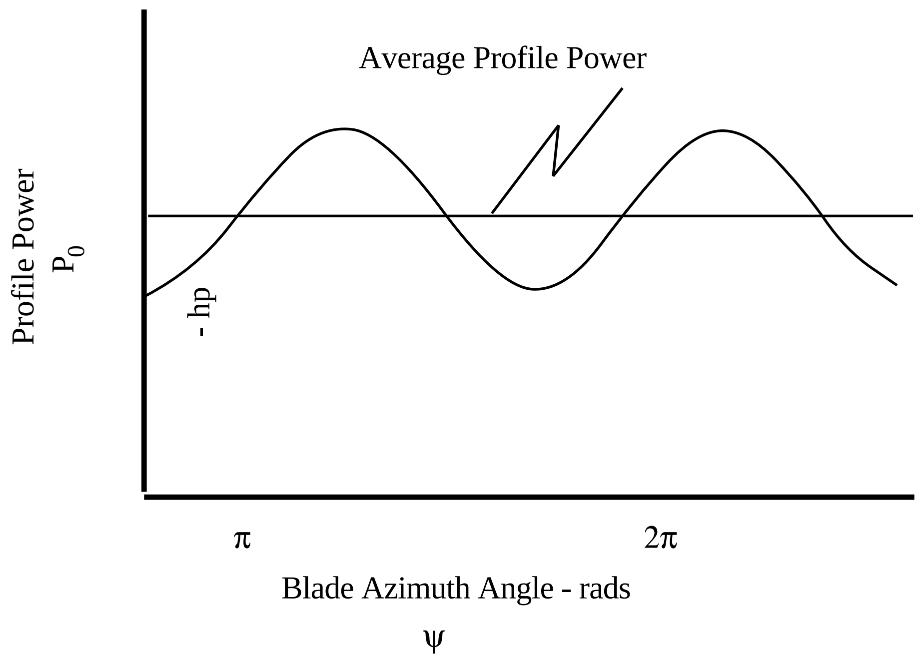


Figure 7.4
Variation of Profile Power with Azimuth Angle

The average profile power for b number of blades is obtained by integrating Equation 7.8 with respect to radius (dr) and azimuth (dψ):

Error! Not a valid link. *eq 7.9*

Where:

P_0	- Profile power
b	- Number of blades
C_{d_0}	- Blade element profile drag coefficient
ρ	- Density
Ωr	- Blade element rotational velocity
V_f	- Forward flight velocity
ψ	- Blade azimuth angle
c	- Blade chord.

The integration yields:

Error! Not a valid link. *eq 7.10*

Where:

P_0	- Profile power
σ_R	- Rotor solidity ratio, $bc/\pi R$
$\overline{C}_{d_0}$	- Average blade element profile drag coefficient
ρ	- Density
A_D	- Rotor disc area
ΩR	- Blade rotational velocity
μ	- Advance ratio, $V_f / \Omega R$.

The following term in Equation 7.10:

$$\frac{3}{8}\sigma_R \, \overline{C}_{d_0} \, \rho A_D \, (\Omega R)^3 \, \mu^2$$

accounts for the forward flight contribution to profile power because of chordwise flow.

Spanwise flow over the rotor blades requires one further consideration in the determination of profile power. When the blade is at an azimuth angle of 90 degrees or 270 degrees, there is no spanwise flow. When the blade is at any other azimuth angle, the spanwise flow produces a force (H_2 force) which is also cyclic in magnitude (Figure 7.5).

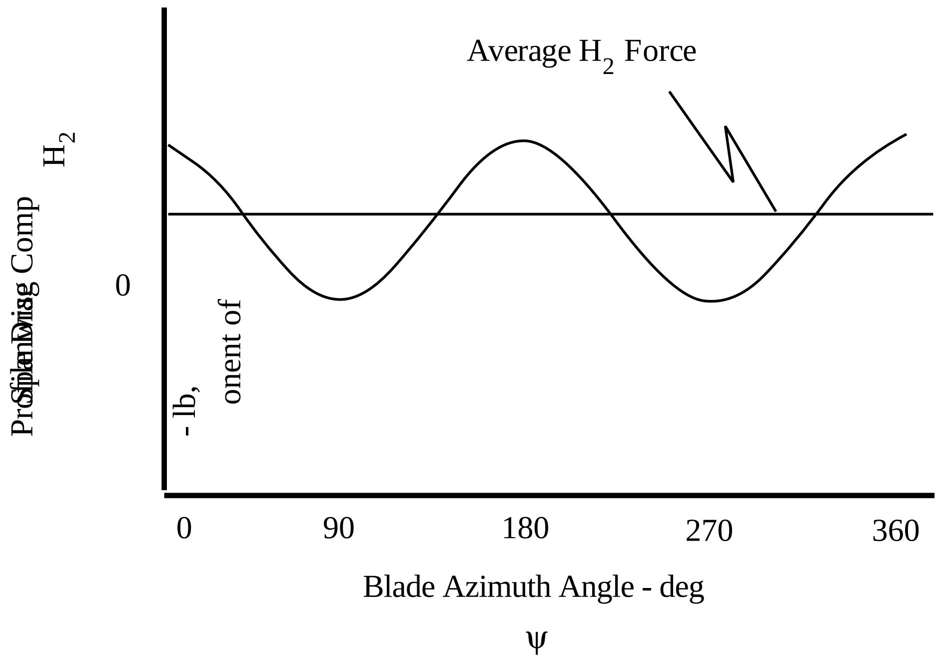


Figure 7.5
Spanwise Drag Variation with Azimuth Angle

An expression for the H_2 component of profile drag in forward flight can be stated in terms of the average profile drag coefficient and the area of the blade itself. The power required to overcome the average H_2 force for b number of blades becomes:

Error! Not a valid link. *eq 7.11*

Where:

$P_{0_{H2}}$	- Profile power due to H_2 force
σ_R	- Rotor solidity ratio
$\overline{C}_{d_0}$	- Average blade element profile drag coefficient
ρ	- Density
A_D	- Rotor disc area
ΩR	- Blade rotational velocity
μ	- Advance ratio.

The total profile drag expression now becomes:

Error! Not a valid link. *eq 7.12*

Where:

P_0	- Profile power
σ_R	- Rotor solidity ratio
$\overline{C}_{d_0}$	- Average blade element profile drag coefficient
ρ	- Density
A_D	- Rotor disc area
ΩR	- Blade rotational velocity
μ	- Advance ratio.

7.3.4 Parasite Power

Parasite drag of a fuselage is composed of two elements: skin friction and pressure drag. Since the fuselage is seldom a pure symmetrical shape, the components of the fuselage (landing gear, rotor mast, vertical fin, etc.) are assigned drag coefficients and summed into a total effective drag coefficient or equivalent flat plate area (f).

The fuselage drag using the equivalent flat plate area can be expressed as:

Error! Not a valid link. *eq 7.13*

Where:

D_p - Parasite drag

ρ - Density

C_{D_p} - Parasite drag coefficient

A_f - Frontal area

V - Velocity.

The parasite power in forward flight is a product of drag times velocity:

Error! Not a valid link. *eq 7.14*

or:

Error! Not a valid link. *eq 7.15*

Where:

P_p - Parasite power

ρ - Density

V_f - Forward flight velocity

ΩR - Blade rotational velocity

f - Equivalent flat plate area, $C_{D_p} A_f$

μ - Advance ratio.

7.3.5 Miscellaneous Power

Miscellaneous power in forward flight is comprised of the same elements as in a hover: transmission losses, accessory power, and tail rotor power. Transmission losses are dependent on rotor speed. Accessory power requirements vary considerably with the number of accessories driven from the main transmission such as: hydraulic pumps, generators, and oil coolers. Accessory power is generally small in terms of percent total power required but can become significant under heavy electrical or hydraulic loads.

Tail rotor power is composed of two elements: induced and profile power. If the thrust required by the tail rotor is known, the expressions used for main rotor induced and profile power can be used to determine the tail rotor components. The thrust required by the tail rotor in equilibrium flight is defined as:

$$T_{TR} = \frac{Q_{MR}}{\ell_{TR}} \qquad \textit{eq 7.16}$$

Where:

T_{TR}	- Tail rotor thrust
Q_{MR}	- Main rotor torque
ℓ_{TR}	- Tail rotor moment arm.

The tail rotor power required becomes:

$$P_{TR} = \left(\frac{T_{TR}^2}{2\,\rho\,A_D\,V_f}\right)_{TR} + \left[\frac{1}{8}\,\sigma_R\,\overline{C}_{d_0}\,\rho\,A_D\,(\Omega R)^3\left(1 + 4.65\mu^2\right)\right]_{TR} \qquad \textit{eq 7.17}$$

Where:

P_{TR}	- Tail rotor power required
T_{TR}	- Tail rotor thrust
ρ	- Density
A_D	- Rotor disc area
V_f	- Forward flight velocity

σ_R - Rotor solidity ratio

$\overline{C}_{d_0}$ - Average blade element profile drag coefficient

ΩR - Blade rotational velocity

μ - Advance ratio.

When discussing total engine shaft horsepower, miscellaneous power (P_m) is generally expressed in terms of a mechanical efficiency factor (η_m). The mechanical efficiency includes the three elements of miscellaneous power and varies as a function of forward speed. The use of the mechanical efficiency factor is not entirely accurate when changing rotor shaft horsepower to engine shaft horsepower due to the density and rotor speed effects on tail rotor power requirements. However, for a first approximation, miscellaneous power is considered only a function of forward speed.

7.3.6 Total Power Required

The total power required in forward flight is the summation of induced, profile, parasite, and miscellaneous power:

$$P_{TOTAL} = P_i + P_0 + P_p + P_m \qquad \textit{eq 7.18}$$

Where:

P_{TOTAL} - Total power

P_i - Induced power

P_0 - Profile power

P_p - Parasite power

P_m - Miscellaneous power.

The main rotor power required is:

$$P_{MR} = \left(\frac{W^2}{2\,\rho\,A_D\,V_f}\right)_{MR} + \left[\frac{1}{8}\,\sigma_R\,\bar{C}_{d_0}\,\rho\,A_D\,(\Omega R)^3\left(1 + 4.65\mu^2\right)\right]_{MR} + \left(\frac{1}{2}\,\rho\,V_f^3\,f\right)$$

eq 7.19

Where:

P_{MR}	- Main rotor power
W	- Weight
ρ	- Density
A_D	- Rotor disc area
V_f	- Forward flight velocity
σ_R	- Rotor solidity ratio
$\bar{C}_{d_0}$	- Average blade element profile drag coefficient
ΩR	- Blade rotational velocity
μ	- Advance ratio
f	- Equivalent flat plate area.

The relationship between main rotor power required and the variables in the above equation provides the basis for the level flight method of test. The variables for each element of the total power required are as follows:

Induced:	Weight (W), density (ρ), and forward flight velocity (V_f).
Profile:	Density (ρ), rotor speed (ΩR), and advance ratio (μ).
Parasite:	Density (ρ) and forward flight velocity (V_f).

The power required to maintain forward flight is dependent on gross weight, density, forward flight velocity, and rotor speed. The variables in terms of nondimensional coefficients provide a means of evaluating level flight performance. The total power required by the main rotor in nondimensional form becomes:

$$C_P = \frac{C_T^2}{2\mu} + \left[\frac{1}{8}\sigma_R\,\bar{C}_{d_0}\left(1 + 4.65\mu^2\right)\right] + \left(\frac{1}{2}C_{D_p}\,\mu^3\right) \qquad \textit{eq 7.20}$$

Where:

C_P	- Power coefficient
C_T	- Thrust coefficient
μ	- Advance ratio
σ_R	- Rotor solidity ratio
$\bar{C}_{d_0}$	- Average blade element profile drag coefficient
C_{D_p}	- Parasite drag coefficient.

The power coefficient (C_P) is a function of the thrust coefficient (C_T) and advance ratio (μ). Therefore, to obtain level flight power required data, μ is varied for constant values of C_T. The expression of C_T is:

$$C_T = \frac{\left(\frac{W}{\sigma_T}\right)}{\rho_{ssl}\,A_D\,(\Omega R)^2} \qquad \textit{eq 7.21}$$

Where:

C_T	- Thrust coefficient
W	- Weight
σ_T	- Test density ratio
ρ_{ssl}	- Standard sea level air density
A_D	- Rotor disc area
ΩR	- Blade rotational velocity.

Examining Equation 7.21, it is evident that a constant C_T may be obtained in several ways:

1. Maintain (ΩR) constant and vary W and σ to keep W/σ constant.
2. Maintain σ constant and vary W and (ΩR) to keep $W/(\Omega R)^2$ constant.

The easiest method is to maintain a constant rotor speed and vary the density ratio as fuel is consumed to maintain a constant W/σ.

7.3.7 Factors Affecting Level Flight Performance

7.3.7.1 REVERSED FLOW

When determining rotor profile drag, the effect of reversed flow was not considered. Reverse flow occurs on the retreating blade as a function of azimuth angle and advance ratio. Since the resultant velocity on a blade element is the sum of the rotational speed and the forward velocity component ($\Omega r + V_f \sin \psi$), the forward velocity component ($V_f \sin \psi$) on the retreating side of the disc will be negative (ψ = 180 to 360 degrees). A region of reversed flow develops at some particular radius and extends toward the rotor hub as shown in Figure 7.6.

The radius, at which the resultant velocity is zero, is the locus of the reversed flow boundary and is defined as:

$$r = -\mu \, R \, \sin \psi \qquad \textit{eq 7.22}$$

Where:

r	- Radius where $V_R = 0$
μ	- Advance ratio
R	- Rotor radius
ψ	- Blade azimuth angle.

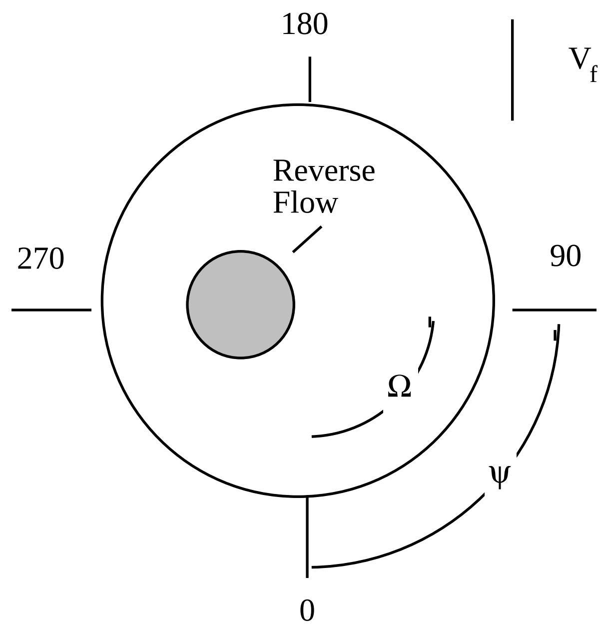


Figure 7.6
Area of Reverse Flow

The profile drag equation including reversed flow effects is:

$$P_0 = \frac{1}{8}\sigma_R \bar{C}_{d_0} \rho A_D (\Omega R)^3 \left(1 + 4.65\mu^2 - \frac{3}{8}\mu^4\right) \qquad \textit{eq 7.23}$$

Where:

P_0	- Profile power
σ_R	- Rotor solidity ratio
$\bar{C}_{d_0}$	- Average blade element profile drag coefficient
ρ	- Density
A_D	- Rotor disc area
ΩR	- Blade rotational velocity
μ	- Advance ratio.

The quantity ($3/8\ \mu^4$) is the contribution of reversed flow to the total profile drag and is sufficiently small in magnitude to be neglected in most performance calculations.

7.3.7.2 COMPRESSIBILITY (MACH EFFECTS)

As discussed in Chapter 5, Hover Performance, compressibility is an aerodynamic phenomenon that results in an abrupt power required increase as the advancing tip speed approaches the speed of sound. At a constant lift coefficient (C_L), the profile drag coefficient (C_{d_0}) will increase with increasing Mach number of the advancing blade tip (Figure 7.7).

If the advancing blade tip Mach number (M_{TIP}) is low, the effects of compressibility on performance are negligible. However, once the M_{TIP} becomes sufficiently large (0.8), the increase in power required due to compressibility becomes a significant factor in performance test results.

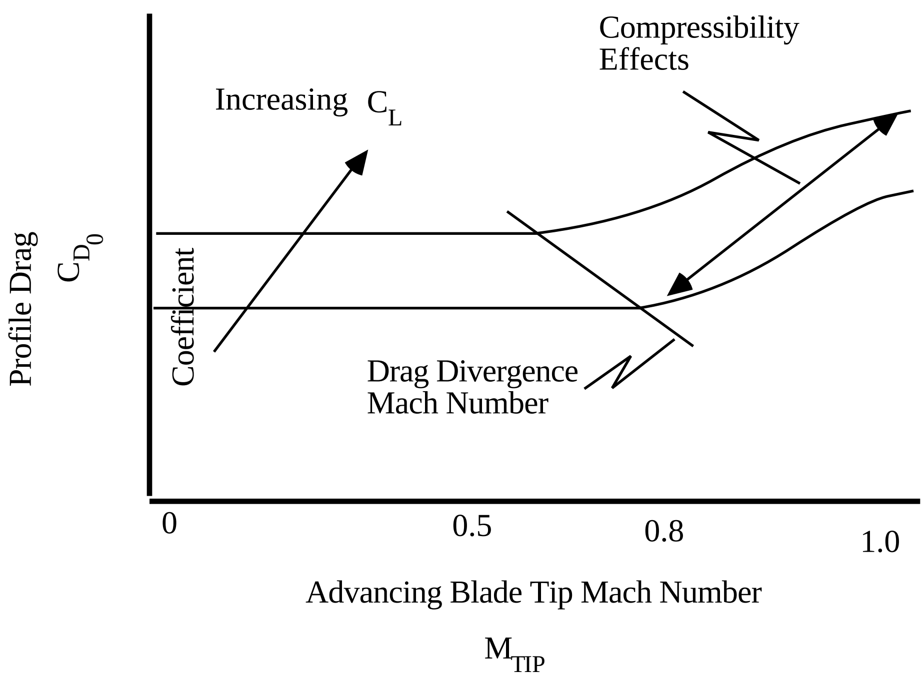


Figure 7.7
Advancing Blade Tip Mach Number (M_{TIP})

7.3.8 Referred Level Flight Performance

Typical values of C_T and C_P are 0.005 and 0.0003, respectively. Typical maximum values of the advance ratio (μ) are 0.3 to 0.4. The nondimensional parameters (C_P, C_T, and μ) may not be grasped easily by the pilot. Therefore, as was done in the hover performance calculations, forward flight performance data are often presented in terms of the referred system.

When C_P, C_T, and μ are multiplied by standard values of the nondimensionalizing terms, the following referred terms are obtained:

$$RSHP_{ref} = \frac{C_P\ \rho_{ssl}\ A_D\ \left(\Omega R_S\right)^3}{550} = \left(\frac{RSHP_T}{\sigma_T}\right)\left(\frac{N_{R_S}}{N_{R_T}}\right)^3 \qquad \textit{eq 7.24}$$

$$W_{ref} = C_T\ \rho_{ssl}\ A_D\ \left(\Omega R_S\right)^2 = \frac{W_T}{\sigma_T}\left(\frac{N_{R_S}}{N_{R_T}}\right)^2 \qquad \textit{eq 7.25}$$

$$V_{T_{ref}} = \mu\left(\Omega R_T\right) = V_T\left(\frac{N_{R_S}}{N_{R_T}}\right) \qquad \textit{eq 7.26}$$

Where:

$RSHP_{ref}$	- Rotor shaft horsepower
$RSHP_T$	- Test rotor shaft horsepower
C_P	- Power coefficient
ρ_{ssl}	- Standard sea level air density
A_D	- Rotor disc area
ΩR_S	- Standard blade rotational velocity
ΩR_T	- Test blade rotational velocity
σ_T	- Test density ratio

W_{ref}	- Referred weight
W_T	- Test weight
C_T	- Thrust coefficient
N_{R_S}	- Standard main rotor speed
N_{R_T}	- Test main rotor speed
$V_{T_{ref}}$	- Referred true airspeed
V_T	- True airspeed
μ	- Advance ratio.

Since referred power may be either RSHP or ESHP, the level flight performance evaluation will be based on ESHP. The computation of RSHP is accomplished using the mechanical efficiency as long as the limitations of the relationship are understood. Mechanical efficiency (η_m) is presented as a function of true airspeed (Figure 7.8). In many test programs, both the main and tail rotors are instrumented to separate main and tail rotor power requirements.

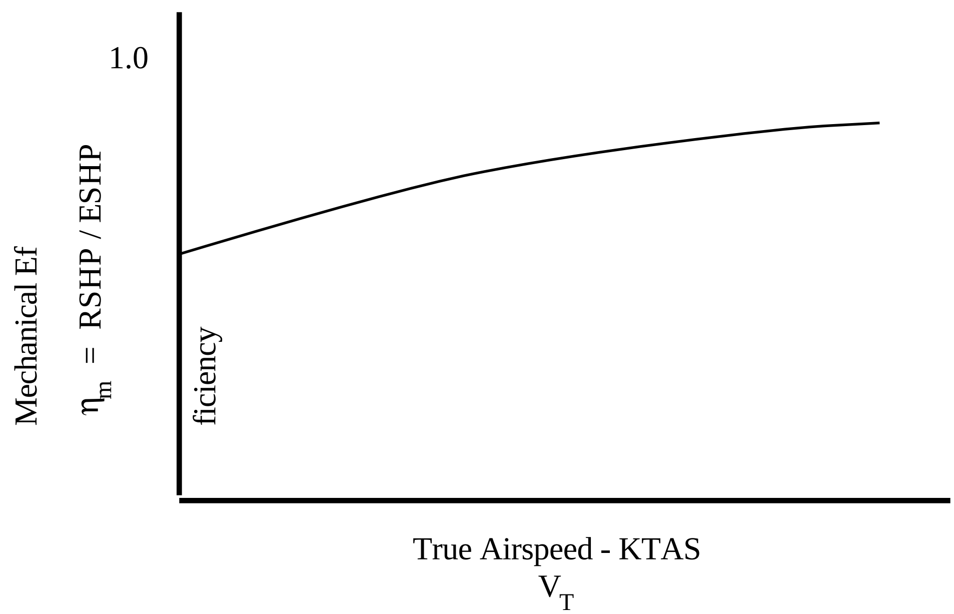


Figure 7.8
Mechanical Efficiency

The variation of tail rotor power with ambient conditions should not be ignored when computing engine shaft horsepower in a helicopter performance evaluation. The expression for the referred engine shaft horsepower is:

$$ESHP_{ref} = \frac{ESHP_T}{\sigma_T}\left(\frac{N_{R_S}}{N_{R_T}}\right)^3 \qquad \textit{eq 7.27}$$

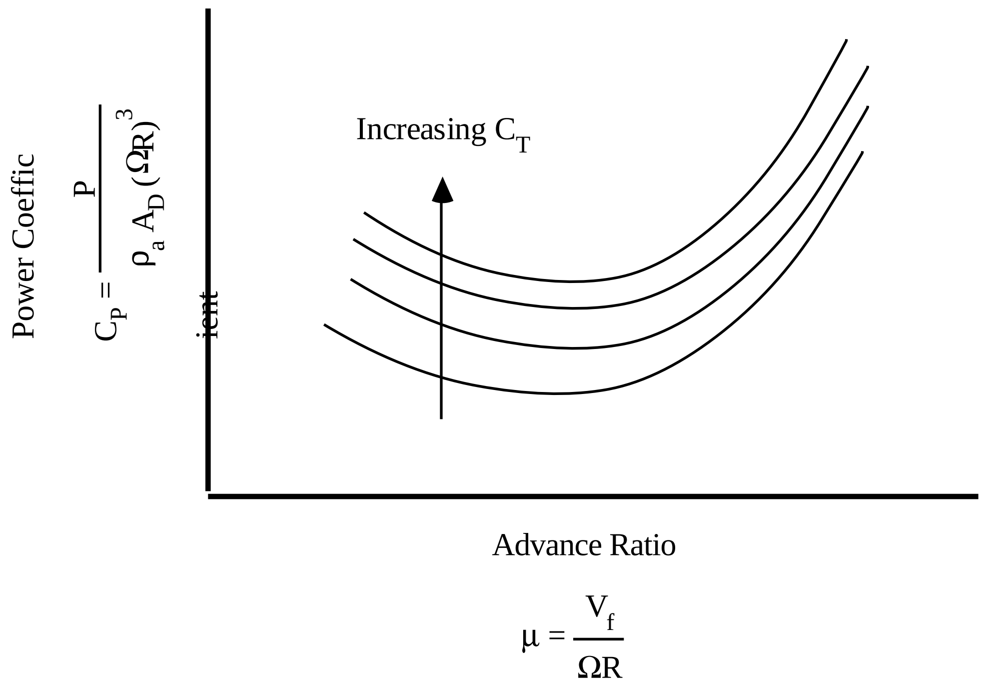
Where:

$ESHP_{ref}$	- Referred engine shaft horsepower
$ESHP_T$	- Test engine shaft horsepower
σ_T	- Test density ratio
N_{R_S}	- Standard main rotor speed
N_{R_T}	- Test main rotor speed.

7.3.9 Generalized Level Flight Performance

The previous paragraphs established that level flight power required data generalizes as a function of the advance ratio for given values of thrust coefficient. Level flight power required data also generalizes for true airspeed and gross weight when reduced through the referred system to sea level, standard day conditions. Figure 7.9 illustrates a method of presenting generalized level flight performance.

Theoretically, either plot of Figure 7.9 can be used to determine the level flight performance at any altitude as long as the C_T or W/σ corresponds to the one on the plot. This assumption is not entirely accurate as variations in profile and parasite drag resulting from altitude effects may significantly change the power required. For this reason, level flight performance data should not be extrapolated to ambient conditions which are considerably different than actual test conditions.



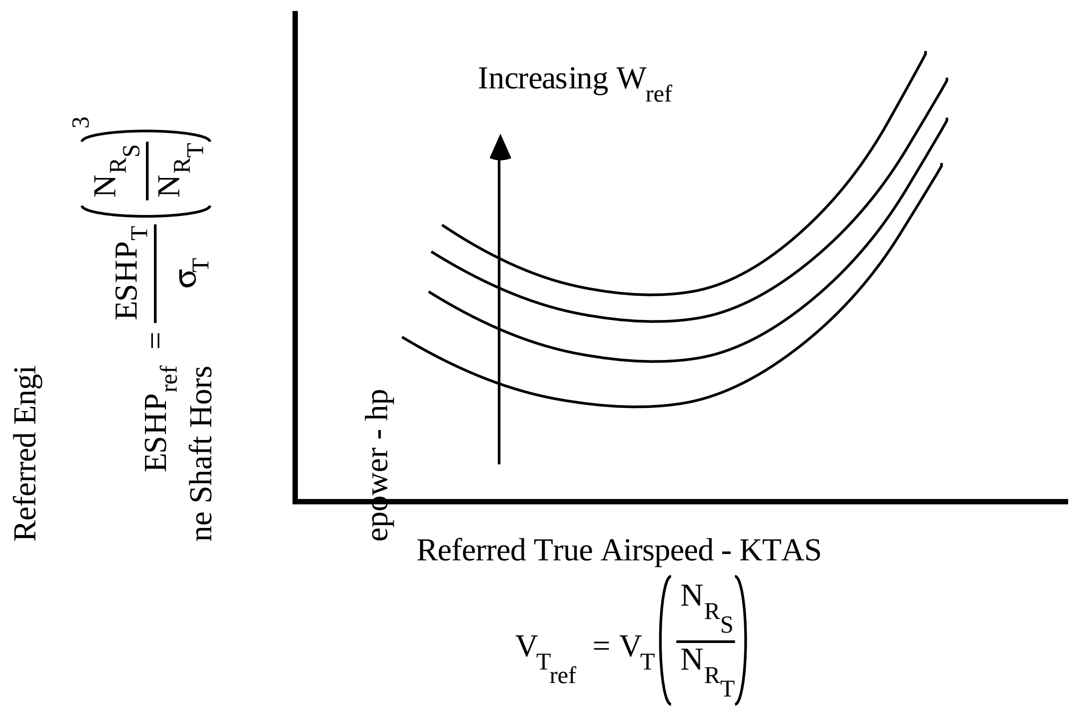


Figure 7.9
Level Flight Data

7.3.10 Airframe Characteristics

Airframe characteristics are by definition all characteristics of the aircraft which are not a function of the engine operation. The analysis of airframe characteristics neglects fuel consumption and engine power available. Airframe characteristics are a function of the airframe drag and may be identical for single engine or multiengine operation (neglecting intake drag and drag due to sideslips with engine out operation). An evaluation of the airframe characteristics is accomplished by examining a power polar and determining three particular airspeeds of interest:

1. Minimum power required airspeed.
2. Airframe maximum range airspeed $(P/L)_{min}$.
3. Maximum level flight airspeed (airframe limited).

7.3.10.1 POWER POLAR

The power polar is a graphic presentation of airframe performance in terms of horsepower required and airspeed. Figure 7.10 illustrates a power polar and the corresponding airspeeds of interest.

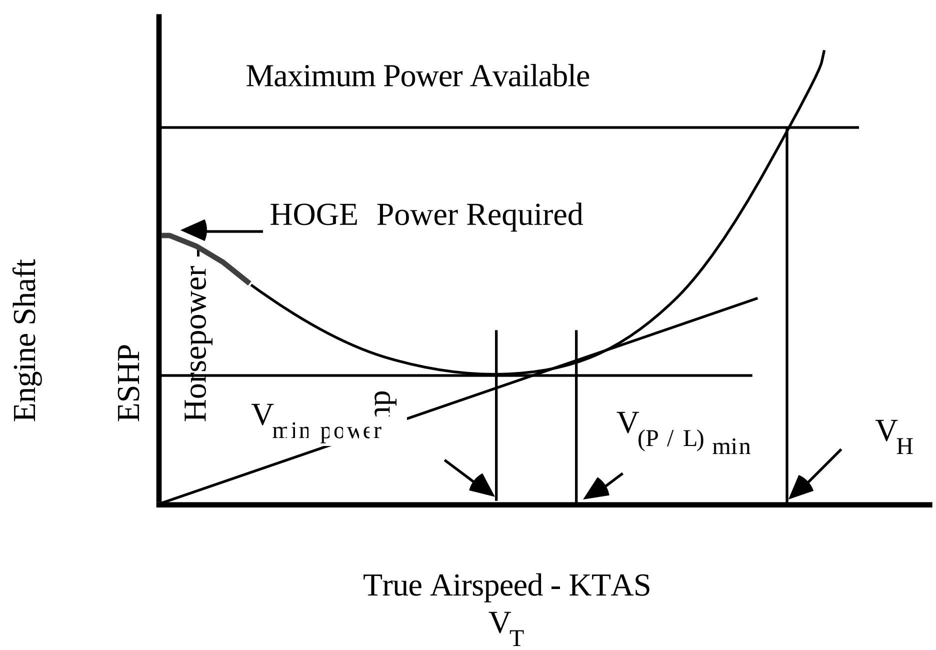


Figure 7.10
Power Polar

7.3.10.2 MINIMUM POWER REQUIRED

The airspeed for minimum power required (Figure 7.11) is roughly equal to the airspeed for maximum endurance and a fair approximation of the airspeed for maximum rate of climb. The shape of the power required curve in the region of minimum power required is of particular importance as it affects the amount of pilot effort required to fly the aircraft at minimum power.

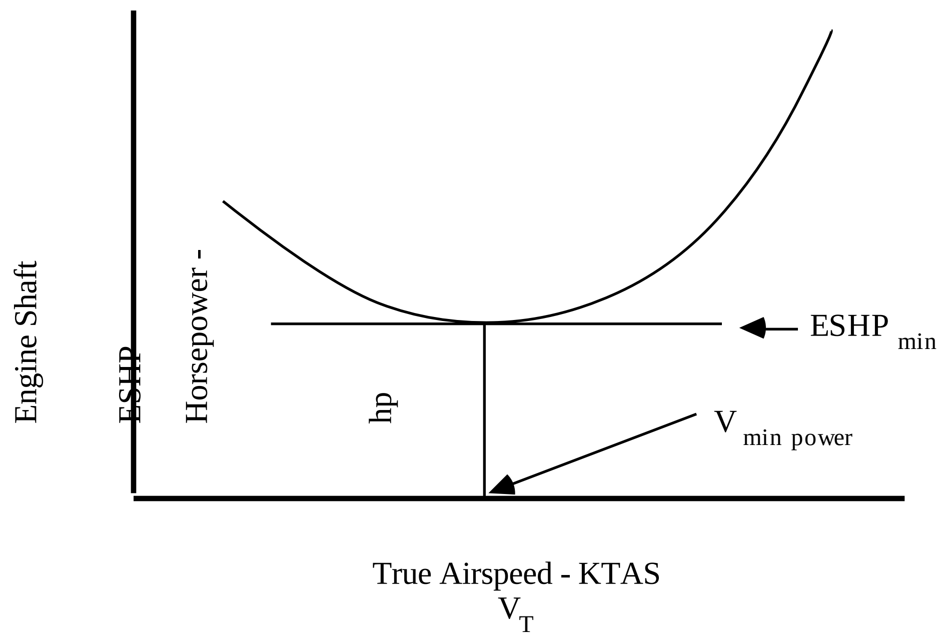


Figure 7.11
Minimum Power Required

7.3.10.3 AIRFRAME MAXIMUM RANGE - $(P/L)_{MIN}$

The airspeed for $(P/L)_{min}$ is the airspeed where the ratio of power required to velocity is minimum. The power required could be expressed as:

$$P = D_{eff} V_f \qquad \textit{eq 7.28}$$

Where:

P	- Power
D_{eff}	- Effective drag
V_f	- Forward flight velocity.

D_{eff} is the drag that would absorb the same power at the velocity along the flight path as the power being supplied through the main rotor shaft. Hence, at the airspeed for $(P/L)_{min}$, D_{eff} is a minimum, or the vehicle is operating at the maximum effective lift to drag (L/D) ratio. The airspeed for $(P/L)_{min}$ is the maximum range airspeed, considering airframe characteristics only. The speed for $(P/L)_{min}$ is also a good first estimate of the speed for maximum range during an autorotative descent. The airspeed for $(P/L)_{min}$ corresponds to the point of tangency of a ray from the origin to the power polar (Figure 7.10).

7.3.10.4 MAXIMUM FORWARD FLIGHT AIRSPEED (AIRFRAME LIMITED)

The maximum level flight airspeed is defined as the airspeed where power available meets power required (which is an aircraft characteristic) or the airspeed above which flight is not permitted (an airframe characteristic). The airframe limit is generally based on structural strength of component parts, blade stall, or Mach effects (compressibility).

7.3.11 Engine Characteristics

Power available and fuel flow characteristics of the installed engine are a primary concern during a performance flight test program. Power available is determined for sea level and altitudes required by the performance guarantee. However, as discussed in Chapter 4, power available should be determined for all altitudes and airspeeds within the flight envelope of the aircraft. Flight test should be conducted to determine the variation between test cell power and installed power and to determine the inlet losses resulting from the installation.

7.3.11.1 INLET LOSSES

Losses result from location of the duct in relation to airframe structure and the inlet design. The effects of inlet condition on power available was discussed in Chapter 4. In addition to inlet distortion in balanced flight, moderate sideslip angles or airframe inflow angles can cause inlet distortion and produce power available losses. In multiengine installations, inlet distortion can cause power management problems or unexpected reductions in power available during single engine operation. Careful consideration of inlet design and duct location will normally suggest a test for determining inlet distortion.

7.3.12 Aircraft Characteristics

Aircraft characteristics are defined as those resulting from the combination of engine and airframe performance. When discussing service suitability or specification compliance, the aircraft characteristics must be used. Engine and airframe characteristics have been discussed to investigate the contribution each makes to the final performance capabilities of the test aircraft.

7.3.12.1 ENDURANCE

Fuel flow variation with airspeed is used in determining endurance rather than ESHP variation with airspeed. Comparing the airspeed range within 1% of minimum power required with the airspeed range within 1% of maximum endurance, the difference between airframe power required and aircraft endurance is seen. The airspeed range within 1% of maximum specific endurance is larger because of decreasing specific fuel consumption with increasing power on both sides of the airspeed for minimum power required.

Aircraft maximum endurance airspeed occurs at the low point of the fuel flow curve. The guarantee for maximum endurance is computed using the detail specification and Military Specification MIL-C-5011A and generally is computed for one set of conditions: at sea level, standard day conditions for the standard gross weight and rotor speed.

When recommending an endurance airspeed, factors such as vibration levels, field of view, and flight path stability should be considered as well as the specific endurance information obtained during flight tests.

7.3.12.2 RANGE

Range information is presented as specific range. Cruise speed is defined by Military Specification MIL-C-5011A as the airspeed corresponding to the higher value of 99 percent maximum specific range. The determination of the cruise speed is shown in Figure 7.12.

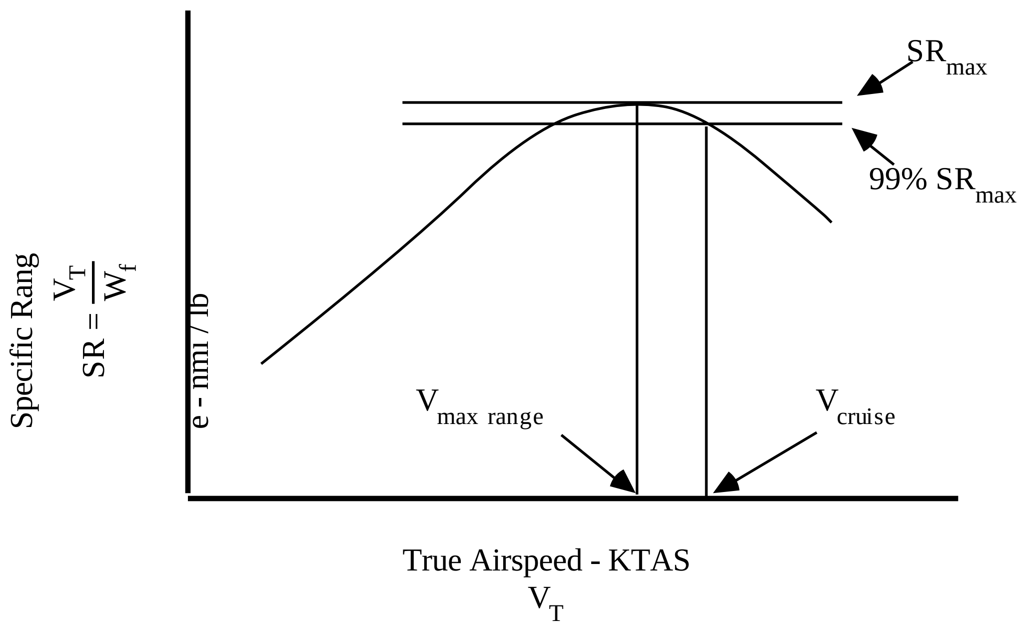


Figure 7.12
Specific Range

Specific range information is an aircraft characteristic and differs from the airframe maximum range, $(P/L)_{min}$, airspeed. The cruise speed as determined from 99 percent SR_{max} will be higher generally than the airframe maximum range airspeed due to increased efficiency provided by the engine specific fuel consumption characteristics.

7.4 TEST METHODS AND TECHNIQUES

7.4.1 General

Not considering compressibility effects on the advancing blade, there are two basic methods of maintaining a constant thrust coefficient (C_T) . The density can be held constant and rotor speed varied to maintain $W/(\Omega R)^2$ constant or the rotor speed held constant, and the density altitude increased as fuel is consumed to keep W/σ constant. The latter method, the W/σ method, is the easiest and most efficient. The W/σ ratio is held constant and airspeed is varied to obtain the level flight power required data. By repeating the test for various values of W/σ, a complete level flight airspeed-altitude envelope can be constructed.

When compressibility effects are suspected to be a problem, M_{TIP} greater than 0.8 Mach, the W/δ method is employed. In the W/δ method, the weight to pressure ratio,W/δ, is held constant and the main rotor speed to the square root of the temperature ratio, $N_R/\sqrt{\theta}$, is also held constant. The change in density is accounted for through the pressure ratio and the temperature ratio separately. By holding $N_R/\sqrt{\theta}$ constant while altitude is increased (decreasing temperature) as fuel is burned, the effects of a change in compressibility at the blade tip are reduced.

A comparison of the two methods gives the following results:

W/δ , $N_R/\sqrt{\theta}$:

1. Results are generally more accurate since any compressibility present is accounted for.
2. Results are at a standard M_{TIP}.
3. Pilot is required to vary N_R.
4. Measurement of pressure is easier than density, therefore test altitude is fixed based on weight without iterating for nonstandard temperatures.
5. Power and engine performance curves are corrected using a similar method.

W/σ:

1. Must be used if N_R can't be varied (fixed turbine engine).
2. Iterate pressure altitude and temperature to obtain correct density ratio.
3. One less parameter to control (N_R).

Since a direct reading of density ratio, pressure ratio, and temperature ratio are not available to the pilot, a considerable amount of preflight planning is required to determine a corresponding pressure altitude. Using a known engine start gross weight, a curve is constructed to show the variation of density ratio or pressure ratio with fuel consumed for a constant test referred weight (W/σ or W/δ). A complete discussion of chart construction is contained in Appendix VII.

7.4.2 Techniques

The chart used during the level flight test is constructed using fuel counter numbers in terms of fuel consumed or other parameters to account for GW changes. This allows the pilot to be at the exact gross weight desired and is very useful to plan succeeding data points. To ensure the aircraft is stabilized, one minute of stabilized flight is required before recording data.

When establishing the pressure altitude for each data point, the aircraft is stabilized at a constant airspeed before determining the existing outside air temperature (OAT). Indicated OAT readings may change with variations in airspeeds. The constant altitude technique should be used on the “front” side of the power curve, and the constant airspeed technique used on the “back” side of the power curve. On the front side, fix the engine power then adjust attitude, maintaining altitude, to obtain stabilized airspeed. On the back side stabilize on airspeed by adjusting attitude then adjust power to maintain constant altitude. When recording data the collective is fixed. For airspeed near the airspeed for minimum power, a combination of the two techniques is used. Cyclic and directional control inputs are minimized during data recording.

If the level flight performance tests are conducted within the established flight envelope, the converging airspeed method of gathering data is used to expedite data acquisition and to prevent excessive altitude variations due to fuel consumption. The data are recorded for decreasing airspeeds beginning with the maximum allowable airspeed. If the level flight tests are conducted near the boundaries of the envelope, a more conservative approach of incrementally increasing the airspeed to maximum is used.

The normal test procedure is for the pilot to have an aim rather than exact airspeed for each data point. Power is added, or subtracted, until the aircraft is near the aim airspeed. The power is fixed while the pilot adjusts rotor speed, attitude, and airspeed. Small accelerations are difficult to perceive. A thorough, rapid scan, directing attention outside the aircraft on the horizon is important in observing small changes in attitude. Data are recorded in wings level, ball-centered, level (no vertical velocity), stabilized (unaccelerated) flight. When no change has been observed for one minute, the parameters are stabilized. Data are recorded starting with the most important first, airspeed and power.

The pilot must calculate or be furnished the new altitude for the next data point to maintain a constant W/σ or W/δ ratio. A suggested procedure is as follows:

1. Determine the current gross weight from the fuel used or remaining indicator.
2. Estimate fuel consumption while climbing and stabilizing on the next data point. If using the converging airspeed method, excess airspeed is used to climb while reducing power. If increasing airspeed, increased power is used to climb and accelerate.
3. From steps 1 and 2, determine the approximate gross weight for the test point.
4. Enter the W/σ or W/δ chart to obtain the proper altitude (for the test gross weight).
5. Apply instrument and position corrections to the calibrated altitude obtained from the chart to obtain an observed altitude.
6. When on altitude and approximate airspeed, obtain the indicated temperature and correct the altitude if necessary to achieve constant W/σ. Adjust the N_R as appropriate to maintain a constant $N_R/\sqrt{\theta}$ (W/δ method only).

7. To minimize the delay between data points, the pilot climbs a nominal amount and either accelerates or decelerates a nominal amount while the flight test engineer calculates the next data point.

The speed increment between points will vary; however, the entire speed range from approximately 30 knots to V_H is covered. Place emphasis on the area surrounding the bucket airspeed or airspeed for minimum power. A minimum of eight data points is required to define one level flight curve.

Maintaining an inflight progress chart of test airspeed and engine torque (power) will give an indication of coverage of the airspeed range. If large gaps in the data are observed, obtain additional data points to fill in the curve.

Tests are conducted at specification and representative CG and aircraft configuration. Tests are conducted at the forward and aft CG extremes to quantify the effects on performance.

7.4.2.1 DATA REQUIRED

V_o boom and aircraft system, Q, N_R, W_f, FU or FC, T_o, H_{P_o}.

7.4.2.2 TEST CRITERIA

1. Balanced (ball centered), wings level, unaccelerated level flight.
2. Engine bleed air systems off or operated in normal flight mode.
3. Stabilized engine power demand.

7.4.2.3 DATA REQUIREMENTS

1. Stabilize 60 s minimum prior to recording data.
2. Record stabilized data 60 s.
3. $V_o \pm 1$ kn.
4. $H_{P_o} \pm 20$ feet.
5. $N_R \pm 0.2\%$.

7.4.2.4 SAFETY CONSIDERATIONS/RISK MANAGEMENT

If the level flight tests are conducted within an established flight envelope, consider normal operating limitations. The converging airspeed method may be used. If the tests are conducted beyond, on or near a boundary to the established flight envelope, use normal incremental build up procedures. Be careful of unusual handling characteristics and vibrations at both high and low airspeeds especially when varying rotor speed.

7.5 DATA REDUCTION

7.5.1 General

USNTPS uses a computer based data reduction routine for helicopter performance data reduction. This section covers a manual method of level flight performance data reduction with an introduction to the computer based data reduction routine.

Whether W/σ or W/δ test methods are used, the data reduction required is very similar. The airspeed, gross weight, and power required must be calculated.

All observed data must be corrected for instrument and position errors as appropriate.

7.5.2 Manual Data Reduction

The following equations are used in the manual level flight performance data reduction:

$$H_{P_c} = H_{P_o} + \Delta H_{P_{ic}} + \Delta H_{pos} \qquad \textit{eq 7.29}$$

Where:

H_{P_c}	- Calibrated pressure altitude
H_{P_o}	- Observed pressure altitude
$\Delta H_{P_{ic}}$	- Altimeter instrument correction
ΔH_{pos}	- Altimeter position error.

$$T_a = T_o + \Delta T_{ic}$$ *eq 7.30*

Where:

T_a	- Ambient temperature
T_o	- Observed temperature
ΔT_{ic}	- Temperature instrument correction.

$$T_a(^\circ K) = T_a(^\circ C) + 273.15$$ *eq 7.31*

Where:

$T_a(^\circ K)$	- Ambient temperature, ˚K
$T_a(^\circ C)$	- Ambient temperature, ˚C.

$$\sigma_T = \frac{\delta}{\theta} = \frac{\delta}{\left(\frac{T_a}{T_{ssl}}\right)}$$ *eq 7.32*

Where:

σ_T	- Test density ratio
δ	- Pressure ratio
θ	- Temperature ratio
T_a	- Ambient temperature, ˚K
T_{ssl}	- Standard sea level temperature, 288.15 °K.

$$ESHP_T = K_Q\,(Q)\,(N_{R_T})$$ *eq 7.33*

Where:

$ESHP_T$	- Test engine shaft horsepower
K_Q	- Engine torque constant (unique for each aircraft)
Q	- Engine torque
N_{R_T}	- Test main rotor speed.

$$ESHP_{ref} = \frac{ESHP_T}{\sigma_T}\left(\frac{N_{R_S}}{N_{R_T}}\right)^3 \qquad \textit{eq 7.27}$$

$$W_T = ESGW - FU \qquad \textit{eq 7.34}$$

Where:

W_T	- Test weight
ESGW	- Engine start gross weight
FU	- Fuel used.

$$W_{ref} = \frac{W_T}{\sigma_T}\left(\frac{N_{R_S}}{N_{R_T}}\right)^2 \qquad \textit{eq 7.35}$$

Where:

W_{ref}	- Referred weight
W_T	- Test weight
σ_T	- Test density ratio
N_{R_S}	- Standard main rotor speed
N_{R_T}	- Test main rotor speed.

$$\Omega R = K_{GR}\left(N_{R_T}\right)\left(\frac{2\pi}{60}\right)(R) \qquad \textit{eq 7.36}$$

Where:

ΩR	- Rotational velocity
K_{GR}	- Gear ratio (unique to each helicopter)
N_{R_T}	- Test main rotor speed
R	- Rotor radius.

$$C_P = \frac{P}{\rho_a (A_D)(\Omega R)^3}$$ *eq 7.37*

Where:

C_P	- Power coefficient
P	- Power
ρ_a	- Ambient density
A_D	- Rotor disc area
ΩR	- Blade rotational velocity.

$$C_T = \frac{T}{\rho_a (A_D)(\Omega R)^2}$$ *eq 7.38*

Where:

C_T	- Thrust coefficient
T	- Thrust
ρ_a	- Ambient density
A_D	- Rotor disc area
ΩR	- Blade rotational velocity.

$$V_c = V_o + \Delta V_{ic} + \Delta V_{pos}$$ *eq 7.39*

Where:

V_c	- Calibrated airspeed
V_o	- Observed airspeed
ΔV_{ic}	- Airspeed instrument correction
ΔV_{pos}	- Airspeed position error.

$$V_T = \frac{V_c}{\sqrt{\sigma}} \qquad \textit{eq 7.40}$$

Where:

V_T - True airspeed

V_c - Calibrated airspeed

σ - Density ratio.

$$V_{T_{ref}} = V_T \left(\frac{N_{R_S}}{N_{R_T}} \right) \qquad \textit{eq 7.41}$$

Where:

$V_{T_{ref}}$ - Referred true airspeed

V_T - True airspeed

N_{R_S} - Standard main rotor speed

N_{R_T} - Test main rotor speed.

$$\mu = 1.69 \left(\frac{V_T}{\Omega R} \right) \qquad \textit{eq 7.42}$$

Where:

μ (Mu) - Advance ratio

V_T - True airspeed

ΩR - Blade rotational velocity.

$$a = 38.967 \sqrt{T_a \ (°K)} \qquad \textit{eq 7.43}$$

Where:

a - Speed of sound

T_a - Ambient temperature.

$$V_{TIP} = \left(\frac{\Omega R}{1.69}\right) + V_T \qquad \textit{eq 7.44}$$

Where:

V_{TIP} - Blade tip velocity

ΩR - Blade rotational velocity

V_T - True airspeed.

$$M_{TIP} = \frac{V_{TIP}}{a} \qquad \textit{eq 7.45}$$

Where:

M_{TIP} - Blade tip Mach number

V_{TIP} - Blade tip velocity

a - Speed of sound.

From observed level flight performance data, compute $ESHP_{ref}$, W_{ref}, $V_{T_{ref}}$, C_P, C_T, M_{TIP}, and μ as follows:

Parameter	Notation	Formula	Units	Remarks
(1) Observed pressure altitude	H_{P_o}		ft	
(2) Calibrated pressure altitude	H_{P_c}	Eq 7.29	ft	Instrument & position corrections
(3) Observed temperature	T_o		°C	
(4) Ambient temperature	T_a	Eq 7.30	°C	Instrument correction (ignore recovery and Mach correction)
(5) Ambient temperature	T_a	Eq 7.31	°K	Conversion

(6) Pressure ratio	δ			From tables at step (2)
(7) Test density ratio	σ_T	Eq 7.32		
(8) Test main rotor speed	N_{R_T}		%	
(9) Engine torque	Q		%, psi	Total torque
(10) Test ESHP	$ESHP_T$	Eq 7.33	hp	
(11) Referred ESHP	$ESHP_{ref}$	Eq 7.27	hp	
(12) Fuel counts	FC		cts	
(13) Fuel used	FU	FC (gal/ct) (lb/gal)	lb	gal/ct - Lab calibration, lb/gal - fuel density
(14) Test weight	W_T	Eq 7.34	lb	
(15) Referred weight	W_{ref}	Eq 7.35	lb	
(16) Rotational velocity	ΩR	Eq 7.36	ft/s	
(17) Power coefficient	C_P	Eq 7.37		
(18) Thrust coefficient	C_T	Eq 7.38		
(19) Observed airspeed	V_o		KOAS	
(20) Calibrated airspeed	V_c	Eq 7.39	KCAS	Instrument & position correction
(21) True airspeed	V_T	Eq 7.40	KTAS	
(22) Referred true airspeed	$V_{T_{ref}}$	Eq 7.41	KTAS	
(23) Advance ratio	μ	Eq 7.42		

(24) Speed of sound	a	Eq 7.43	
(25) Tip speed	V_{TIP}	Eq 7.44	KTAS
(26) Blade tip Mach number	M_{TIP}	Eq 7.45	

Plot $ESHP_{ref}$ as a function of $V_{t_{ref}}$ (referred level flight performance) (Figure 7.13).

Plot % W_{ref} deviation from average versus $V_{T_{ref}}$ (Figure 7.14).

Plot the following nondimensional level flight performance parameters, C_P versus μ (Figure 7.15), and M_{TIP} versus μ (Figure 7.16).

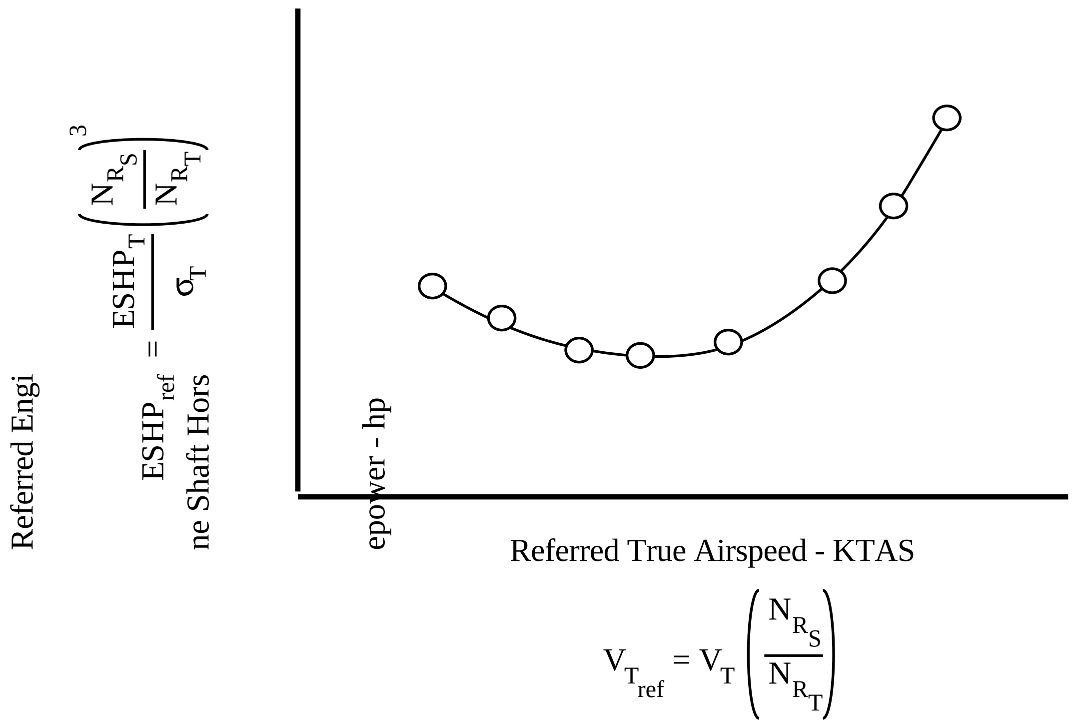


Figure 7.13
Referred Level Flight Performance

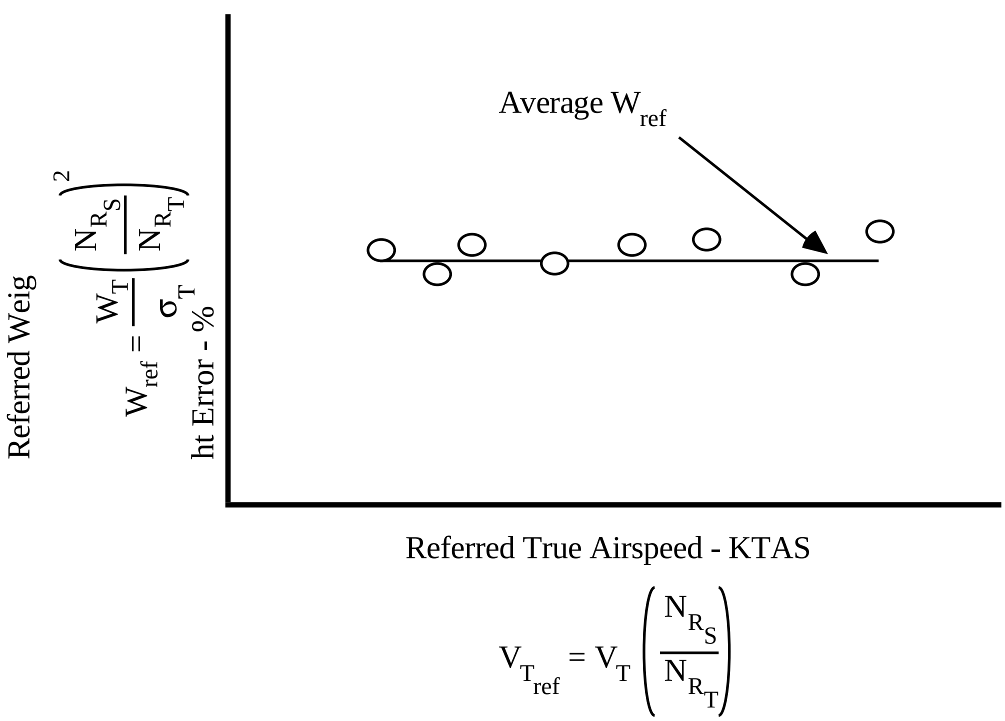


Figure 7.14
Referred Weight Deviation

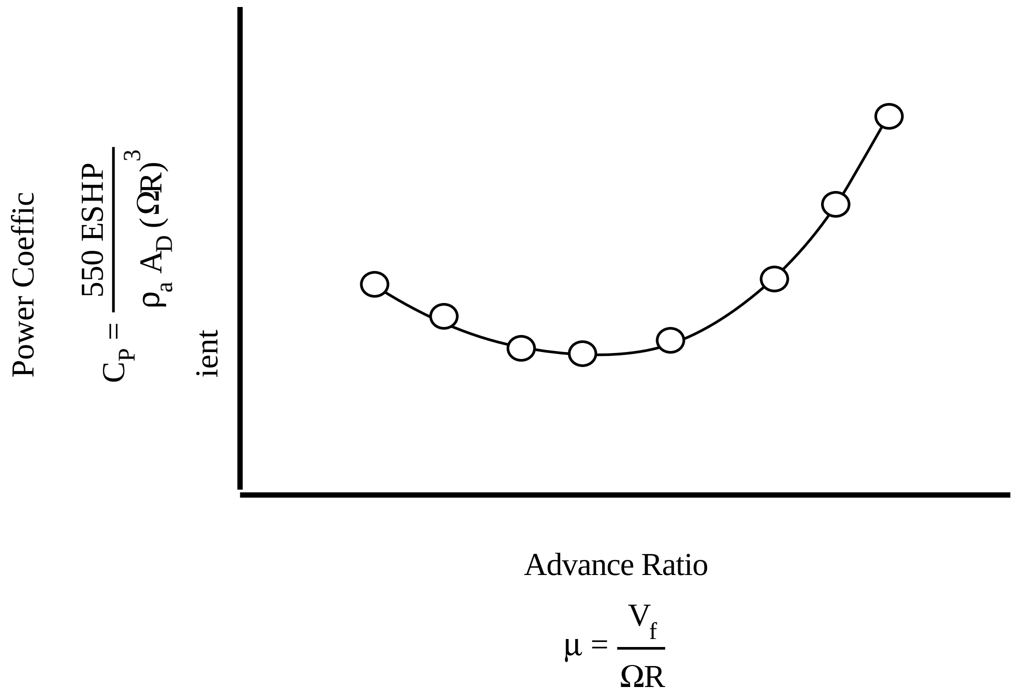


Figure 7.15
Power Coefficient Versus Advance Ratio

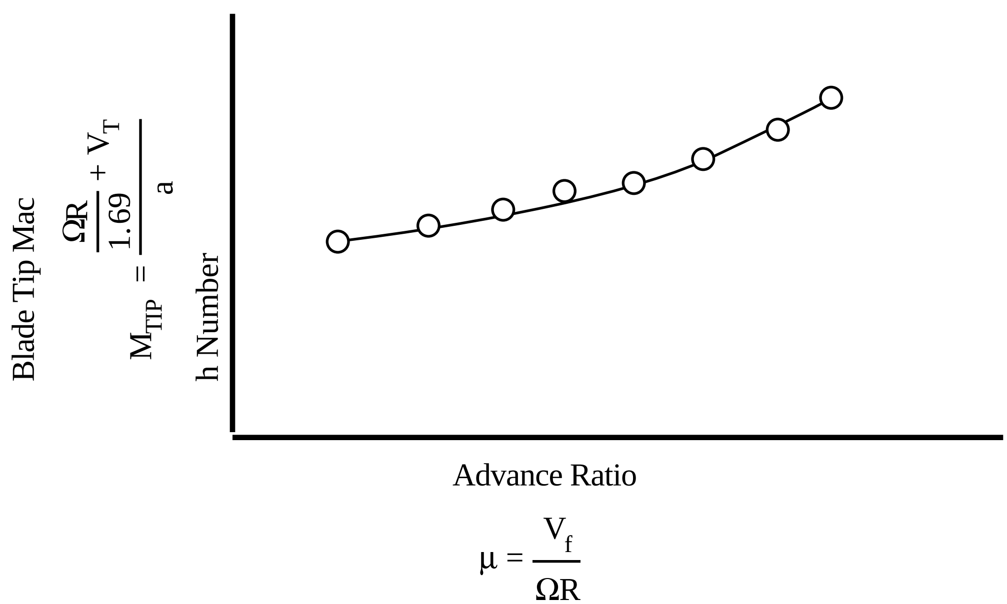


Figure 7.16
Blade Tip Mach Number Versus Advance Ratio

7.5.3 USNTPS Computer Data Reduction

This section contains a brief summary of the assumptions and logic used in the USNTPS helicopter WSIGMA (level flight) computer data reduction program.

Level flight performance is assumed to be in the form:

$$ESHP_{ref} = f\left(W_{ref}, V_{T_{ref}}\right) \qquad \textit{eq 7.46}$$

Where:

$ESHP_{ref}$	- Referred engine shaft horsepower
W_{ref}	- Referred weight
$V_{T_{ref}}$	- Referred true airspeed.

Specifically $ESHP_{ref}$ is assumed to be a fourth order expression in $V_{T_{ref}}$:

$$ESHP_{ref} = E_0 + E_1 V_{T_{ref}} + E_2 V_{T_{ref}}^2 + E_3 V_{T_{ref}}^3 + E_4 V_{T_{ref}}^4 \qquad \textit{eq 7.47}$$

Where:

$ESHP_{ref}$	- Referred engine shaft horsepower
E_0, E_1, E_2, E_3, E_4	- Constants
$V_{T_{ref}}$	- Referred true airspeed.

The following equations are used in the calculations:

$$ESHP_{ref} = \frac{ESHP_T}{\sigma_T}\left(\frac{N_{R_S}}{N_{R_T}}\right)^3 \qquad \textit{eq 7.27}$$

$$W_{ref} = C_T\, \rho_{ssl}\, A_D \left(\Omega R_S\right)^2 = \frac{W_T}{\sigma_T}\left(\frac{N_{R_S}}{N_{R_T}}\right)^2 \qquad \textit{eq 7.25}$$

$$ESHP_T = K_Q\,(Q)\,(N_{R_T}) \qquad \textit{eq 7.33}$$

$$\delta = \frac{P_a}{P_{ssl}} = \left[1 - \left(\frac{\lambda_{ssl}\,H_P}{T_{ssl}}\right)\right]^{\frac{g_{ssl}}{g_c\,\lambda_{ssl}\,R}} \qquad \textit{eq 7.48}$$

Where:

δ	- Pressure ratio
P_a	- Ambient pressure
P_{ssl}	- Standard sea level pressure, 2116.2 psf
T_{ssl}	- Standard sea level temperature, 288.15 °K
H_P	- Pressure altitude, ft
λ_{ssl}	- Standard sea level lapse rate, 0.00198 °K/ft
g_{ssl}	- Standard sea level gravity, 32.17 ft/ s^2
g_c	- 32.17 lb - mass / slug
R	- Engineering gas constant for air, 96.03 ft - lb/ lb_m- °K.

$$\theta = \frac{T_a}{T_{ssl}} = \frac{OAT + 273.15}{288.15} \qquad \textit{eq 7.49}$$

Where:

θ	- Temperature ratio
T_a	- Ambient temperature
T_{ssl}	- Standard sea level temperature, 288.15 °K
OAT	- Outside air temperature (total temperature).

$$\sigma = \frac{\rho}{\rho_{ssl}} = \frac{\delta}{\theta} \qquad \text{eq 7.50}$$

Where:

σ - Density ratio

ρ - Density

ρ_{ssl} - Standard sea level density, 0.0023769 slug/ ft^3

δ - Pressure ratio

θ - Temperature ratio.

$$W_T = ESGW - FU \qquad \text{eq 7.34}$$

$$V_T = \frac{V_c}{\sqrt{\sigma_T}} \qquad \text{eq 7.51}$$

Where:

V_T - True airspeed

V_c - Calibrated airspeed

σ_T - Test density ratio.

$$V_{T_{ref}} = \mu\left(\Omega R_T\right) = V_T\left(\frac{N_{R_S}}{N_{R_T}}\right) \qquad \text{eq 7.26}$$

$$C_P = \frac{P}{\rho_a A_D (\Omega R)^3} \qquad \text{eq 7.37}$$

$$C_T = \frac{T}{\rho_a A_D (\Omega R)^2} \qquad \text{eq 7.38}$$

$$\Omega R = K_{GR}\left(N_{R_T}\right)\left(\frac{2\pi}{60}\right)(R) \qquad \textit{eq 7.36}$$

$$\mu = 1.69\left(\frac{V_T}{\Omega R}\right) \qquad \textit{eq 7.42}$$

$$M_{TIP} = 1.68781 V_T + \left(\frac{\Omega R_T}{\sqrt{\gamma\, g_c\, R\, T_{ssl}\, \theta}}\right) \qquad \textit{eq 7.52}$$

Where:

M_{TIP}	- Blade tip Mach number
V_T	- True airspeed
ΩR_T	- Test rotational velocity
γ	- Ratio of specific heats (air = 1.4)
g_c	- 32.17 lb - mass / slug
R	- Engineering gas constant for air, 96.03 ft - lb/ lb_m- °K
T_{ssl}	- Standard sea level temperature
θ	- Temperature ratio.

To assist in fitting the fourth order curve, the zero airspeed power required point is calculated from hover performance data. The user inputs hover performance in the form of the coefficients from the HOVER performance data reduction:

$$ESHP_{ref} = A_0 + A_1\, W_{ref}^{\frac{3}{2}} \qquad \textit{eq 7.53}$$

Where:

$ESHP_{ref}$	- Referred engine shaft horsepower
A_0, A_1	- Constants
W_{ref}	- Referred weight.

To determine fuel flow, user inputs corrected fuel flow data from the ENGINE program in the form of the coefficients of the following:

$$W_{f_{corr}} = B_0 + B_1\ ESHP_{corr} + B_2\ ESHP^2_{corr} + B_3\ ESHP^3_{corr} \qquad \textit{eq 7.54}$$

Where:

$W_{f_{corr}}$	- Corrected fuel flow
B_0, B_1, B_2, B_3	- Constants
$ESHP_{corr}$	- Corrected engine shaft horsepower.

To determine power available, user inputs inlet performance data from the ENGINE program in the form of the coefficients for the following equations:

$$\frac{P_{T_2}}{P_a} = C_0 + C_1\ V_c + C_2\ V_c^2 + C_3\ V_c^3 \qquad \textit{eq 7.55}$$

Where:

P_{T_2}	- Inlet total pressure
P_a	- Ambient pressure
C_0, C_1, C_2, C_3	- Constants
V_c	- Calibrated airspeed.

$$T_{T_2} - T_a = D_0 + D_1\ V_c + D_2\ V_c^2 + D_3\ V_c^3 \qquad \textit{eq 7.56}$$

Where:

T_{T_2}	- Inlet total temperature
T_a	- Ambient temperature
D_0, D_1, D_2, D_3	- Constants
V_c	- Calibrated airspeed.

$ESHP_{ref}$, W_{ref}, $V_{T_{ref}}$, C_T, C_P, μ, and M_{TIP}, are output as tabular data and graphic data in the form of Figures 7.13, 7.14, 7.15, and 7.16.

The user specifies a pressure altitude and main rotor speed of interest, usually sea level and normal operating main rotor speed. ESHP, V_T, W_f + 5%, SR for the corresponding weight is extracted from the referred data and output in tabular and graphic form as Figures 7.17, 7.18, and 7.19.

The weight corresponding to the selected H_P and N_R is computed as follows:

$$W_{ref} = C_T \, \rho_{ssl} \, A_D \left(\Omega R_S\right)^2 = \frac{W_T}{\sigma_T}\left(\frac{N_{R_S}}{N_{R_T}}\right)^2 \qquad \textit{eq 7.25}$$

The specific range is computed using the 5% margin as follows:

$$SR = \frac{V_T}{1.05 \, W_f} \qquad \textit{eq 7.57}$$

Where:

SR	- Specific range
V_T	- True airspeed
W_f	- Fuel flow.

7.6 DATA ANALYSIS

7.6.1 General

Observed level flight performance data are reduced to nondimensional and generalized (also called referred) level flight performance data. These data are analyzed to determine level flight performance, endurance performance, and range performance.

7.6.2 Level Flight Performance

Level flight performance in the form of power required as a function of true airspeed can be extracted manually from referred level flight and engine performance in much the same manner as the USNTPS WSIGMA computer data reduction.

The following equations are used in the manual data analysis of level flight performance:

$$W = \frac{W_{ref}\,(\sigma)}{\left(\dfrac{N_{R_S}}{N_R}\right)^2} \qquad \textit{eq 7.58}$$

Where:

- W - Weight
- W_{ref} - Referred weight
- σ - Density ratio
- N_{R_S} - Standard main rotor speed
- N_R - Main rotor speed.

$$V_T = \frac{V_{T_{ref}}}{\left(\dfrac{N_{R_S}}{N_R}\right)} \qquad \textit{eq 7.59}$$

Where:

- V_T - True airspeed
- $V_{T_{ref}}$ - Referred true airspeed
- N_{R_S} - Standard main rotor speed
- N_R - Main rotor speed.

$$\text{ESHP} = \frac{\text{ESHP}_{\text{ref}}\,(\sigma)}{\left(\frac{N_{R_S}}{N_R}\right)} \qquad \textit{eq 7.60}$$

Where:

ESHP	- Engine shaft horsepower
ESHP_{ref}	- Referred engine shaft horsepower
σ	- Density ratio
N_{R_S}	- Standard main rotor speed
N_R	- Main rotor speed.

Determine ESHP as a function of V_T. Select pressure altitude and main rotor speed of interest. Sea level, standard day, and standard operating rotor speed are conditions of interest.

Parameter	Notation	Formula	Units	Remarks
(1) Main rotor speed	N_R			Condition of interest
(2) Pressure altitude	H_P		ft	Condition of interest
(3) Density ratio	σ			From tables at step (2)
(4) Referred weight	W_{ref}		lb	From Figure 7.13
(5) Weight	W	Eq 7.58	lb	
(6) Referred true airspeed	$V_{T_{ref}}$		KTAS	From Figure 7.13
(7) True airspeed	V_T	Eq 7.59	KTAS	
(8) Referred engine shaft horsepower	ESHP_{ref}		hp	Corresponding to $V_{T_{ref}}$ (Figure 7.13)

(9) Engine shaft horsepower	ESHP	Eq 7.60	hp

(10) Repeat as necessary to cover range of ESHP and V_T.

(11) Determine engine shaft horsepower available as a function of V_T from Figure 4.11.

Plot level flight performance, engine shaft horsepower (ESHP), required and available, as a function of true airspeed (V_T) (Figure 7.17).

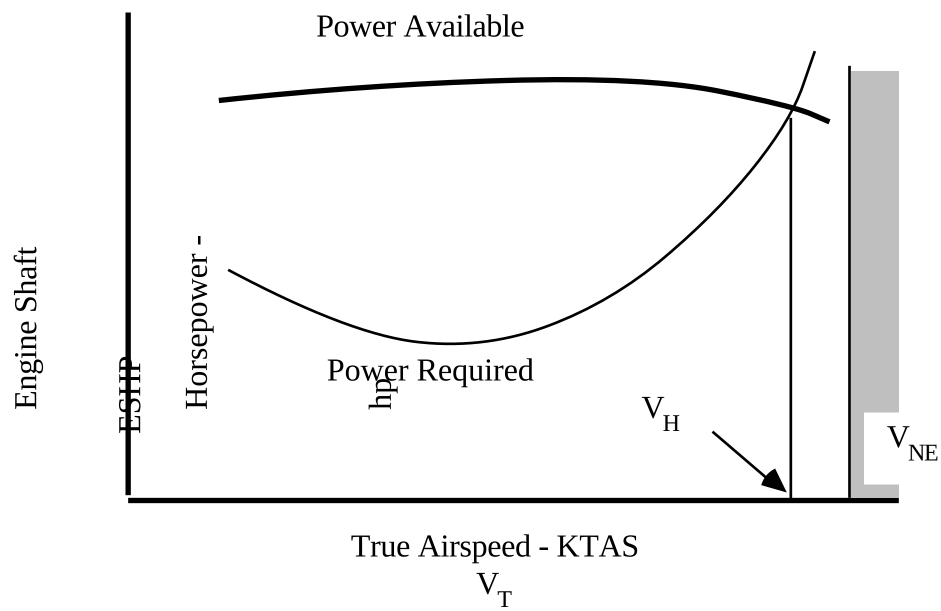


Figure 7.17
Level Flight Performance

The aircraft maximum level flight airspeed, V_H, (airframe and engine dependent) is established as the airspeed corresponding to the intersection of the power required and the power available curves.

7.6.3 Endurance

The following equations are used in the analysis of aircraft endurance:

$$V_c = V_T \sqrt{\sigma} \qquad \text{eq 7.61}$$

Where:

V_c - Calibrated airspeed

V_T - True airspeed

σ - Density ratio.

$$ESHP_{corr} = \frac{ESHP}{\delta_{T_2}\sqrt{\theta_{T_2}}} \qquad \text{eq 7.62}$$

Where:

$ESHP_{corr}$ - Corrected engine shaft horsepower

ESHP - Engine shaft horsepower

δ_{T_2} - Inlet pressure ratio

θ_{T_2} - Inlet temperature ratio.

$$W_f = W_{f_{corr}}\left(\delta_{T_2}\right)\sqrt{\theta_{T_2}} \qquad \text{eq 7.63}$$

Where:

W_f - Fuel flow

$W_{f_{corr}}$ - Corrected fuel flow

δ_{T_2} - Inlet pressure ratio

θ_{T_2} - Inlet temperature ratio.

Fuel flow as a function of true airspeed is determined from level flight and engine performance as follows:

Parameter	Notation	Formula	Units	Remarks
(1) Pressure altitude	H_P			User selected
(2) True airspeed	V_T		KTAS	From Figure 7.17
(3) Density ratio	σ			From tables at (1)
(4) Calibrated airspeed	V_c	Eq 7.61	KCAS	
(5) Inlet pressure ratio	δ_{T_2}			From Figure 4.3 for V_c
(6) Inlet temperature ratio	θ_{T_2}			From Figure 4.4 for V_c
(7) Engine shaft horsepower	ESHP		hp	From Figure 7.17 for V_T
(8) Corrected engine shaft horsepower	$ESHP_{corr}$	Eq 7.62	hp	
(9) Corrected fuel flow	$W_{f_{corr}}$		lb/h	From Figure 4.8 for $ESHP_{corr}$
(10) Fuel flow	W_f	Eq 7.63	lb/h	
(11) Fuel flow plus 5 %	W_f + 5%	1.05 W_f	lb/h	

(12) Repeat as necessary to cover the range of V_T.

Plot endurance in the form of fuel flow plus 5% (1.05 W_f) as a function of true airspeed (V_T) (Figure 7.18).

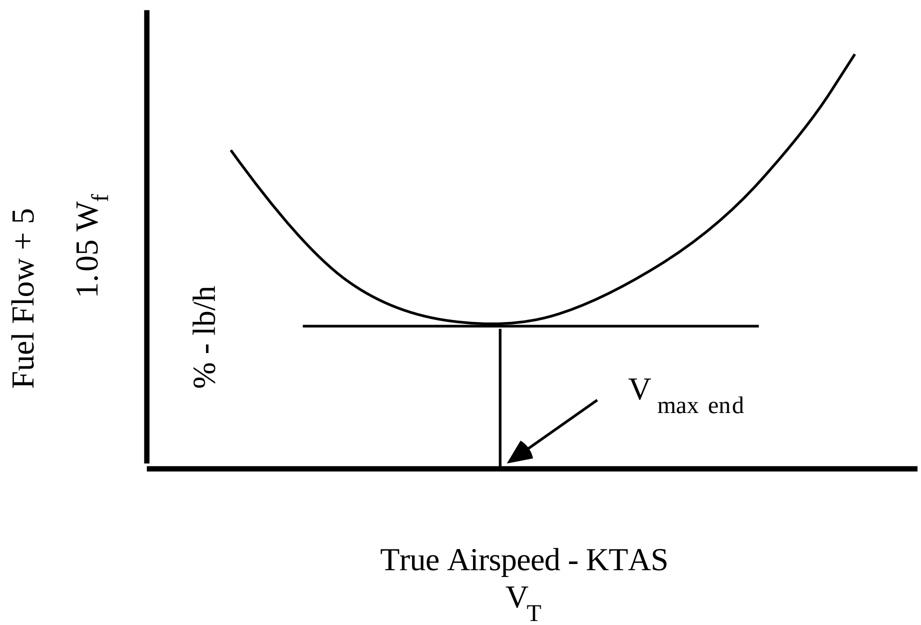


Figure 7.18
Endurance Performance

The maximum endurance is found at the minimum fuel flow point. The maximum endurance in hours is determined as follows:

$$\text{Endurance} = \frac{\text{Fuel available}}{1.05\ W_f}$$ *eq 7.64*

Where:

Endurance	- Maximum endurance in hours
Fuel available	- Maximum fuel available in lb
W_f + 5%	- Minimum fuel flow in lb/h.

The maximum endurance airspeed, $V_{max\ end}$, is the airspeed which corresponds to the minimum fuel flow. This airspeed is called the bucket airspeed on the fuel flow curve.

Bear in mind that not all helicopter detail specifications require the 5% increase in fuel flow as a tolerance for service conditions.

7.6.4 Range

The following equation is used in the analysis of aircraft range:

$$SR = \frac{V_T}{1.05\ W_f}$$

eq 7.57

Aircraft range is determined from airframe and engine performance in the form of specific range, SR, as a function of true airspeed, V_T:

Parameter	Notation	Formula	Units	Remarks
(1) True airspeed	V_T		KTAS	From Figure 7.18
(2) Fuel flow plus 5%	$1.05\ W_f$		lb/h	From Figure 7.18 for (1)
(3) Specific range	SR	Eq 7.57	nmi/lb	
(4) Repeat as necessary to cover range of true airspeed.				

Plot specific range, SR, as a function of true airspeed, V_T, (Figure 7.19).

The maximum specific range occurs at the maximum of the SR curve.

The airspeed corresponding to the maximum SR is the airspeed for maximum range, $V_{max\ range}$.

The airspeed corresponding to the greater of 99% maximum SR is the cruise airspeed, V_{cruise}.

Combat radius is calculated from the data in Figures 7.18 and 7.19 in accordance with the flight profile contained in MIL-C-5011A.

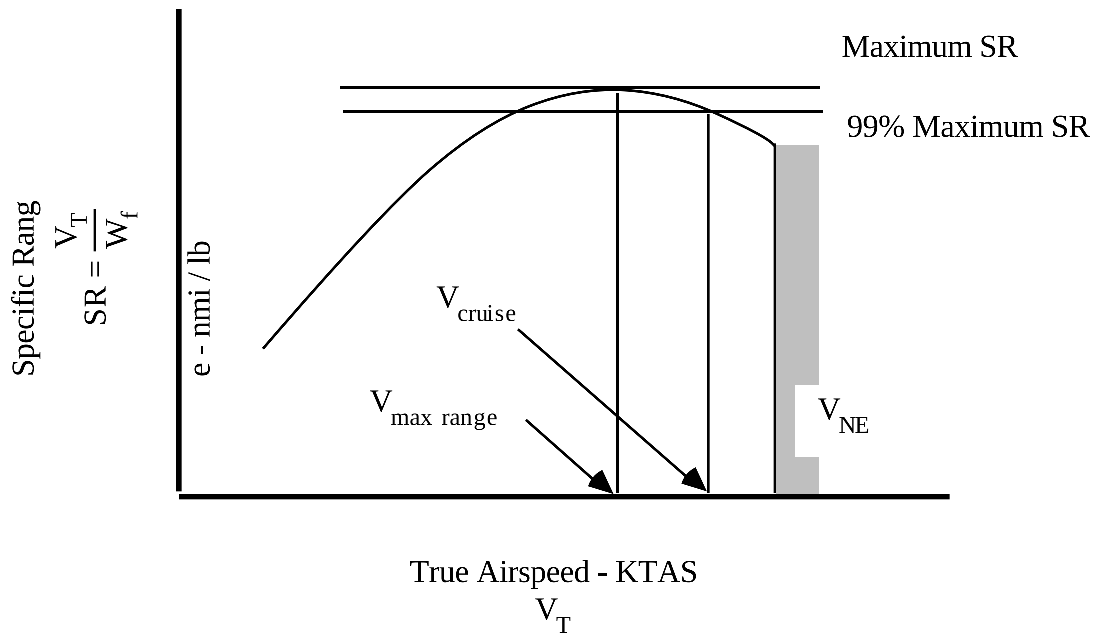


Figure 7.19
Specific Range Performance

Maximum range is calculated from the data in Figures 7.18 and 7.19 using the definitions in MIL-C-5011A.

7.7 MISSION SUITABILITY

7.7.1 General

Service suitability conclusions concerning level flight performance should include, but not be limited to performance test results and specification conformance. Test results reflect the performance capabilities of the aircraft, while service suitability conclusions also include the flying qualities associated with specific airspeeds. Consider the following items when recommending an airspeed for maximum endurance and range:

1. Vibration level.
2. Flight path stability.
3. Attitude.
4. Field of view.
5. Pilot workload to maintain desired/recommended speed.

In general, level flight mission suitability performance testing should begin with realistic mission profiles. What environment will the end user be expected to operate in? Is this aircraft expected to operate in conjunction with other aircraft types, and are the airspeeds compatible? What threat environment is the aircraft expected to operate in? Is the aircraft expected to operate from aboard ship, and will this necessitate higher minimum on-deck fuel states and periods of flight at maximum endurance airspeed waiting for clearance to land? Is the combat radius and combat range sufficient to support the mission profile with the expected mission payload? These and other questions must be answered based on the requirements of mission suitability, independent of the requirements of the specification.

7.7.2 Vibrations

Vibrations, their source and attenuation, are a very complex subject. A complete treatment is not given here. It is important, however, in performance testing for the pilot to be aware of the primary variables involved and to evaluate the vibration level present in the cockpit. Vibrations affect mission capability; e.g., attack helicopter tracking, fatigue during demanding tasks, medical evacuation of combat casualties, etc.

7.7.2.1 REVERSE FLOW, BLADE STALL

The retreating rotor blade in high-speed flight is subject to an aerodynamic condition commonly known as “reverse flow”. In addition to the reverse flow phenomenon, a portion of the retreating blade is subject to blade stall. Reverse flow occurs at the inboard edge of the rotor while blade stall occurs at the outboard edge of the rotor. The combination of stall and reverse flow acts as an excitation force. Depending on the blade natural frequencies and the number of blades, a force, cyclic in nature, is transmitted through the rotor head, transmission, and airframe. Increased rotor speed should reduce the vibration level in this situation. If the problem does not abate, the cause may be a structural one of blade-airframe damping at the given excitation frequency or one of compressibility.

7.7.2.2 NATURAL NULL

If the vibration is associated with rotor speed but not attributable to blade stall or reverse flow, the problem may be operating out of the "natural null" of the aircraft. The natural null is determined in the cockpit as that rotor speed which produces the smoothest flight. Some aircraft incorporate attenuating devices which are intended to minimize cockpit vibrations at the design operational rotor speed. These devices must be tuned or they become a forcing function and drive the aircraft to higher vibration levels than the basic aircraft. The rotor speed corresponding to the natural null is determined by conducting a rotor speed sweep at an airspeed where rotor roughness is pronounced and noting the rotor speed for the smoothest flight. The two most important aspects of the test results are:

1. The proximity of the natural null to the operational rotor speed.
2. Movement of the natural null as the aircraft accrues flight time or as the gross weight changes.

7.7.2.3 COMPRESSIBILITY (MACH EFFECTS)

The tip of the advancing blade may encounter high Mach numbers at relatively high forward speeds when operating in cold atmospheric conditions. Mach numbers of 0.8 and higher are most significant as they result in a rapid drag rise and loss of lift on the advancing blade. The loss of lift causes an unbalanced rotor condition which acts as a forcing function for the airframe spring mass system.

7.7.2.4 PILOT REPORTING

The test pilot must discuss the vibrations encountered without benefit of instrumentation. The pilot's report may initiate the first requirement for such instrumentation. Vibrations are reported with respect to their frequency, relative amplitudes, airspeeds where encountered, configurations where encountered, and physiological effects.

Vibrations affect different areas of the cockpit at the same time but in different ways. Generally, the displacement of the pilot, instrument panel, instrument pointer, etc., depend upon the natural frequency of the individual spring mass system. Report the apparent frequency of the oscillation of interest and indicate the apparent source. A one per rev vibration will be most apparent in the pilot's seat and in the instrument panel.

Vibratory frequencies, from one per rev to one per blade passage, are relatively easy to identify. Vibratory amplitudes can be compared by stating the period of time that the pilot would be willing to endure the flight condition, 5 minutes, 30 minutes, or the complete mission. Medium to high frequency vibrations are normally detected by touch with the foot or hand and sometimes can be visually observed in the airframe or instruments. In these cases, the pilot should report what has been observed.

When reporting vibrations experienced in flight, it is advisable to define what is meant by the terms used. The following table is a suggested guide (Table 7.I).

Table 7.I
Vibration Rating Scale

Degree of Vibration	Description	Pilot Rating
No Vibration		0
Slight	Not apparent to experienced aircrew fully occupied by their tasks, but noticeable if their attention is directed to it or if not otherwise occupied.	1 2 3
Moderate	Experienced aircrew are aware of the vibration, but it does not affect their work, at least over a short period.	4 5 6
Severe	Vibration is immediately apparent to experienced aircrew even when fully occupied. Performance of primary task is affected or tasks can be done only with difficulty.	7 8 9
Intolerable	Sole preoccupation of aircrew is to reduce vibration.	10

Based on the Subjective Vibration Assessment Scale developed by the Aeroplane and Armament Experimental Establishment, Boscombe Down, England.

Realize the test pilot can only identify a problem area. A full vibration survey and consultation with flight test engineers is required to fully define the problem and its effect.

7.8 SPECIFICATION COMPLIANCE

7.8.1 General

Level flight performance guarantees are stated in the detail specification for the model and Military Specification MIL-C-5011A. The detail specification generally states guarantees for the following items:

1. Cruising speed (KTAS)
2. Maximum level flight speed (KTAS)
3. Maximum endurance (h).
4. Operating radius at cruising speed (nmi).

MIL-C-5011A further defines requirements for and methods of presenting characteristics and performance data for U.S. military piloted aircraft. Deviations from this specification are permissible, but in all cases must be approved by the procuring activity. Generally level flight performance guarantees are stated for sea level, standard day conditions and mission or takeoff gross weight.

7.8.2 Cruising Speed

Cruising speed defined as, in MIL-C-5011A,: “The airspeed for long-range operation shall be the greater of the two speeds at which 99 percent of the maximum miles per pound of fuel are attainable at the momentary weight and altitude.” Generally the detail specification will limit the cruising speed to that obtainable with normal rated power.

7.8.3 Maximum Level Flight Speed

The maximum level flight airspeed, as defined in MIL-C-5011A, is: “The highest speed attainable in level flight, at a given weight, altitude, and engine power rating. Such speeds shall be within all operating restrictions.” Power ratings are stated in the detail specification as military or normal rated power.

7.8.4 Maximum Endurance

Maximum endurance is the time in hours an aircraft can maintain flight at a particular airspeed and initial loading conditions. The airspeed for maximum endurance operation, as defined in MIL-C-5011A, is: “The airspeed for maximum endurance shall correspond to the speed for minimum fuel flow at a momentary gross weight and altitude except as limited by acceptable handling characteristics of the aircraft.” The gross weight and useful fuel load corresponding to the maximum endurance time will include calculations for start and warm-up and a given percentage of the initial load as a reserve. The time required for start and warm-ups and the percentage of fuel required as a reserve are defined in MIL-C-5011A for the particular mission and may further be defined in the detail specification.

7.8.5 Operating Radius (Combat Radius)

Operating radius (combat radius), as defined in MIL-C-5011A, is: “The distance attainable on a practicable flight to the target and return a distance equal to that flown out, carrying a specific load to or from the target according to a sequence of operations.” When computing the operating radius, the following items shall be included:

1. Start and warm-up fuel required for a given amount of time and at a given engine power rating.
2. Climb to specific altitude using up to maximum power.
3. Cruise to remote base at cruising speed at a specific altitude.
4. Spend specific amount of time over remote base where stores may be expended.
5. Climb and cruise to home base.
6. Descent at home base (no range credit).
7. Retain specific percentage of fuel allowance for a reserve.

The above items are stated in MIL-C-5011A for each type aircraft and may be modified in the detail specification. All speeds shall be level flight true airspeeds unless otherwise explained or noted. Additionally, MIL-C-5011A states: “Fuel consumption data, regardless of source, shall be increased by 5 percent for all engine power conditions as a service tolerance to allow for practicable operations.” Some helicopter detail specification specifically do not incorporate a 5 % fuel consumption margin.

7.9 GLOSSARY

7.9.1 Notations

a	Speed of sound	kn
a_{ssl}	Standard sea level speed of sound	661.23 kn
A_0, A_1	Constants	
A_D	Rotor disc area	ft^2
A_f	Frontal area	ft^2
b	Number of blades	
B_0, B_1, B_2, B_3	Constants	
c	Blade chord	ft
C_0, C_1, C_2, C_3	Constants	
C_{d_0}	Blade element profile drag coefficient	
$\overline{C}_{d_0}$	Average blade element profile drag coefficient	
C_{D_0}	Profile drag coefficient	
C_{D_p}	Parasite drag coefficient	
C_L	Lift coefficient	
CG	Center of gravity	in
C_P	Power coefficient	
C_T	Thrust coefficient	
dD_0	Blade element profile drag	lb
dP	Blade element power required	lb
dP_0	Blade element profile power	lb
dr	Blade element radius	ft
dR	Blade element resultant aerodynamic force	lb
D_0, D_1, D_2, D_3	Constants	
D_{eff}	Effective drag	lb
D_p	Parasite drag	lb
E_0, E_1, E_2, E_3	Constants	
ESGW	Engine start gross weight	lb
ESHP	Engine shaft horsepower	hp

$ESHP_{corr}$	Corrected engine shaft horsepower	hp
$ESHP_{ref}$	Referred engine shaft horsepower	hp
$ESHP_S$	Standard engine shaft horsepower	hp
$ESHP_T$	Test engine shaft horsepower	hp
f	Equivalent flat plate area	ft^2
F	Force	lb
FC	Fuel counts	cts
FU	Fuel used	lb
g_{ssl}	Standard sea level gravity	32.17 ft/s^2
g_c	32.17 lb - mass / slug	
GW	Gross weight	lb
HOGE	Hover out-of-ground effect	
H_D	Density altitude	ft
H_P	Pressure altitude	ft
H_{P_c}	Calibrated pressure altitude	ft
H_{P_o}	Observed pressure altitude	ft
$\Delta H_{P_{ic}}$	Altimeter instrument correction	ft
ΔH_{pos}	Altimeter position error	ft
K_{GR}	Gear ratio (aircraft unique)	
K_Q	Engine torque constant (aircraft unique)	
ℓ_{TR}	Moment arm of the tail rotor	
m	Mass	lb m
M	Mach number	
M_{TIP}	Blade tip Mach number	
N_R	Main rotor speed	%, rpm
N_{R_S}	Standard main rotor speed	%, rpm
N_{R_T}	Test main rotor speed	%, rpm
OAT	Outside air temperature (total temperature)	°C
P_a	Ambient pressure	psi, psf
P_{ssl}	Standard sea level pressure	2116.2 psf

P_T	Total pressure (free stream)	psi, psf
P_{T_2}	Inlet total pressure	psi
P	Power	hp
P_{Acc}	Accessory power required	hp
P_i	Induced power	hp
P_{i_f}	Induced power in forward flight	hp
P_{loss}	Power loss	hp
P_{MR}	Main rotor power required	hp
P_m	Miscellaneous power	hp
P_0	Profile power	hp
$P_{0_{H2}}$	Profile power due to spanwise flow	hp
P_p	Parasite power	hp
P_{TOTAL}	Total power required	hp
P_{TR}	Tail rotor power required	hp
Q	Engine torque	%, psi
Q_{MR}	Main rotor torque	%, psi
Q_{TR}	Torque developed by the tail rotor	%, psi
r	Radius to blade element	ft
R	Engineering gas constant for air	96.03 ft - lb/ lb_m- °K
	Rotor radius	ft
RSHP	Rotor shaft horsepower	hp
$RSHP_{ref}$	Referred rotor shaft horsepower	hp
$RSHP_S$	Standard rotor shaft horsepower	hp
$RSHP_T$	Test rotor shaft horsepower	hp
SR	Specific range	nmi/lb
T	Temperature	°C, °K
T_a	Ambient temperature	°C
T_o	Observed temperature	°C
T_{ssl}	Standard sea level temperature	15°C, 288.15°K
T_{T_2}	Inlet total temperature	°C

ΔT_{ic}	Temperature instrument correction	°C
T	Thrust	lb
T_h	Thrust at a hover	lb
T_v	Thrust in a vertical climb	lb
T_f	Thrust in forward flight	lb
T_{TR}	Tail rotor thrust	lb
v_e	Element velocity	ft/s
v_i	Induced velocity at hover	ft/s
v_{i_f}	Induced velocity in forward flight	ft/s, ft/min
v_{i_v}	Induced velocity in a vertical climb	ft/s, ft/min
v_w	Ultimate wake velocity	ft/s
V	Velocity	kn
V_c	Calibrated airspeed	KCAS
V_{cruise}	Cruise airspeed	kn
V_f	Forward flight velocity	kn
V_H	Maximum level flight airspeed	kn
V_{max}	Airframe limited level flight airspeed	kn
$V_{max\ end}$	Maximum endurance airspeed	kn
$V_{max\ range}$	Maximum range airspeed	kn
V_{NE}	Never exceed airspeed	kn
V_o	Observed airspeed	KOAS
$V_{(P/L)min}$	Airframe maximum range airspeed	kn
V_R	Resultant velocity through rotor	ft/s
V_T	True airspeed	KTAS
$V_{T_{ref}}$	Referred true airspeed	KTAS
V_{TIP}	Blade tip velocity	ft/s
V_v	Actual vertical velocity	ft/min
ΔV	Change in velocity	ft/s, ft/min, kn
ΔV_{ic}	Airspeed instrument correction	kn
ΔV_{pos}	Airspeed position error	kn

W	Weight	lb
W_{ref}	Referred weight	lb
W_S	Standard weight	lb
W_T	Test weight	lb
W_f	Fuel flow	lb/h
$W_{f_{corr}}$	Corrected fuel flow	lb/h

7.9.2 Greek Symbols

δ (Delta)	Pressure ratio	
δ_{T_2}	Inlet pressure ratio	
η_m (Eta)	Mechanical efficiency	
γ (Gamma)	Ratio of specific heat (Air: $\delta = 1.4$)	
λ_{ssl} (Lambda)	Standard sea level lapse rate	0.00198 °K/ft
μ (Mu)	Advance ratio	
θ (Theta)	Temperature ratio	
θ_{T_2}	Inlet temperature ratio	
ρ (Rho)	Density	slug/ft^3
ρ_a	Ambient air density	slug/ft^3
ρ_S	Standard density	slug/ft^3
ρ_T	Test density	slug/ft^3
ρ_{ssl}	Standard sea level air density	0.0023769 slug/ft^3
σ_i (Sigma)	Induced angle	
σ	Density ratio	
σ_T	Test density ratio	
σ_R	Rotor solidity ratio	

Ω (Omega)	Rotor angular velocity	rad/s
Ωr	Blade element rotational velocity	ft/s
ΩR	Blade rotational velocity	ft/s
ΩR_S	Standard blade rotational velocity	ft/s
ΩR_T	Test blade rotational velocity	ft/s
ψ (psi)	Blade azimuth angle	deg

7.10 REFERENCES

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CHAPTER EIGHT

CLIMB AND DESCENT PERFORMANCE

CHAPTER EIGHT

FIGURES

CHAPTER EIGHT

TABLES

CHAPTER EIGHT

EQUATIONS

$$R/C = \frac{(ESHP_A - ESHP_{req})\ (33{,}000)}{W}$$ *eq 8.1*

$$\frac{Power_C - Power_A}{Power_B - Power_A} = \frac{(R/C)_C}{(R/C)_B}$$ *eq 8.2*

$$W = \rho\ A_D\ V_R\ (2v_{i_v})$$ *eq 8.3*

$$V_R = \sqrt{V_f^2 + (V_V + v_{i_v})^2}$$ *eq 8.4*

$$v_i = \sqrt{\frac{W}{2\ \rho\ A_D}}$$ *eq 8.5*

$$v_i^2 = v_{i_v}\sqrt{V_f^2 + (V_V + v_{i_v})^2}$$ *eq 8.6*

$$\frac{v_i}{v_{i_v}} = \sqrt{\left(\frac{V_f}{v_i}\right)^2 + \left(\frac{V_V}{v_i} + \frac{v_{i_v}}{v_i}\right)^2}$$ *eq 8.7*

$$P_{climb} \approx \left(P_i + P_o + P_p\right)_{level\ flight} + \frac{WV'}{33{,}000}$$ *eq 8.8*

$$V' = \frac{\left(P_{climb} - P_{level\ flight}\right)}{W}\ (33{,}000)$$ *eq 8.9*

$$\tan\varphi = \frac{V_V}{\Omega r}$$ *eq 8.10*

$$\varphi = \tan^{-1}\left(\frac{V_V}{\Omega r}\right)$$ *eq 8.11*

$$\tan \varphi' = \frac{D}{L} = \frac{C_D}{C_L}$$ *eq 8.12*

$$\varphi' = \tan^{-1}\left(\frac{C_D}{C_L}\right)$$ *eq 8.13*

$$\alpha = \theta + \varphi$$ *eq 8.14*

$$\varphi' = f(\alpha)$$ *eq 8.15*

$$P = \frac{d(PE)}{dt}$$ *eq 8.16*

$$P = W\left(\frac{dh}{dt}\right)$$ *eq 8.17*

$$P = W V_V$$ *eq 8.18*

$$K_{H_P} = \frac{T_a + 273.15}{T_S + 273.15}$$ *eq 8.19*

$$H_{P_c} = H_{P_o} + \Delta H_{P_{ic}} + \Delta H_{pos}$$ *eq 8.20*

$$T_a = T_o + \Delta T_{ic}$$ *eq 8.21*

$$\sigma_T = \frac{\delta}{\theta} = \frac{\delta}{\left(\frac{T_a}{T_{ssl}}\right)}$$ *eq 8.22*

$$ESHP_{lf} = K_Q\ Q_{lf}\ N_{R_{lf}} \qquad \textit{eq 8.23}$$

$$ESHP_{climb} = K_Q\ Q_{climb}\ N_{R_{climb}} \qquad \textit{eq 8.24}$$

$$ESHP_{descent} = K_Q\ Q_{descent}\ N_{R_{descent}} \qquad \textit{eq 8.25}$$

$$\Delta ESHP = ESHP_{climb} - ESHP_{lf} \qquad \textit{eq 8.26}$$

$$\Delta ESHP = ESHP_{descent} - ESHP_{lf} \qquad \textit{eq 8.27}$$

$$GW = ESGW - FU \qquad \textit{eq 8.28}$$

$$V' = \frac{33{,}000\ (\Delta ESHP)}{GW} \qquad \textit{eq 8.29}$$

$$V_V = \left(\frac{dh}{dt}\right) K_{H_P} \qquad \textit{eq 8.30}$$

$$v_i = 60 \sqrt{\frac{GW}{2\ \rho\ A_D}} \qquad \textit{eq 8.31}$$

$$V_c = V_o + \Delta V_{ic} + \Delta V_{pos} \qquad \textit{eq 8.32}$$

$$V_T = \frac{V_c}{\sqrt{\sigma_T}} \qquad \textit{eq 8.33}$$

$$\omega = \frac{\theta}{\Delta t} \qquad \textit{eq 8.34}$$

$$R/D = \frac{\Delta H_{P_c}}{\Delta t}$$ *eq 8.35*

$$V' = \frac{(ESHP_A - ESHP_{req})\,(33{,}000)}{W}$$ *eq 8.36*

$$V_v = \frac{\left(\frac{V_v}{V'}\right)}{V'}$$ *eq 8.37*

$$ESHP_{ref} = E_0 + E_1\,V_{T_{ref}} + E_2\,V^2_{T_{ref}} + E_3\,V^3_{T_{ref}} + E_4\,V^4_{T_{ref}}$$ *eq 8.38*

$$ESHP_{ref} = \frac{ESHP_T}{\sigma_T}\left(\frac{N_{R_S}}{N_{R_T}}\right)^3$$ *eq 8.39*

$$W_{ref} = \frac{W_T}{\sigma_T}\left(\frac{N_{R_S}}{N_{R_T}}\right)^2$$ *eq 8.40*

$$ESHP_T = K_Q\,(Q)\,(N_{R_T})$$ *eq 8.41*

$$ESHP_{ref} = A_0 + A_1\,W_{ref}^{\frac{3}{2}}$$ *eq 8.42*

$$\theta = \frac{T_a}{T_{ssl}} = \frac{OAT + 273.15}{288.15}$$ *eq 8.43*

$$\delta = \left[1 - \left(\frac{\lambda_{ssl}\,H_P}{T_{ssl}}\right)\right]^{\frac{g_{ssl}}{g_c\,\lambda_{ssl}\,R}}$$ *eq 8.44*

$$\sigma = \frac{\delta}{\theta} \qquad \textit{eq 8.45}$$

$$V' = \frac{\left(ESHP_{clim\ b} - ESHP_{lf}\right)(33{,}000)}{W_T} \qquad \textit{eq 8.46}$$

$$V' = \frac{\left(ESHP_{descent} - ESHP_{lf}\right)(33{,}000)}{W_T} \qquad \textit{eq 8.47}$$

$$\frac{T_T}{T_S} = \frac{OAT + 273.15}{T_{ssl} - \lambda_{ssl} H_p} \qquad \textit{eq 8.48}$$

$$V_V = 60\left(\frac{T_T}{T_S}\right)\left(\frac{\Delta H_p}{\Delta t}\right) \qquad \textit{eq 8.49}$$

$$V_i = 60\sqrt{\frac{W_T}{2\ \rho_{ssl}\ \sigma\ A_D}} \qquad \textit{eq 8.50}$$

$$\theta = 1 - \left(\frac{\lambda_{ssl}}{T_{ssl}}\right) H_P \qquad \textit{eq 8.51}$$

$$\theta = \frac{312.61 - \lambda_{ssl} H_p}{T_{ssl}} \qquad \textit{eq 8.52}$$

CHAPTER EIGHT

CLIMB AND DESCENT PERFORMANCE

8.1 INTRODUCTION

This section deals with aspects of determining the forward flight climb and autorotative descent performance characteristics of the helicopter. Theory relating to forward flight climbs and autorotation is an extension of the vertical climb and level flight power required theory. The method of test used at the USNTPS is the "Sawtooth Climb" method, a term which is derived from the flight profile flown during the tests. Other climb and descent data correction methods are also presented. The requirements of the specification are examined where applicable; however, the primary emphasis of this test is based on service suitability. The climb performance data are used in conjunction with level flight power required data to establish a climb schedule for a climb to service ceiling.

8.2 PURPOSE OF TEST

The purpose of this test is to examine the forward flight climb and autorotative performance characteristics to achieve the following objectives:

1. Determine the following points of interest:
    a. Airspeed for maximum rate of climb, $V_{max\ R/C}$.
    b. Airspeed for minimum rate of descent, $V_{min\ R/D}$.
    c. Airspeed for maximum autorotation glide range, $V_{max\ glide}$.
2. Define mission suitability problem areas in forward flight climbs and autorotations.
3. Evaluate the pertinent requirements of the detailed model specification.

8.3 THEORY

8.3.1 Forward Flight Climb Performance

8.3.1.1 GENERAL

The rate of climb of an aircraft climbing at a constant airspeed is directly proportional to the power available in excess of that required to maintain level flight at the same airspeed. At a given altitude and gross weight, there will be an airspeed where the maximum excess power, and therefore, the maximum rate of climb exists.

8.3.1.2 ENERGY ANALYSIS

Climb performance is affected by changes in gross weight and by variation in power available. The maximum rate of climb airspeed (from an energy analysis) will correspond to the speed for minimum power required in level flight. The relationship for rate of climb (in feet per minute) can be expressed as:

$$R/C = \frac{(ESHP_A - ESHP_{req})\,(33{,}000)}{W} \qquad \textit{eq 8.1}$$

Where:

$ESHP_A$	- Engine shaft horsepower available
$ESHP_{req}$	- Engine shaft horsepower required
W	- Weight.

Therefore, neglecting changes in induced, profile, and parasite power with rates of climb, the rate of climb can be computed easily. A level flight speed-power polar is obtained at the test gross weight and altitude and plotted against power available data as shown in Figure 8.1.

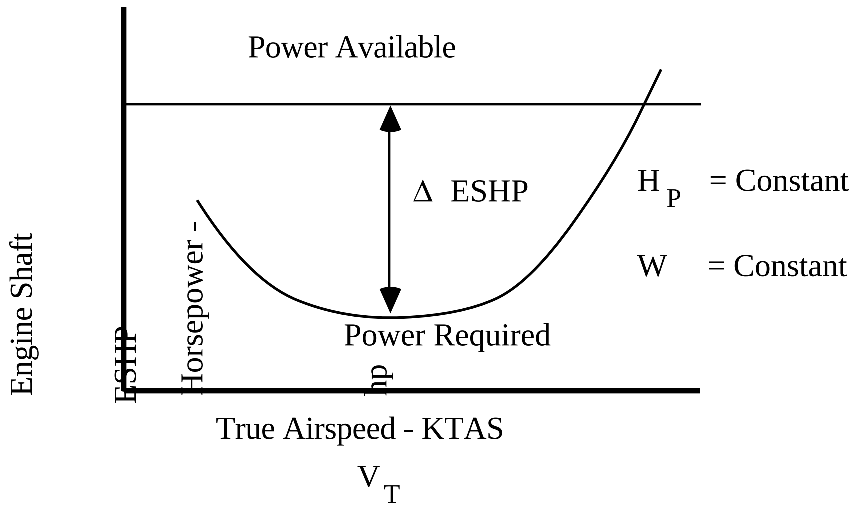

Figure 8.1
Speed-Power Polar

An energy analysis can be used to estimate climb performance for conditions other than those tested using Figure 8.2 and the following relationships:

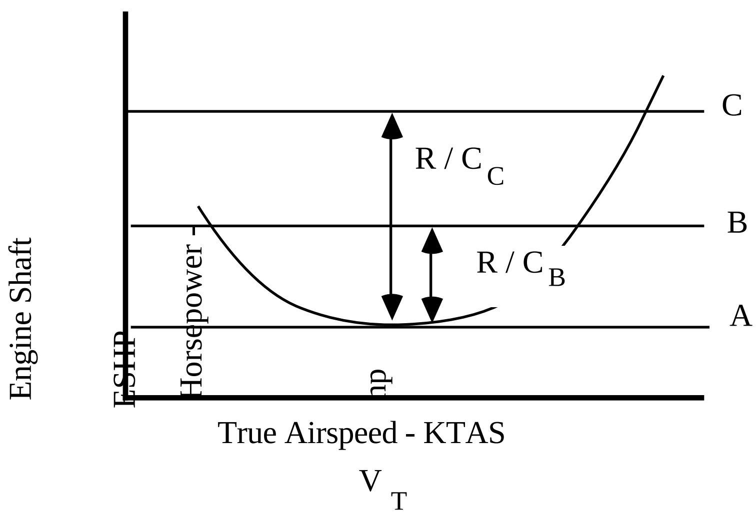

Figure 8.2
Energy Analysis of Climb Performance

Then:

$$\frac{\text{Power}_C - \text{Power}_A}{\text{Power}_B - \text{Power}_A} = \frac{(R/C)_C}{(R/C)_B} \qquad \textit{eq 8.2}$$

Where:

Power_A	- Minimum power required
Power_B	- Tested climb power
Power_C	- Power considered for estimation
$(R/C)_B$	- Tested rate of climb
$(R/C)_C$	- Rate of climb considered for estimation.

This type of analysis can be used to estimate climb performance variation with variations in altitude, gross weight, and power available. Although significant changes in rates of climb will occur with changing gross weight, altitudes, or ambient temperatures, the airspeed at which the maximum rate of climb occurs varies little. A plot of generalized climb performance is presented in Figure 8.3.

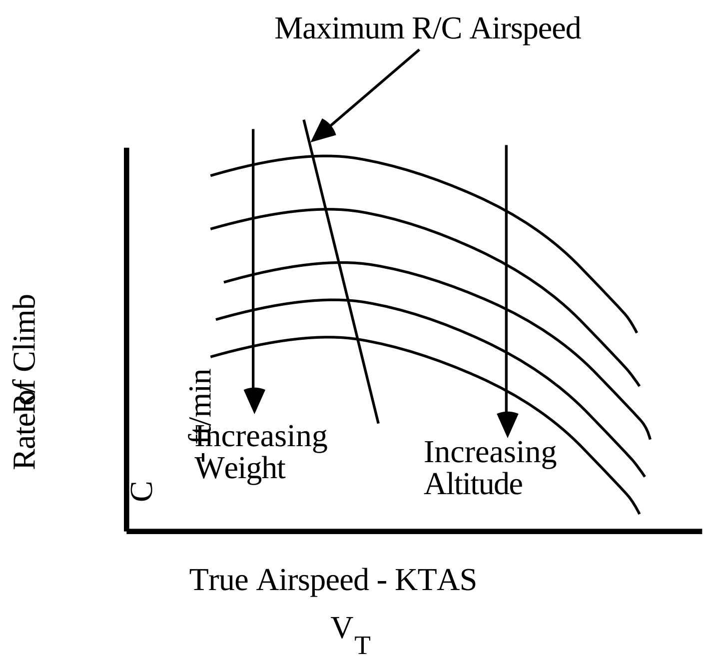

Figure 8.3
Generalized Climb Performance

8.3.1.3 MOMENTUM ANALYSIS

The effect of vertical climb and forward flight on the induced velocity, and thus on induced power, have been examined individually by simple momentum theory. By a similar development, the induced power variation can be determined approximately for an actuator disc subjected to both forward flight velocity (V_f) and vertical velocity (V_V) as shown in Figure 8.4. From simple momentum analysis, assuming the vertical force produced by the disc is equal to the aircraft weight (neglecting vertical drag):

$$W = \rho \, A_D \, V_R \, (2v_{i_v}) \qquad \textit{eq 8.3}$$

Where:

W	- Weight
ρ	- Density
A_D	- Rotor disc area
V_R	- Resultant velocity
v_{i_v}	- Induced velocity in a vertical climb.

The resultant velocity is given by the following formula and shown graphically in Figure 8.4:

$$V_R = \sqrt{V_f^2 + (V_V + v_{i_v})^2} \qquad \textit{eq 8.4}$$

Where:

V_R	- Resultant velocity
V_f	- Forward flight velocity
V_V	- Actual rate of climb
v_{i_v}	- Induced velocity in a vertical climb.

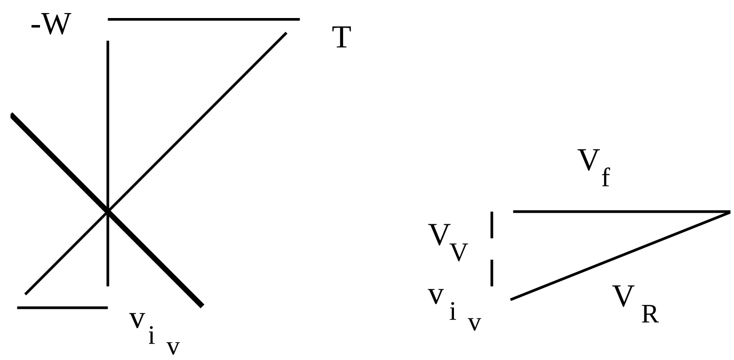

Figure 8.4
Actuator Disc in Forward Flight Climb

Using these relationships and recalling that the induced velocity in the absence of any translational velocity (hover) :

$$v_i = \sqrt{\frac{W}{2\,\rho\,A_D}} \qquad \textit{eq 8.5}$$

Where:

v_i	- Induced velocity at hover
W	- Weight
ρ	- Density
A_D	- Rotor disc area.

Producing the following:

$$v_i^2 = v_{i_v}\sqrt{V_f^2 + (V_V + v_{i_v})^2} \qquad \textit{eq 8.6}$$

Where:

v_i	- Induced velocity at hover
V_f	- Forward flight velocity
v_{i_v}	- Induced velocity in a vertical climb
V_V	- Actual rate of climb.

Rearranging into nondimensional form as:

$$\frac{v_i}{v_{i_v}} = \sqrt{\left(\frac{V_f}{v_i}\right)^2 + \left(\frac{V_V}{v_i} + \frac{v_{i_v}}{v_i}\right)^2} \qquad \textit{eq 8.7}$$

Where:

v_i - Induced velocity at hover

V_f - Forward flight velocity

v_{i_v} - Induced velocity in a vertical climb

V_V - Actual rate of climb.

The reduction of induced velocity, as a result of climbing, can be compared to the level flight results shown in Figure 8.5. Assuming the vertical force is equal to the weight, the above ratio is also that of the induced power. Considering a specified amount of maximum power available, the power available to climb will be more than the excess power in level flight (as was also observed in the vertical climb analysis), the difference being most significant at low forward speeds. This effect is shown in Figure 8.6 which considers only induced power varying with climb speed and also shows that the speed at which maximum power is available for climb changes because of this effect. If the reduction of induced power is sizable, the forward speed for maximum rate of climb will be significantly slower than that for maximum excess power in level flight.

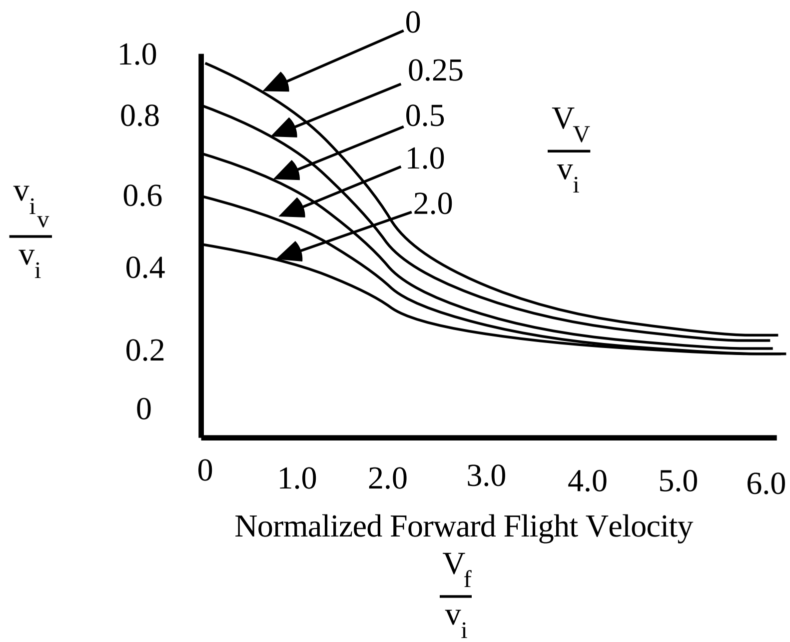

Figure 8.5
Induced Velocity in Forward Flight Climb

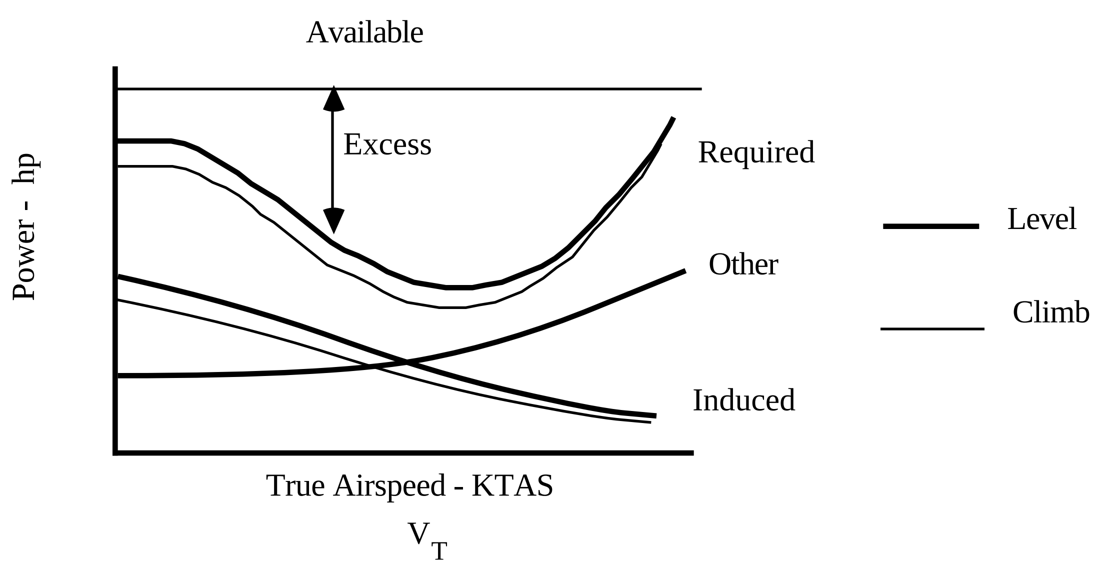

Figure 8.6
Excess Power in Forward Flight Climb

8.3.1.4 CLIMB CORRECTION FACTOR

The power required to climb is approximately the power required in level flight plus the power required to change the potential energy of the aircraft. Hence:

$$P_{climb} \approx \left(P_i + P_0 + P_p\right)_{level\ flight} + \frac{WV'}{33{,}000} \qquad \textit{eq 8.8}$$

Where:

P_{climb}	- Power required in a climb
P_i	- Induced power
P_0	- Profile power
P_p	- Parasite power
W	- Aircraft gross weight
V'	- Calculated rate of climb

However, as shown in Figure 8.5 and 8.6, the induced power in a climb will differ from that in level flight. In addition, the parasite drag in a climb will differ from that in level flight. Hence, for a constant climb power at a specific airspeed, the actual rate of climb, V_v, will differ from the calculated rat of climb, V'. The ratio

$$\frac{V_v}{V'}$$

is defined as the climb correction factor. From a flight test perspective, the purpose of performing climb tests is to determine the climb correction factor.

The power required and the power available in level flight are easily determined from flight test. The climb power can be adjusted from power for level flight to maximum power available. Hence for a given climb power, a predicted rate of climb, V', can be calculated using the expression:

$$V' = \frac{\left(P_{climb} - P_{level\ flight}\right)}{W}\left(33{,}000\right) \qquad \textit{eq 8.9}$$

Where:

V'	- Calculated rate of climb
P_{climb}	- Power required in a climb
$P_{level\ flight}$	- Power required in level flight
W	- Aircraft gross weight

During flight tests, if a climb is established at a specified airspeed and power setting, then the actual rate of climb can be measured and the climb correction factor determined.

The theory underlying forward flight climb correction factors is the same as that underlying the vertical climb correction factor, which is discussed in detail in section 6.3.4. The principal difference between the two is that with forward flight climbs, each airspeed generates a separate climb correction factor curve, whereas vertical climbs collapse to a single curve since airspeed is held at zero. Further discussion of forward flight climb correction factors can also be found in section 8.4.2.4.

8.3.2 Descent and Autorotative Performance

8.3.2.1 GENERAL

In each of the flight conditions considered previously, engine power has been required to drive the rotor against the aerodynamic forces generated in the plane of rotation by the lifting rotor. If engine power were not available to apply torque to the main rotor, rpm could not be maintained. Under certain descent conditions the lifting rotor can be driven in rotation and the rpm can be self-sustained (autorotation) by aerodynamic forces on the blades in much the same manner by which airspeed is maintained for a fixed wing aircraft in a power-off descent. Because autorotative flight provides a means of descending in the event of engine failure, the helicopter transmission is so constructed that the rotor is free wheeling and does not drive the engine. However, power must be supplied by the rotor to overcome frictional losses and to drive the tail rotor and accessories.

To simplify the analysis, the rotor is considered first in axial flight, which avoids the dissymmetry of conditions of the disc associated with translational flight. Also, it is recognized that the usual rates of descent are much smaller than the rotor blade tip speed so that, as an approximation, the local velocity on the blade is essentially that due to rotation.

8.3.2.2 BLADE ELEMENT ANALYSIS

This analysis approaches the problem by examining a discrete element, or section, of the rotor blade. Although local flow conditions, blade geometry and aerodynamic forces may well change with radial position; the general situation can be presented by analyzing an average blade element which may be thought of as representing the average conditions across the blade.

The conditions under which the rotor must operate for self-sustained rpm can be determined by observing the origin of the forces in the plane of rotation. In hover, for instance, the velocity of the air approaching the blade is in the plane of rotation and produces the resultant aerodynamic force, R, which is not normal to the velocity because of: (1) induced effects producing drag, D, and (2) viscous effects producing profile drag. The inclination of R with respect to the rotor plane of rotation produces a component of R in that plane, and in this case, a decelerating torque must be overcome by the engine to sustain rpm.

Next, consider a vertical climb. The change in direction of the velocity approaching the blade changes the direction of R in a corresponding manner such that the component of R in the plane of rotation increases, explaining the increase in power required to climb. On the other hand, if the rotor descends, R is tipped forward as the descent velocity increases, reducing the power required and ultimately, at a high enough descent rate, producing a component , F, in the plane of rotation which will actually make power available from the windmilling rotor (Figure 8.7).

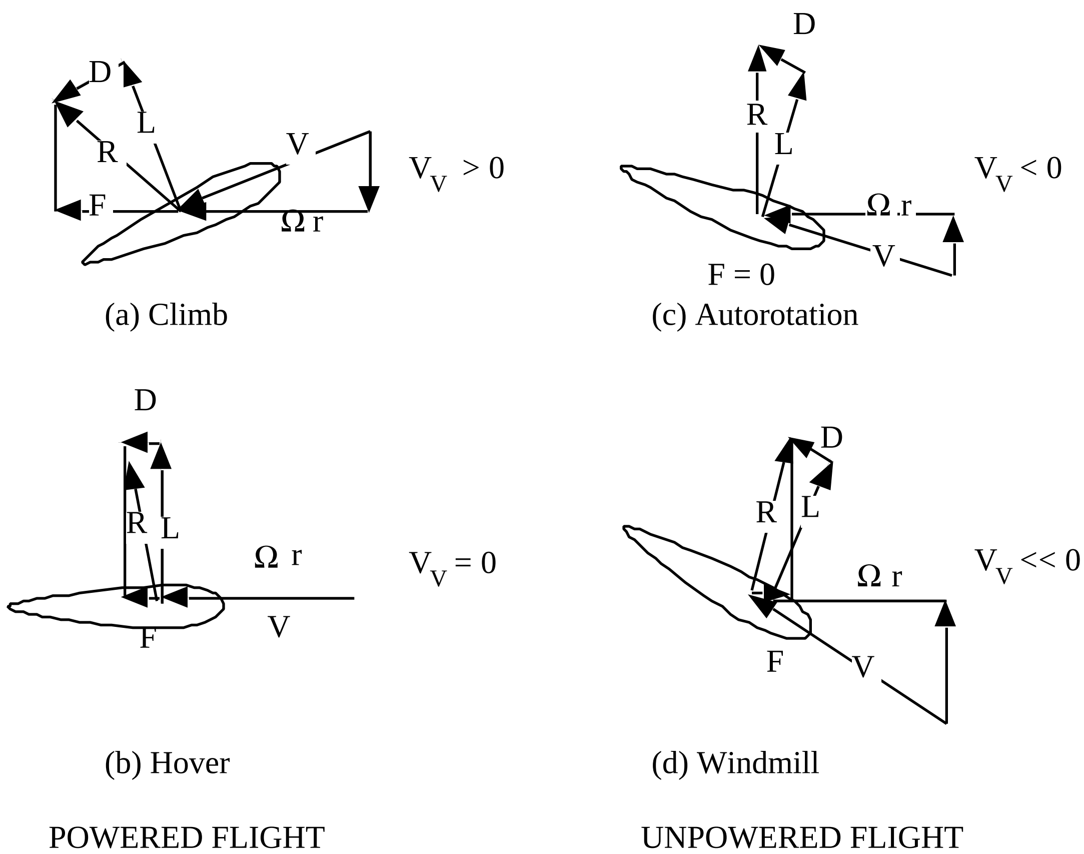

Figure 8.7
Axial Flight States of the Rotor

Autorotation is the state, at moderate rates of descent, when the resultant aerodynamic reaction, as an average across the blade, is normal to the rotor plane of rotation and the rotor is in equilibrium with zero power required. In stabilized autorotation, the resultant aerodynamic force of the average blade element is normal to the plane of rotation. Actually, the local angle of attack changes across the blade because of the variation in Ωr, resulting in the sections of the blade near the hub producing "driving" forces in the plane of rotation (windmilling) and sections near the tip have "dragging" forces in the plane of rotation (absorbing the power made available by the inboard sections). This variation is shown in Figure 8.8.

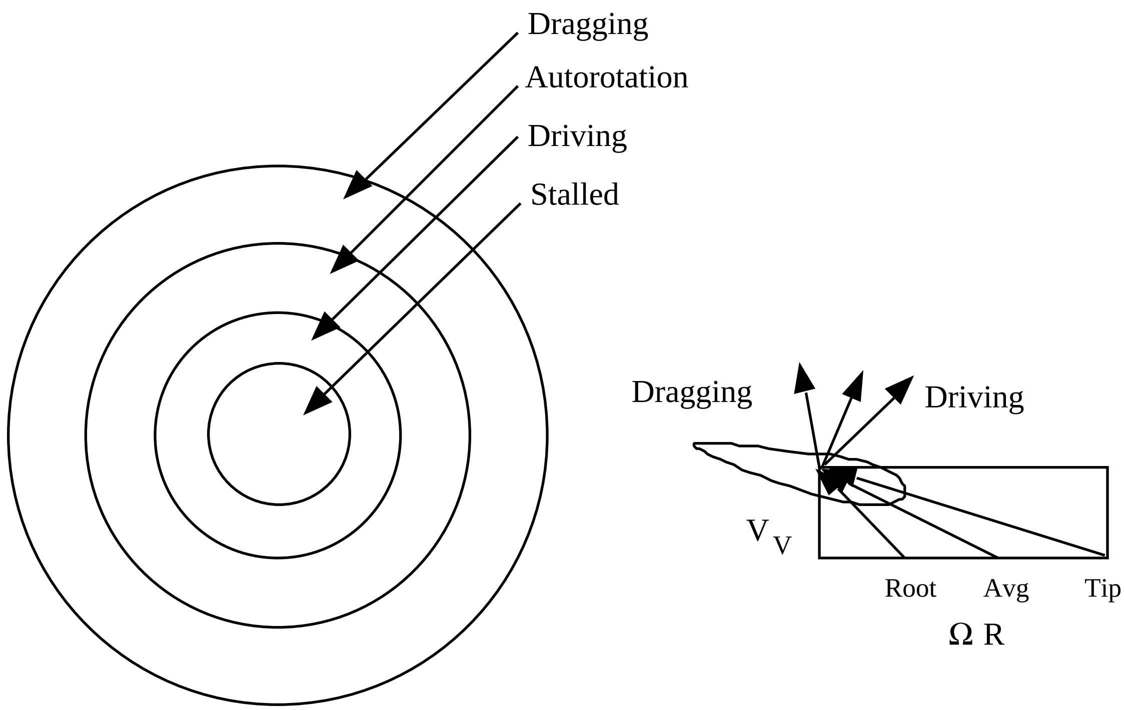

Figure 8.8
Rotor Blade Forces in Autorotation

The inclination of the resultant aerodynamic force with respect to the rotor's plane of rotation is dependent upon: (1) the angle at which the velocity is approaching the rotor disc, φ, and (2) the angle between a normal to the velocity and the resultant force, φ′. The angle φ is referred to as the inflow angle (not considering induced velocity) and can be expressed as:

$$\tan\varphi = \frac{V_V}{\Omega r}$$

eq 8.10

Or:

$$\varphi = \tan^{-1}\left(\frac{V_V}{\Omega r}\right)$$

eq 8.11

Where:

φ	- Inflow angle
V_V	- Actual rate of climb
Ω	- Rotor angular velocity
r	- Radius to blade element.

Defining lift as the component of the resultant aerodynamic force normal to the velocity and drag as the component (including induced drag) parallel to the velocity:

$$\tan\varphi' = \frac{D}{L} = \frac{C_D}{C_L} \qquad \textit{eq 8.12}$$

Or:

$$\varphi' = \tan^{-1}\left(\frac{C_D}{C_L}\right) \qquad \textit{eq 8.13}$$

Where:

φ'	- Angle between resultant and lift vectors
D	- Drag
L	- Lift
C_D	- Coefficient of drag
C_L	- Coefficient of lift.

It can be seen geometrically that the magnitude and direction of the in-plane component of R is dictated by the relative size of φ and φ' (Figure 8.9).

From the above, it can be concluded that:

1. If $\varphi' = \varphi$ the resultant force has no component in the plane of rotation and the rpm will remain constant.
2. If $\varphi' > \varphi$ the resultant force will have a component in the plane of rotation which will cause a decrease in rpm.
3. If $\varphi' < \varphi$ the component in the plane of rotation will cause an increase in rpm.

The angle φ′ is dictated by the rotor blade lift-drag ratio (L/D) which is a function of blade angle of attack (Figure 8.10).

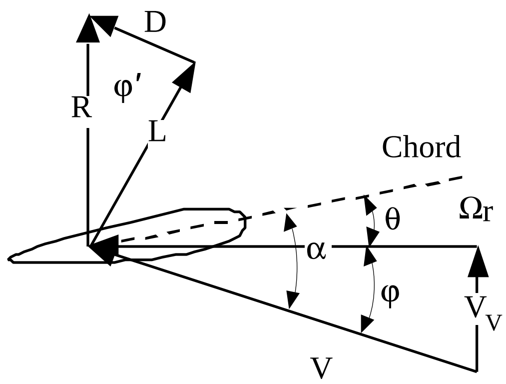

Figure 8.9
Blade Element in Equilibrium

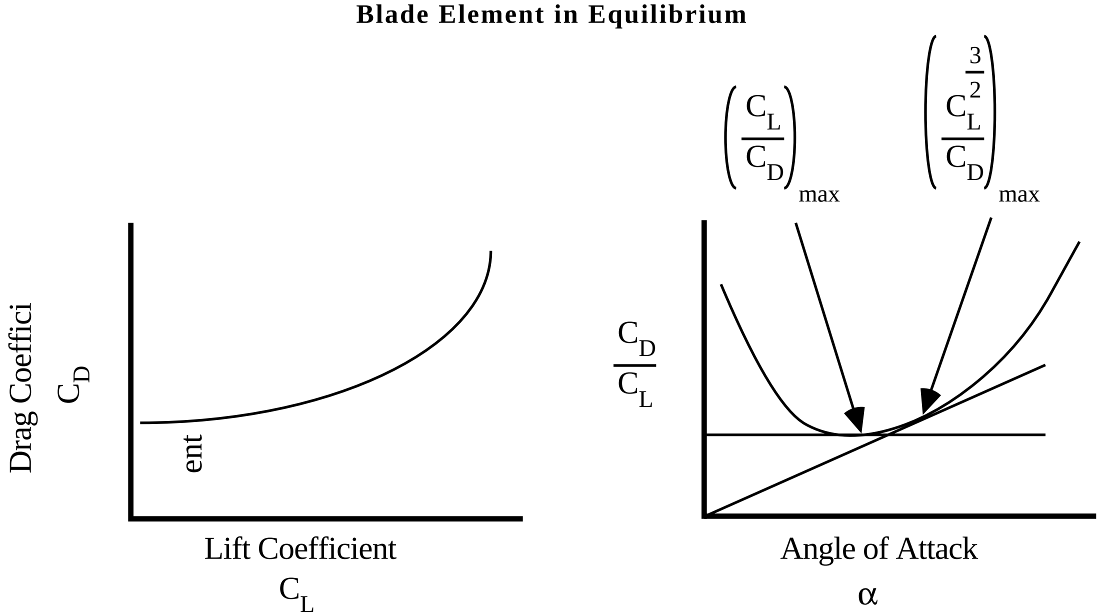

Figure 8.10
Typical Lift-Drag Ratio

The angle of attack is controlled, although somewhat indirectly, by the collective blade angle, θ, as indicated by the following relationships (Figure 8.9).

$$\alpha = \theta + \varphi \qquad \textit{eq 8.14}$$

When stabilized:

$$\varphi = \varphi'$$

Therefore:

$$\varphi' = f(\alpha) \qquad \textit{eq 8.15}$$

Where:

α	- Blade element angle of attack
φ	- Inflow angle
φ'	- Angle between the resultant and lift vectors.
θ	- Collective blade angle.

Therefore, in a stabilized autorotation, the collective pitch selected by the pilot dictates the angle of attack. Energy analysis will show that the rate of descent in vertical autorotations is inversely proportional to the ratio $C_L^{3/2}/C_D$ which indicates that if descents are made at various collective settings, the rate of descent will vary and be a minimum for a given W/α when the angle of attack is such that, as an average, the ratio of $C_L^{3/2}/C_D$ is a maximum.

Next, consider the effect a change in collective has on the rpm during autorotation of a rotor initially stabilized at an angle of attack corresponding to point “a”, Figure 8.11, for example. If conditions are stabilized, $\varphi' < \varphi$ which imposes that the pitch angle (θ) measured along the angle of attack scale and the point of operation on the L/D curve must lie on a 45 degree line, both axes being drawn to the same scale. If the collective angle is increased, the immediate consequence is a change in angle of attack and a different L/D (φ') with no changes in velocities (φ). Thus, a transient situation exists where $\varphi' \neq \varphi$, producing a force in the plane of rotation and causing the rotor rpm to change. If the step in collective causes a reduction in the lift-drag ratio ($\varphi' > \varphi$) the rpm will decrease (Figure 8.11), whereas an increase in L/D ($\varphi' < \varphi$)will increase the rpm with maximum rpm occurring at maximum L/D.

The proceeding discussion has shown how the collective selected by the pilot controls the rotor angle of attack and the lift-drag ratio. Operation near $(C_L/C_D)_{max}$ produces maximum rotor rpm and near $(C_L^{3/2}/C_D)_{max}$ produces minimum rate of descent. It should be pointed out that the variation of C_L/C_D and $C_L^{3/2}/C_D$ versus angle of attack is relatively flat near the maximum value (Figure 8.12) and, therefore, the rate of descent and rpm are not sensitive functions of collective position.

A final point of interest on the L/D curve is where the slope of the curve is unity (45 degree) because this represents the highest angle of attack for equilibrium in autorotation. It is important to note that for pitch angles below this angle of attack, the autorotation is a stable phenomenon, that is, a disturbance which increases or decreases the rotor rpm changes φ and produces in-plane forces in such a manner as to return the rpm to the initial value. Above the point where the slope is unity, a decrease in rpm causes an increase in φ' larger than that of the angle of attack, causing the rotor rpm to decrease further. Recovery from this condition would require a reduction in collective; however, this condition is usually instinctively avoided because of the low rotor speed and high rate of descent associated with operation at high angles of attack.

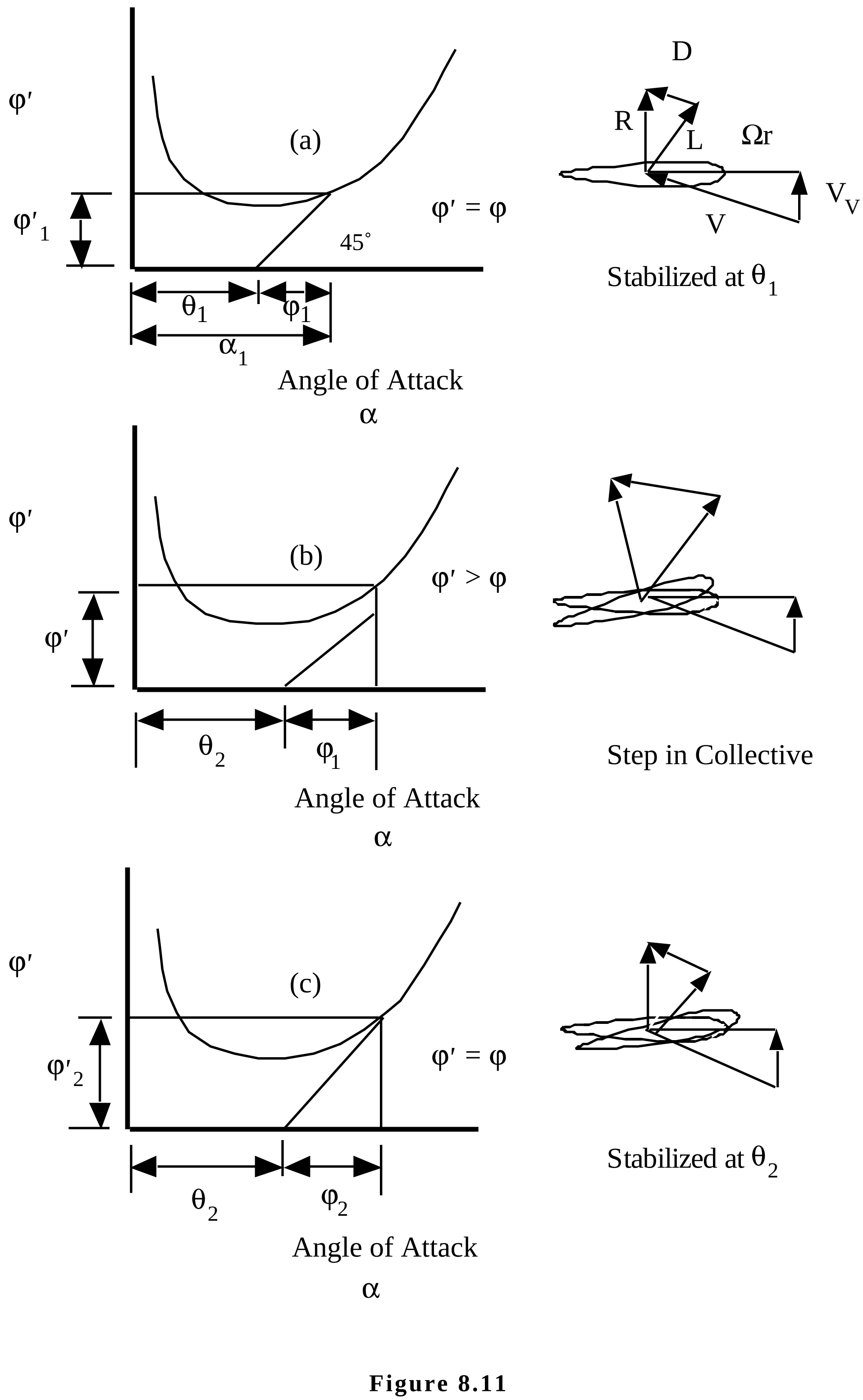

Figure 8.11
Effects of Step in Collective

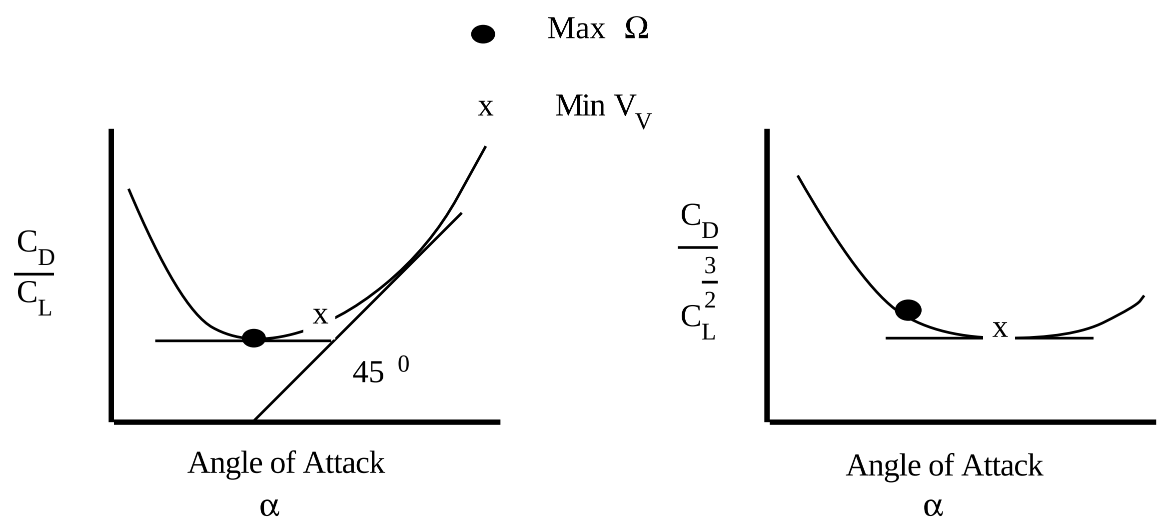

Figure 8.12
Typical Lift-Drag Variation

8.3.2.3 MINIMUM RATE OF DESCENT AIRSPEED

Since the main rotor power requirements in an autorotation are affected by essentially the same variables present in level flight, the airspeed for minimum rate of descent should coincide closely with the airspeed for minimum power required in level flight.

In autorotation, when no engine power is available, the deficit in power required must result in an equivalent reduction of either the potential energy of the helicopter (descent) or the kinetic energy of the rotor (decreasing rpm). During a constant rpm descent, the power required is equal to the time rate of change of potential energy.

$$P = \frac{d(PE)}{dt} \qquad \textit{eq 8.16}$$

Or:

$$P = W\left(\frac{dh}{dt}\right) \qquad \textit{eq 8.17}$$

Or:

$$P = W\,V_V \qquad \textit{eq 8.18}$$

Where:

P	- Power
PE	- Potential energy
t	- Time
W	- Weight
h	- Height
V_V	- Actual rate of climb or descent.

For a given gross weight and density altitude, the rate of descent in autorotation varies as the power required for level flight. Typical test results of an autorotation and level flight power required evaluation are shown in Figure 8.13.

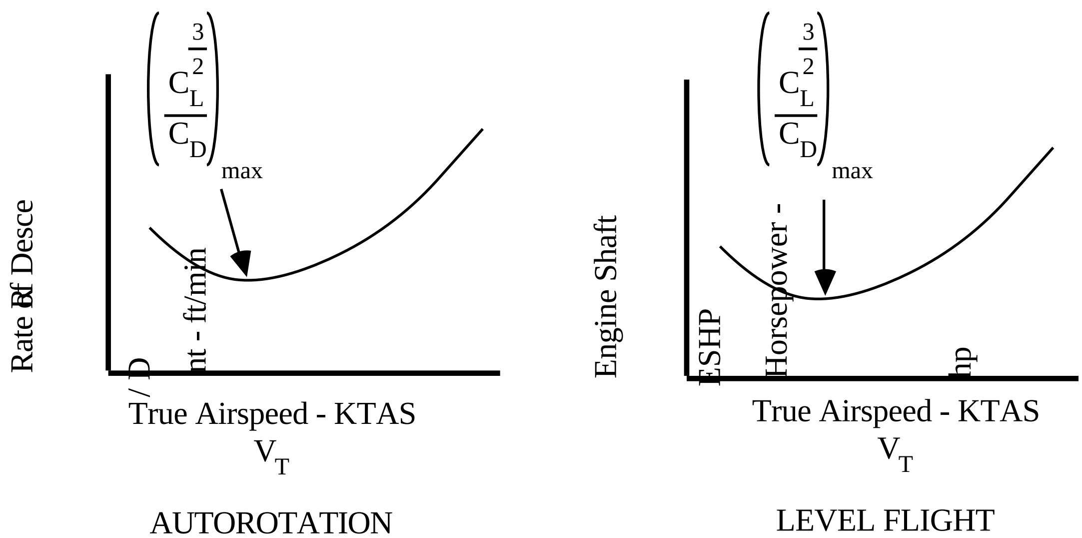

Figure 8.13
Power Requirement in Autorotation and Level Flight

8.3.2.4 MAXIMUM GLIDE RANGE AIRSPEED

The airspeed for airframe maximum range in level flight is defined as that airspeed occurring at the maximum effective lift to drag ratio. The maximum range airspeed in an autorotation will occur at the airspeed that results in the maximum lift to drag ratio and will coincide closely with that for airframe maximum range in level flight. The maximum range airspeed whether defined in level flight or an autorotation is generally higher than that for minimum rate of descent or minimum power required. Figure 8.14 illustrates the relationship between minimum rate of descent airspeed and maximum range airspeed.

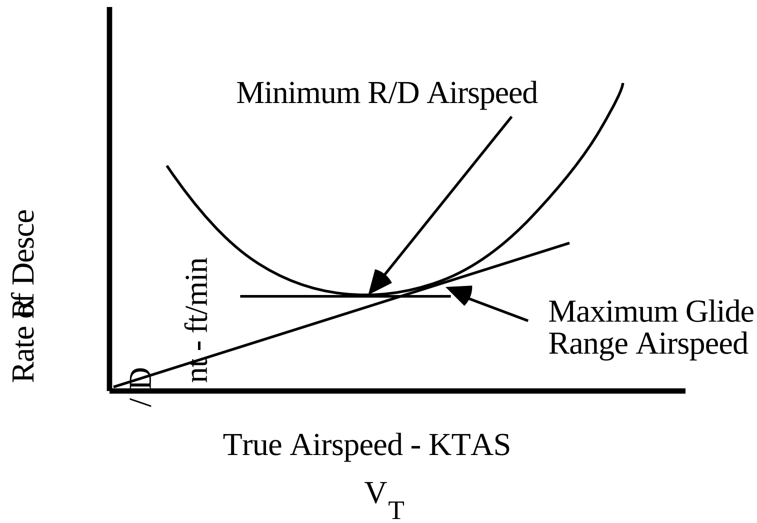

Figure 8.14
Maximum Glide Range Airspeed

The fact that the main rotor power required (consequently, the rate of descent) is a function of forward speed and rotor speed for a constant gross weight and density altitude is the basis for the autorotation flight test evaluation. Under these conditions there will be optimum combinations of forward speed and rotor speed that yield the minimum rate of descent and maximum glide range. Table 8.I presents the effects of variables on autorotative performance.

Table 8.I
Effects of Variables on Autorotative Flight

CONSTANTS	VARIABLE	RESULTS	
Airspeed Gross Weight Density Altitude	<u>Rotor Speed</u> (a) Optimum (b) High (c) Low	Rate of <u>Descent</u> Minimum High High	Power[1] <u>Required</u> Minimum High High
Rotor Speed[2] Gross Weight Density Altitude	<u>Airspeed</u> (a) Minimum R/D (b) Higher (c) Lower	Rate of <u>Descent</u> Minimum High High	Power <u>Required</u> Minimum High High
Rotor Speed[2] Airspeed Density Altitude	<u>Gross Weight</u> (a) Normal (b) Increasing (c) Decreasing	Rate of <u>Descent</u> Normal Note 3 Note 3	Power <u>Required</u> Normal High Low
Rotor Speed[2] Airspeed Gross Weight	<u>Density Altitude</u> Low High	Rate of <u>Descent</u> Lowest Highest	Power <u>Required</u> Low High

Notes: *(1) Total power required of the main rotor*
(2) Obtained by varying blade pitch angle
(3) Changes in rate of descent as gross weight is varied depends on airspeed. For low airspeeds (typically below bucket speed), decreasing gross weight results in a decreased rate of descent, and increasing gross weight results in an increased rate of descent. However, for high airspeeds (typically above bucket speed), the results are counter-intuitive, with decreasing gross weight resulting in an increased rate of descent and increasing gross weight resulting in a decreased rate of descent.

8.4 TEST METHODS AND TECHNIQUES

8.4.1 General

Forward flight climb and descent performance data gathering normally is combined to take advantage of alternating increase and decrease in altitude.
The flight profile resembles the teeth of a hand saw and thus this test is frequently called sawtooth climbs.

There are an infinite number of combinations of climb/descent rate, forward flight speed, rotor speed, gross weight, CG, configuration and altitude. Designing a test to define adequately the climb/descent performance characteristics with limited flight hours and calendar time is indeed demanding. Frequently these tests get less priority than optimum and therefore handbook data can be less than accurate. The test designer must take advantage of every opportunity to collect data to define these various flight conditions. For simplicity, climb and descent flight test techniques will be discussed separately although they are combined on the data flight.

8.4.2 Climb and Descent Performance Procedures

8.4.2.1 GENERAL

Forward flight climbs are conducted to determine several unknowns, such as the airspeed for maximum climb rate, service ceiling and correction factors to predict climb rates for conditions not tested. Regardless of the reason for the test or the purpose of the particular series of climbs, the method is essentially the same. The primary data collection centers on accurately determining the amount of power provided by the helicopter to establish the observed flight condition and recording the climb (or descent) as a function of time.

Initially climbs are conducted to establish the airspeed for the best rate of climb. In these tests increments of climb power are applied up to maximum allowable at various airspeeds to evaluate the resultant climb rate. First a trimmed, level flight data point is recorded at the test altitude. Then the aircraft is descended below the test altitude to a point such that the climb can be established and stabilized at least 400 feet below the bottom of the test altitude band. The altitude band can be any value but normally is at least ± 500 feet from the test altitude. Once the climb is established, engine power parameters must be recorded as well as time history information during the climb through the test altitude band. If an automatic data system is available the task is greatly simplified, if not, the pilot and data collector must coordinate their efforts to efficiently and accurately record the large number of data elements. A stopwatch is used to time the climb through the altitude band and at least one intermediate point should be taken. After climbing through the test altitude band a descent is made (usually combined with descent performance up to and including autorotative descent) to accomplish the next data point.

Airspeed is varied in 10 knot increments when establishing the maximum climb rate airspeed. Data should be collected in 5 knot increments near the maximum climb rate airspeed to ensure curve definition. These climb data points should be taken in as short a time span as possible to minimize gross weight change due to fuel consumption.

Rotor speed can be varied to determine climb or descent rate sensitivity, but normally the recommended standard rotor speed is sufficient.

The effect of wind gradient on test data is frequently overlooked. The base weather service can provide wind velocity and direction information as a function of altitude. Wind shear greater than a few knots in the desired test altitude band should be cause for alarm in that these effects will introduce errors which will appear as scatter in the reduced data. A possible solution to this problem is to fly the data collection runs crosswind to the predicted wind shear direction, but predictions are sometimes wrong and the best solution is to choose another altitude band.

One other variation in test method is worthy of discussion. Under optimum circumstances the initial level flight trim data point can be eliminated by using the previously documented level flight performance data to obtain the reference power. This method must be used with caution since it is a potential source of errors. The level flight performance characteristics of the specific aircraft must be known. Two aircraft of the same type can have distinguishable differences in level flight power required. Data recording system calibrations can drift or even change from flight to flight.

The above procedures are used regardless of the method of data reduction. Three commonly used methods of dealing with these data sets are detailed below, beginning with the least complicated and progressing to the more complex.

8.4.2.2 TEST DAY DATA CLIMB AND DESCENTS

The most simple test data procedure only involves correcting the recorded altitude information for non-standard temperature. The recorded rate of climb or descent is simply multiplied by a correction factor given by Equation 8.19. This method is widely used when minimal flight time and resources are available. The data obtained will be applicable only to the test conditions flown. Extrapolation to other gross weights, centers of gravity, configurations, or even altitude bands could yield large errors and would not be considered good flight test methodology.

$$K_{H_P} = \frac{T_a + 273.15}{T_S + 273.15} \qquad \textit{eq 8.19}$$

Where:

K_{Hp} - Temperature correction factor

T_a - Ambient temperature

T_S - Standard temperature (at that altitude).

8.4.2.3 POWER AND WEIGHT CORRECTIONS

Two correction factors are particularly important because they are commonly used to correct test day data and provide the basis for estimates of climb and descent performance. These are K_P (power correction factor) and K_W (weight correction factor). The determination of these factors requires considerable planning and dedicated flight time.

Power correction factor flights require the aircraft be flown through altitude bands (as in the paragraphs above) with incrementally higher and lower power settings above and below that required for level flight. Gross weight should be held nearly constant and therefore frequent ballasting is required. A range of altitudes are flown and the recommended climb airspeed for each altitude is maintained. After this array of data are plotted, normally a family of curves (or if you are lucky, straight lines) will result, thus providing the relationship between power increments and change in rate of climb. A typical power correction factor relationship is shown in Figure 8.15

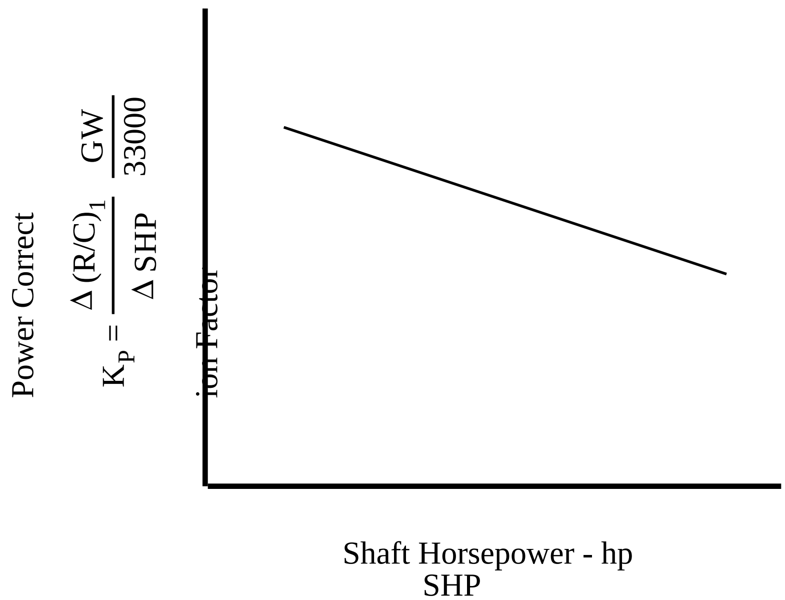

Figure 8.15
Power Correction Factor

Weight correction factor flights are flown in much the same manner except power is fixed, usually at the military rated power (MRP) level for each climb, and gross weight varied to the extremes possible for the specific aircraft. Again, the airspeed is held constant at the airspeed for maximum rate of climb for each of the altitudes investigated. This array of data will normally result in another family of curves depicting the influence of gross weight changes on rate of climb for a specific altitude. In these tests fuel burn works to the advantage of the test team in reducing the ballasting requirements. A typical weight correction relationship is shown in Figure 8.16

Once these relationships are established, a much smaller number of test day data flights can be flown and accurate estimates of climb and descent capabilities can be predicted for test conditions not actually flown. The estimated climb or descent rate for a condition of interest is simply the product of a known climb or descent rate (preferably at a flight condition not greatly different from the condition of interest) and the two correction factors. The primary disadvantage of this method is the requirement to have large amounts of data at widely different gross weights, power increments, and over a wide range of altitudes to define adequately the K_P and K_W relationships. This method does lend itself to computerized iterative predictions of climb and descent performance and is used frequently to develop the data provided in aircraft handbooks.

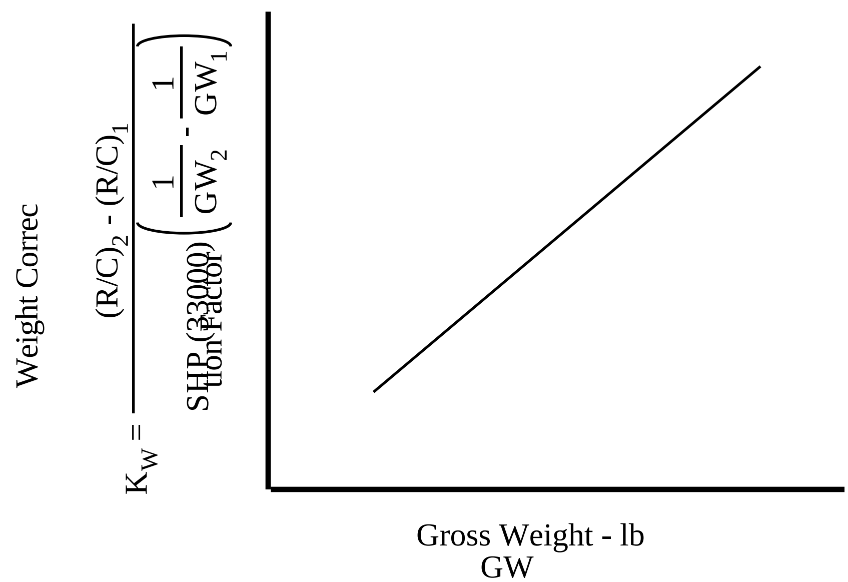

Figure 8.16
Weight Correction Factor

8.4.2.4 CLIMB AND DESCENT CORRECTION FACTOR

The most versatile method of estimating climb and descent performance data is by developing forward flight climb and descent correction factor relationships. This requires a very large amount of data collected over extreme ranges of gross weights, power increments, and altitudes. Ideally, the relationship between a simple prediction (Equation 8.1) and the actual climb performance must be established over the entire range of aircraft capabilities. The data is normalized by dividing by the induced velocity for hover. Predicted versus actual normalized rates of climb should resemble that shown in Figure 8.17. Note that multiple normalized forward flight velocity curves must all be defined.

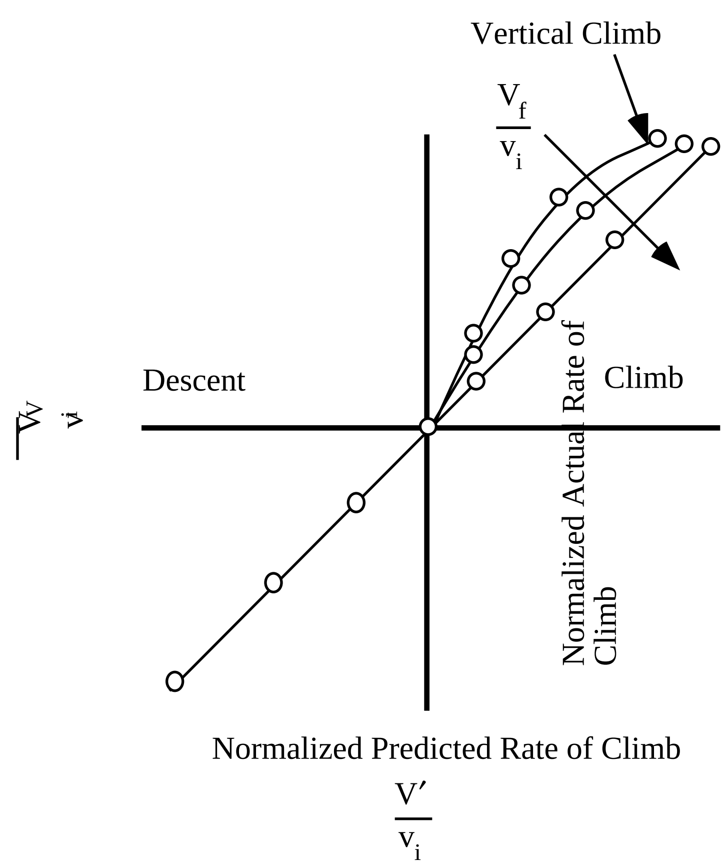

Figure 8.17
Generalized Climb and Descent Performance

This same data in a slightly different form (Figure 8.18) yields a correction factor relationship that although extremely difficult to completely define, is capable of correcting a wide variety of predicted climb performance capabilities to actual climb capabilities. If reasonable care is taken in developing the forward flight climb correction factor relationships, virtually any climb performance values can be obtained with a high degree of accuracy and confidence.

The most important thing to remember about this method is that large amounts of test day data are required to fully determine the families of curves which depict the helicopter's overall climb and descent performance capabilities. Only the regions of these figures that correspond to logical combinations of forward flight speed, gross weight and available climb/descent rates are required. Even with this reduction in data scope, few programs dedicate sufficient time for climb performance evaluations to define fully these figures.

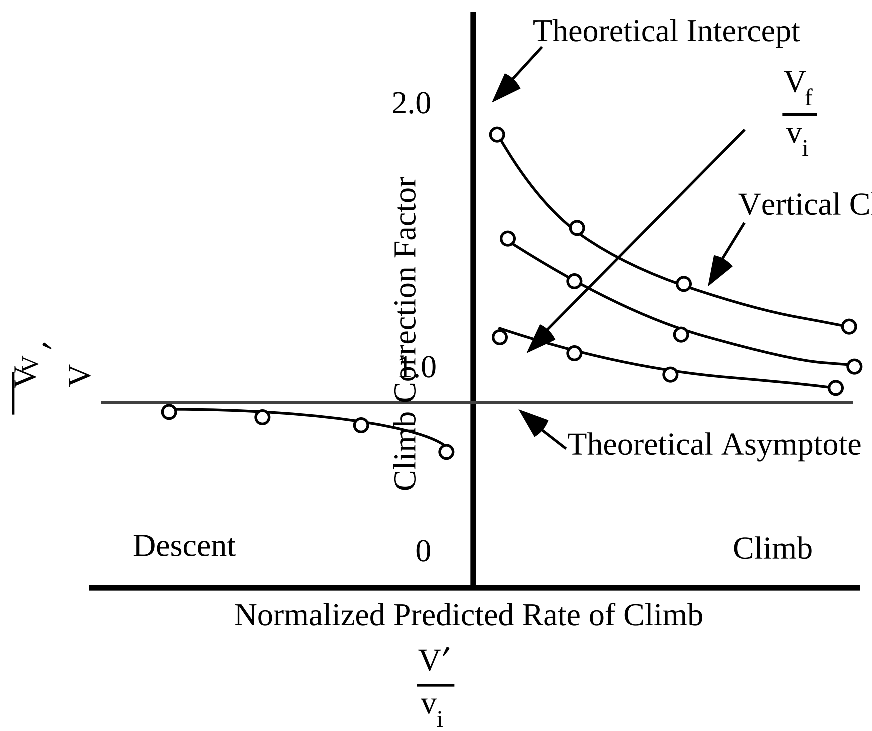

Figure 8.18
Generalized Climb and Descent Correction Factor

8.4.2.5 DATA REQUIRED

Level Flight:

T_o, H_{P_o}, N_R, Q, V_o, fuel, β_o, configuration.

Climb:

T_o, N_R, Q, V_o, fuel, β_o, configuration, time history of climb.

Where:

T_o - Observed temperature

H_{P_o} - Observed pressure altitude

N_R - Main rotor speed

Q - Engine torque

V_o - Observed airspeed

β_o - Observed sideslip angle.

8.4.2.6 TEST CRITERIA

1. Constant airspeed.
2. Constant power setting.
3. Constant rotor speed.
4. Stabilized climb rate.
5. Calm air.
6. Ball centered, wings level, unaccelerated flight.
7. Steady heading.

8.4.2.7 DATA REQUIREMENTS

Level Flight:

1. Airspeed ± 1 knot from schedule.
2 N_R ± 0.5 rpm.
3. Altitude ± 100 feet from test altitude.
4. Heading tolerance ± 2 degrees.

Climb:

1. Airspeed ± 2 knots from schedule.
2. N_R ± 0.5 rpm.
3. Altitude within the test band.
4. Stabilized 200 feet before entering altitude band.
5. Heading tolerance ± 2 degrees.

8.4.2.8 SAFETY CONSIDERATIONS/RISK MANAGEMENT

These tests are not particularly hazardous but good planning can work to the advantage of the flight crew should an emergency arise. Conduct climbs over areas which have adequate forced landing sites. Make power changes smoothly to reduce the chances of compressor stall or surge. Constantly be alert for engine malfunctions since rotor speed decay will be rapid if failure occurs at high power settings. Transitioning from a descent to a high power climb runs the risk of entering vortex ring state, so review recovery procedures. Tail rotor malfunctions at high power settings can rapidly lead to loss of aircraft control, so these procedures for the specific aircraft must be studied. Conducting high power climbs at high altitude increases the risk of encountering compressor stall, surge or even flameout. Study emergency procedures for these occurrences to include engine restart procedures.

8.4.3 Continuous Climb Procedure (Service Ceiling Climb)

Another method of obtaining climb performance data is the continuous climb method. For this procedure, each climb begins immediately after takeoff and continues until certain criteria are met. These criteria may be specific cutoff altitudes; i.e. restricted by operating limits of some system such as hydraulics, or until a predetermined climb rate is reached such as less than 100 fpm. This method requires a climb airspeed schedule be established prior to flight. This schedule is obtained by evaluating the minimum power required airspeeds determined in level flight performance tests and comparing these with the best climb rate airspeeds obtained using the methods discussed in paragraph 8.4.2.2. The schedule can be complex or simple, depending on the purpose of the data collection. If the optimum aircraft climb capability is to be evaluated, the climb schedule may be different for each 1000 foot altitude variation all the way up to the service ceiling. Most of the time the climbs are conducted to evaluate the data to be included in the operator's manual or to generate data for that purpose. In this case the climb schedule which is recommended in the handbook should be used. Typical schedules may indicate a constant climb airspeed below 10,000 foot pressure altitude and a one knot per thousand decrease above that altitude. Regardless of the schedule, strict adherence to these values must be maintained to produce accurate results.

The climb is flown until the stop climb criteria is met. The aircraft is descended for another climb or configuration change for the next data collection requirement. These climbs can be flown in support of climb correction factor data requirements or the parameter correction factor requirements (K_P and K_W). During K_W collection climbs, the power is maintained at MRP throughout the climb. During K_P climbs, the power is maintained at the torque setting specified for the climb. Generally, at least four gross weights are used to define K_W and four power levels between power for level flight and MRP are used to define K_P. During the data reduction process, climb rates are evaluated at selected altitudes to provide specific data points for curve definition.

8.4.3.1 DATA REQUIRED

T_o, N_R, Q, V_o, fuel, β_o, configuration, climb rate.

Note: These data elements are generally continuously recorded on a data system and specific data elements reduced at altitude increments.

8.4.3.2 TEST CRITERIA

1. Adherence to airspeed schedule.
2. Maintain power level in climb.
3. Constant rotor speed.
4. Stable climb rate.
5. Calm air.
6. Ball centered, wings level, unaccelerated flight.
7. Steady heading.

8.4.3.3 DATA REQUIREMENTS

1. Airspeed ± 2 knots from schedule.
2. N_R ± 0.5 rpm.
3. Heading tolerance ± 2 degrees.

8.4.3.4 SAFETY CONSIDERATIONS/RISK MANAGEMENT

The considerations discussed in paragraph 8.4.2 are applicable here also. Particular care must be taken at high altitude to recognize engine problems such as compressor surge, stall or flameout. The flight crew must also use oxygen equipment when conducting these flights above 10,000 ft mean sea level or the particular regulatory requirement for the controlling service. Portable oxygen equipment is available for these flights but the use and operation must be planned and practiced. Be particularly mindful of your own hypoxia symptoms and observe your fellow crew members also. Helicopter pilots flying at high altitude must be alert for high speed traffic because jet traffic are not accustomed to watching out for slow speed traffic at these extreme altitudes. For this reason, it is a good idea to flight follow with the local radar facilities and operate in restricted airspace.

8.4.4 Autorotative Descent Performance

These tests are normally conducted with the climb performance tests. Autorotative descent tests are conducted to determine the airspeed for minimum descent rate, recommended autorotative airspeed, best glide range airspeed, and rotor speed effects on descent rate. Sufficient data must be taken to define fully the allowable aircraft gross weight, rotor speed, center-of-gravity and external stores configurations.

The first test conducted should be the effect of airspeed on descent rate. A good start point is the level flight minimum power required airspeed. Airspeed should be increased and decreased in 10 knot increments to develop fully the curve and 5 knot increments should be flown in the vicinity of the minimum descent rate airspeed. For this series of descents, hold the rotor speed constant to observe airspeed effects. The aircraft descent parameters must be stabilized prior to reaching the test altitude band. A good rule of thumb is the aircraft must be stable at least 400 feet prior to the altitude band to ensure a stable descent rate. Automatic data collection systems record time histories of the descents or a data collector must time the descent through the band. At least one intermediate altitude and time combination should be obtained to verify that the descent was constant throughout the band.

Using only small control inputs smoothly applied to the controls will greatly increase the quality of the data. Abrupt control manipulations cause rotor speed fluctuations that will be unacceptable for data collection purposes. Pilots should use outside references to help provide attitude change cues.

Many aircraft require that the engine speed must be reduced to prevent the idling engine from powering the main rotor. This is done by reducing engine rpm with the throttles or beep controls to obtain a definite rotor/engine disengagement (split needles). The engine must be restored to the powered flight condition prior to recovery from the autorotative descent.

After the minimum descent airspeed has been established, pilot qualitative comments are considered in the establishment of the recommended autorotative descent airspeed. This speed is usually higher than the speed for minimum descent to take advantage of increased stability on the front side of the power required curve and better field of view in the descent.

The best glide airspeed is also determined from these tests. This airspeed provides the most distance traveled horizontally for a given height.

At each of these three speeds (minimum descent rate, pilot recommended and best glide) the effects of rotor speed variation should be determined. This test involves conducting descents at these airspeeds at various rotor speeds to determine the optimum rotor speed and/or the effects on rate of descent of poorly maintained rotor speed. Tests are sometimes conducted to determine the effects of out of trim descents.

Wind shear is a concern when conducting these tests. Changes in wind speed and direction can effect the descent rate data. The test manager should check with the local weather service to obtain wind speed and direction information as a function of altitude. If the tests must be conducted within an area of wind shear, then the flight heading should be oriented crosswind to minimize these effects.

8.4.4.1 DATA REQUIRED

T_o, N_R, Q, V_o, fuel, configuration, time history of descent.

8.4.4.2 TEST CRITERIA

1. Constant airspeed.
2. Constant rotor speed.
3. Stabilized rate of descent.

4. Calm air.
5. Ball centered, wings level, unaccelerated flight.
6. Constant heading.

8.4.4.3 DATA REQUIREMENTS

1. Stabilized 400 feet prior to altitude band.
2. Airspeed ± 2 knots from target.
3. N_R ± 1.0 rpm target.
4. Heading tolerance ± 2 degrees.

8.4.4.4 SAFETY CONSIDERATIONS/RISK MANAGEMENT

The risk of reducing power to idle while in flight is obvious. Increase engine speed prior to recovery or the rotor speed will droop excessively. There is always the chance that the engine will not respond to increased power commands. For these reasons good forced landing areas must be available. Continuously monitor engine instruments to ensure that the engine continues to run. A full recovery must be made prior to 1000 ft above ground level so that if a recovery is not effected the pilot will have some altitude to maneuver for a landing. Rotor speed and engine parameters must be monitored during recovery to prevent overtorques, overspeeds or excessively low rotor speeds. The area below the test aircraft should be cleared prior to entering the descent maneuver. Flights at high altitude require supplemental oxygen. Thoroughly review the autorotative landing procedures for your aircraft. Know and observe aircraft and engine limitations.

8.4.5 Autorotative Turning Performance

Autorotative turning performance is most closely associated with mission suitability comments made by the pilot when conducting autorotations with a turn. The data provided from these tests substantiates pilot comments about rate of descent changes with bank angle, and radius of turn. These tests are conducted at altitude for safety purposes. An autorotative descent is initiated at the recommended autorotative descent airspeed. Turns are conducted both left and right at various bank angles and data are collected much as with straight ahead autorotations except turn information must now be recorded. This is commonly done by reference to the aircraft heading indicator. The final result should be concurrent rate of descent and turn rate information for the aircraft bank angle. The objective of the test is to determine the optimum bank angle for completing a specified turn (usually 180 degrees) with a minimum altitude loss.

There are many piloting technique difficulties involved in this test. Maintaining a constant rotor speed requires constant pilot attention. Smooth collective changes and coordination between cyclic control application and collective are required. For example, aft cyclic control will increase load factor in the turn and generally increase rotor speed. Coordination of these two controls can minimize further adjustments. A rapid cross check is required to maintain aircraft attitude, airspeed and rotor speed.

8.4.5.1 DATA REQUIRED

T_o, N_R, Q, V_o, fuel, configuration, ϕ, time history of turn rate and descent.

8.4.5.2 TEST CRITERIA

1. Constant airspeed.
2. Constant rotor speed.
3. Stabilized rate of descent.
4. Calm air.
5. Ball centered directional trim.
6. Constant bank angle.

8.4.5.3 DATA REQUIREMENTS

1. Airspeed ± 3 knots.
2. N_R ± 3 rpm.
3. ϕ ± 2 degrees.

8.4.5.4 SAFETY CONSIDERATIONS/RISK MANAGEMENT

The safety considerations/risk management for autorotative turning performance are similar to the zero bank angle tests. As the bank angle is increased the pilot will notice an increase in rotor speed. Another common problem with these tests is maintaining proper division of attention. The flight crew can become too involved in monitoring cockpit instruments and become unaware of dangers outside the aircraft. The pilot may become so involved in flying the data point that the aircraft is allowed to descend below safe operating altitudes. These test procedures require practice to include crew rehearsal in the test aircraft.

8.5 DATA REDUCTION

8.5.1 General

The objectives of forward flight climb, descent and autorotative descent performance are to determine test day performance characteristics and compute generalized correction factors to allow prediction of other than test condition performance. The method of test used at USNTPS to obtain forward flight climb and autorotation data is the sawtooth climb method. The name is derived from the flight profile flown during the tests. Correction factor tests are performed to document airspeed, rotor speed, gross weight and power variations. The determination of correction factors is very time consuming and is sometimes omitted from the test program in favor of more critical flight test evaluations.

Each of these tests requires the recording of climb/descent time histories. Test day data are compared to mathematically predicted climb/descent performance and correction factors are determined. These correction factors allow the prediction of climb/descent performance at any referred gross weight, rotor speed or atmospheric condition.

USNTPS uses a computer data reduction routine for helicopter forward flight climb and descent performance data reduction. This section presents a manual method of data reduction as well as an introduction to the computer data reduction routine.

Whether the test is for forward flight climbs, partial power descents, or autorotative descents, the data reduction schemes are similar. Gross weight and corrected engine power must be calculated for each data point including the level flight condition.

All observed data must be corrected for instrument and position errors as appropriate.

8.5.2 Manual Data Reduction

The following equations are used in the manual data reduction calculations:

$$H_{P_c} = H_{P_o} + \Delta H_{P_{ic}} + \Delta H_{pos} \qquad \textit{eq 8.20}$$

Where:

H_{P_c}	- Calibrated pressure altitude
H_{P_o}	- Observed pressure altitude
$\Delta H_{P_{ic}}$	- Altimeter instrument correction
ΔH_{pos}	- Altimeter position error correction.

$$T_a = T_o + \Delta T_{ic} \qquad \text{eq 8.21}$$

Where:

T_a	- Ambient temperature
T_o	- Observed temperature
ΔT_{ic}	- Temperature instrument correction.

$$\sigma_T = \frac{\delta}{\theta} = \frac{\delta}{\left(\frac{T_a}{T_{ssl}}\right)} \qquad \text{eq 8.22}$$

Where:

σ_T	- Test density ratio
δ	- Pressure ratio
θ	- Temperature ratio
T_a	- Ambient temperature
T_{ssl}	- Standard sea level temperature.

$$ESHP_{lf} = K_Q\, Q_{lf}\, N_{R_{lf}} \qquad \text{eq 8.23}$$

$$ESHP_{climb} = K_Q\, Q_{climb}\, N_{R_{climb}} \qquad \text{eq 8.24}$$

$$ESHP_{descent} = K_Q\, Q_{descent}\, N_{R_{descent}} \qquad \text{eq 8.25}$$

Where:

$ESHP_{lf}$	- Engine shaft horsepower in level flight
$ESHP_{climb}$	- Engine shaft horsepower in climb
$ESHP_{descent}$	- Engine shaft horsepower in descent
K_Q	- Engine torque constant unique for each aircraft
Q_{lf}	- Engine torque in level flight
Q_{climb}	- Engine torque in a climb
$Q_{descent}$	- Engine torque in a descent
$N_{R\ lf}$	- Main rotor speed in level flight
$N_{R\ climb}$	- Main rotor speed in a climb
$N_{R\ descent}$	- Main rotor speed in a descent.

$$\Delta ESHP = ESHP_{climb} - ESHP_{lf} \qquad \textit{eq 8.26}$$

$$\Delta ESHP = ESHP_{descent} - ESHP_{lf} \qquad \textit{eq 8.27}$$

Where:

$\Delta ESHP$	- Excess engine shaft horsepower
$ESHP_{climb}$	- Engine shaft horsepower in climb
$ESHP_{descent}$	- Engine shaft horsepower in a descent
$ESHP_{lf}$	- Engine shaft horsepower in level flight.

$$GW = ESGW - FU \qquad \textit{eq 8.28}$$

Where:

GW	- Gross weight
ESGW	- Engine start gross weight
FU	- Fuel used.

$$V' = \frac{33{,}000\,(\Delta ESHP)}{GW} \qquad \textit{eq 8.29}$$

Where:

V'	- Predicted rate of climb or descent
$\Delta ESHP$	- Excess engine shaft horsepower
GW	- Gross weight.

$$K_{H_P} = \frac{T_a + 273.15}{T_S + 273.15} \qquad \textit{eq 8.19}$$

$$V_v = \left(\frac{dh}{dt}\right) K_{H_P} \qquad \textit{eq 8.30}$$

Where:

V_V	- Actual rate of climb or descent
dh/dt	- Slope of test data time history
K_{Hp}	- Temperature correction factor.

$$v_i = 60 \sqrt{\frac{GW}{2\,\rho\,A_D}} \qquad \textit{eq 8.31}$$

Where:

v_i	- Induced velocity at hover
GW	- Gross weight
ρ	- Density
A_D	- Rotor disc area.

$$V_c = V_o + \Delta V_{ic} + \Delta V_{pos} \qquad \textit{eq 8.32}$$

Where:

V_c - Calibrated airspeed

V_o - Observed airspeed

ΔV_{ic} - Airspeed instrument correction

ΔV_{pos} - Airspeed position error.

$$V_T = \frac{V_c}{\sqrt{\sigma_T}} \qquad \textit{eq 8.33}$$

Where:

V_T - True airspeed

V_c - Calibrated airspeed

σ_T - Test density ratio.

$$\omega = \frac{\theta}{\Delta t} \qquad \textit{eq 8.34}$$

Where:

ω - Turn rate

θ - Heading change

Δt - Change in time.

$$R/D = \frac{\Delta H_{P_c}}{\Delta t} \qquad \textit{eq 8.35}$$

Where:

R/D - Rate of descent

ΔH_{P_c} - Change in calibrated pressure altitude

Δt - Change in time.

From observed level flight and forward flight climb or descent data, compute V_V/v_i, V'/v_i, V_V/V', and V_f/v_i as follows:

Parameter	Notation	Formula	Units	Remarks
(1) Observed pressure altitude	H_{P_o}		ft	
(2) Calibrated pressure altitude	H_{P_c}	eq 8.20	ft	Instrument & position corrections
(3) Observed temperature	T_o		°C	
(4) Ambient temperature	T_a	eq 8.21	°C	Instrument correction (Ignore recovery and Mach correction)
(5) Pressure ratio	δ			From tables at step (2)
(6) Test density ratio	σ_T	eq 8.22		
(7) Level flight main rotor speed	$N_{R_{lf}}$		%	(or appropriate units)
(8) Standard main rotor speed	N_{R_S}		%	(or appropriate units)
(9) Climb or descent main rotor speed	$N_{R_{climb}}$ $N_{R_{descent}}$		%	(or appropriate units)
(10) Engine torque in level flight	Q_{lf}		%, psi	(or appropriate units)
(11) Level Flight ESHP	$ESHP_{lf}$	eq 8.23	hp	
(12) Engine torque for climb or descent	Q_{climb} $Q_{descent}$		%, psi	(or appropriate units)

(13) Climb or descent ESHP	$ESHP_{climb}$	eq 8.24	hp	
	$ESHP_{descent}$	eq 8.25		
(14) Engine shaft horsepower for climb or descent	$\Delta ESHP$	eq 8.26	hp	
		eq 8.27		
(15) Fuel counts	FC		cts	
(16) Fuel used	FU	FC (gal/ct) (lb/gal)	lb	ct/gal - lab calibration lb/gal - fuel density
(17) Gross weight	GW	eq 8.28	lb	
(18) Predicted rate of climb or descent	V'	eq 8.29	ft/min	
(19) Slope of test data time history	dh/dt		ft/min	
(20) Standard	T_S		°C	From tables at step (2)
(21) Temperature correction factor	K_{Hp}	eq 8.19		
(22) Actual rate of climb or descent	V_V	eq 8.30	ft/min	
(23) Induced velocity in hover	v_i	eq 8.31	ft/min	
(24) Normalized actual rate of climb or descent	V_V/v_i	(22)/(23)		
(25) Normalized predicted rate of climb or descent	V'/v_i	(18)/(23)		
(26) Climb correction factor	V_V/V'	(22)/(18)		
(27) Observed airspeed	V_o		KOAS	
(28) Calibrated airspeed	V_c	eq 8.32	KCAS	Instrument and position correction

(29) True airspeed	V_T	eq 8.33	KTAS
(30) Normalized forward flight velocity	V_f / v_i	(V_T) (101.34) / v_i	

Plot the generalized climb and descent performance as shown in Figure 8.17.

Plot climb and descent correction factor as shown in Figure 8.18.

From the observed autorotative turning performance data, at each bank angle compute the rate of descent, turn rate and ratio of altitude loss to heading change using the following equations and procedure:

$$H_{P_c} = H_{P_o} + \Delta H_{P_{ic}} + \Delta H_{pos} \qquad \textit{eq 8.20}$$

$$\omega = \frac{\theta}{\Delta t} \qquad \textit{eq 8.34}$$

$$R/D = \frac{\Delta H_{P_c}}{\Delta t} \qquad \textit{eq 8.35}$$

Parameter	Notation	Formula	Units	Remarks
(1) Bank angle	ϕ		deg	Note direction
(2) Observed pressure altitude	H_{P_o}		ft	
(3) Calibrated pressure altitude	H_{P_c}	eq 8.20	ft	Instrument and position corrections
(4) Change in pressure altitude	ΔH_{P_c}		ft	(3) evaluated at two conditions
(5) Change in time	Δt		sec	

(6) Heading change	θ		deg
(7) Turn rate	ω	eq 8.34	deg/s
(8) Rate of descent	R/D	eq 8.35	ft/s
(9) Altitude loss to turn rate ratio	(R/D) / ω	(8) / (7)	ft/deg

Plot bank angle (left and right) versus rate of descent and turn rate as shown in Figure 8.19.

Plot bank angle versus the ratio of altitude loss to heading change as shown in Figure 8.20.

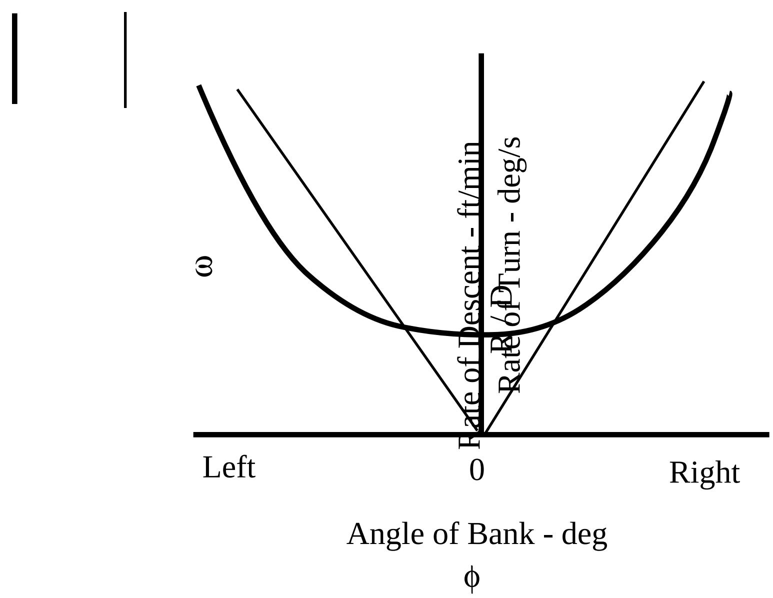

Figure 8.19
Autorotative Turning Performance

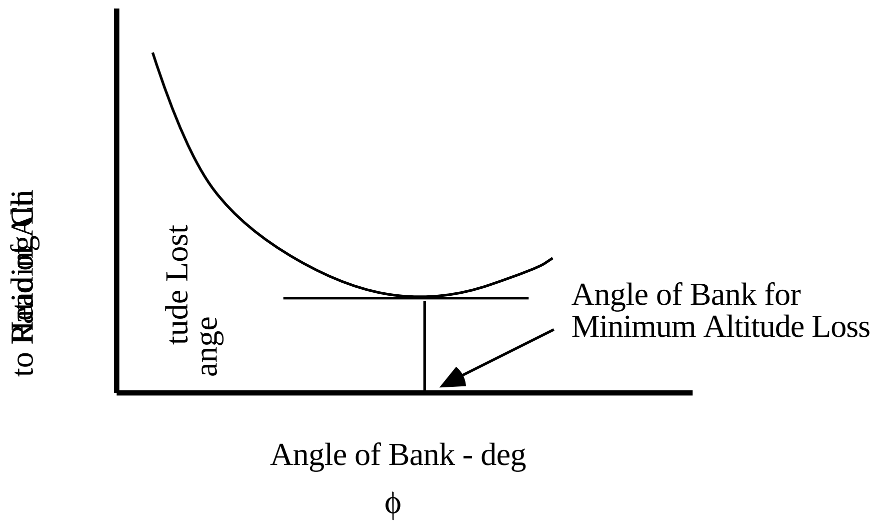

Figure 8.20
Minimum Altitude Loss in Autorotative Turn

8.5.3 USNTPS Computer Data Reduction

This section contains a brief summary of the assumptions and logic used in the USNTPS helicopter climb and descent computer data reduction programs.

Climb/descent performance is estimated using the following equation:

$$V' = \frac{(ESHP_A - ESHP_{req})\,(33{,}000)}{W} \qquad \textit{eq 8.36}$$

Where:

V'	- Predicted rate of climb or descent
$ESHP_A$	- Engine shaft horsepower available
$ESHP_{req}$	- Engine shaft horsepower required (in level flight)
W	- Weight.

Flight test data provides the correction factor and is expressed below. Care must be taken to group data relative to the ratio of forward flight speed to hover induced velocity (V_f/v_i).

V_V/V' (climb correction factor)

Where:

V_V - Actual rate of climb

V' - Predicted rate of climb.

Apply the climb correction factor to any estimated (predicted) climb rate:

$$V_v = \frac{\left(\frac{V_v}{V'}\right)}{V'} \qquad \textit{eq 8.37}$$

Where:

V_V - Actual rate of climb

V' - Predicted rate of climb.

The results of the level flight performance program are input to the climb/descent program in the form of a fourth order expression in $V_{T_{ref}}$:

$$ESHP_{ref} = E_0 + E_1 V_{T_{ref}} + E_2 V_{T_{ref}}^2 + E_3 V_{T_{ref}}^3 + E_4 V_{T_{ref}}^4 \qquad \textit{eq 8.38}$$

Where:

$ESHP_{ref}$ - Referred engine shaft horsepower

E_0, E_1, E_2, E_3, E_4 - Constants

$V_{T_{ref}}$ - Referred true airspeed.

The following equations are also used in the calculations:

$$ESHP_{ref} = \frac{ESHP_T}{\sigma_T}\left(\frac{N_{R_S}}{N_{R_T}}\right)^3 \qquad \textit{eq 8.39}$$

$$W_{ref} = \frac{W_T}{\sigma_T}\left(\frac{N_{R_S}}{N_{R_T}}\right)^2 \qquad \textit{eq 8.40}$$

$$ESHP_T = K_Q (Q) (N_{R_T}) \qquad eq\ 8.41$$

Where:

$ESHP_{ref}$	- Referred engine shaft horsepower
$ESHP_T$	- Test engine shaft horsepower
σ_T	- Test density ratio
K_Q	- Engine torque constant (unique for each aircraft)
Q	- Engine torque
N_{R_T}	- Test main rotor speed
N_{R_S}	- Standard main rotor speed
W_T	- Test weight
W_{ref}	- Referred weight.

The fourth order curve fit is also adjusted to fit the zero airspeed power required point calculated from hover performance data. The user inputs hover performance in the form of the coefficients from the HOVER performance data reduction:

$$ESHP_{ref} = A_0 + A_1 W_{ref}^{\frac{3}{2}} \qquad eq\ 8.42$$

Where:

$ESHP_{ref}$	- Referred engine shaft horsepower
A_0, A_1	- Constants
W_{ref}	- Referred weight.

Power available as a function of pressure altitude is also required for this data reduction scheme and is input in tabular form.

The following equations are used in the computerized calculations:

$$\theta = \frac{T_a}{T_{ssl}} = \frac{OAT + 273.15}{288.15} \qquad eq\ 8.43$$

Where:

θ	- Temperature ratio
T_a	- Ambient temperature
T_{ssl}	- Standard sea level temperature, 288.15°K
OAT	- Outside air temperature (total temperature).

$$\delta = \left[1 - \left(\frac{\lambda_{ssl}\ H_P}{T_{ssl}}\right)\right]^{\frac{g_{ssl}}{g_c\ \lambda_{ssl}\ R}} \qquad \textit{eq 8.44}$$

\Where:

δ	- Pressure ratio
λ_{ssl}	- Standard sea level lapse rate, 0.00198 °K/ft
T_{ssl}	- Standard sea level temperature, 288.15 °K
H_p	- Pressure altitude, ft
g_{ssl}	- Standard sea level gravity, 32.17 ft/s^2
g_c	- 32.17 lbm/slug
R	- Engineering gas constant for air, 96.03 ft - lb/lbm- °K.

$$\sigma = \frac{\delta}{\theta} \qquad \textit{eq 8.45}$$

Where:

σ	- Density ratio
δ	- Pressure ratio
θ	- Temperature ratio.

$$ESHP_{lf} = K_Q\ Q_{lf}\ N_{R_{lf}} \qquad \textit{eq 8.23}$$

$$ESHP_{climb} = K_Q\ Q_{climb}\ N_{R_{climb}} \qquad \textit{eq 8.24}$$

$$ESHP_{descent} = K_Q \, Q_{descent} \, N_{R_{descent}} \qquad \textit{eq 8.25}$$

$$V' = \frac{\left(ESHP_{clim\ b} - ESHP_{lf}\right)(33{,}000)}{W_T} \qquad \textit{eq 8.46}$$

$$V' = \frac{\left(ESHP_{descent} - ESHP_{lf}\right)(33{,}000)}{W_T} \qquad \textit{eq 8.47}$$

Where:

V'	- Predicted rate of climb
$ESHP_{climb}$	- Engine shaft horsepower in a climb
$ESHP_{descent}$	- Engine shaft horsepower in a descent
$ESHP_{lf}$	- Engine shaft horsepower in level flight
W_T	- Test weight.

$$\frac{T_T}{T_S} = \frac{OAT + 273.15}{T_{ssl} - \lambda_{ssl} H_p} \qquad \textit{eq 8.48}$$

Where:

T_T	- Test temperature
T_S	- Standard temperature (at that altitude)
OAT	- Outside air temperature (total temperature)°C
T_{ssl}	- Standard sea level temperature, 288.15 °K
λ_{ssl}	- Standard sea level lapse rate, 0.00198 °K/ft
H_P	- Pressure altitude, ft.

$$V_V = 60 \left(\frac{T_T}{T_S}\right)\left(\frac{\Delta H_p}{\Delta t}\right) \qquad \textit{eq 8.49}$$

Where:

V_V	- Actual rate of climb
T_T	- Test temperature
T_S	- Standard temperature (at that altitude)
$\Delta H_P / \Delta t$	- Flight test rate of climb, ft/s .

$$V_i = 60 \sqrt{\frac{W_T}{2 \ \rho_{ssl} \ \sigma \ A_D}} \qquad \textit{eq 8.50}$$

Where:

v_i	- Induced velocity in hover
W_T	- Test weight
ρ_{ssl}	- Standard sea level density, 0.0023769 slug/ft^3
σ	- Density ratio
A_D	- Rotor disc area.

The user is prompted to provide fuel, outside air temperature, altitude, torque, rotor speed and climb/descent rate data for each data point. Level flight performance data (coefficients of referred format) are entered next. Weighted least squares curve fits are applied to determine the coefficients of the theoretical equation form. Graphic representations are displayed for the generalized climb/descent performance data set (V_V / v_i versus V'/ v_i) and climb correction factor (V_v/V'). Ensure that sufficient data has been provided to determine accurately climb/descent correction factors for the range of forward flight speed and induced velocity ratios (V_f/v_i) required.

The final data are presented in a tabular form.

8.6 DATA ANALYSIS

8.6.1 General

Flight test climb and descent performance data are reduced to nondimensional, generalized terms. Test day data are then compared to mathematically predicted climb and descent data to derive correction factors for various ratios of forward flight speed to induced velocities. These correction factors are used to adjust mathematically predicted climb and descent data for any aircraft gross weight and atmospheric condition of interest. Do not extrapolate data. Performance data predicted outside the limits of the test day data sample used in constructing the correction factor model can be inaccurate.

8.6.2 Specific Conditions For Climb Performance

To analyze climb and descent performance data, one must relate specific aircraft gross weight and atmospheric conditions to a task or mission. The following equations are used to determine climb or descent rate predictions for user chosen aircraft gross weight, rotor speed and atmospheric conditions (normally standard or hot day) as a function of true airspeed:

Standard day:

$$\theta = 1 - \left(\frac{\lambda_{ssl}}{T_{ssl}}\right) H_P \qquad \textit{eq 8.51}$$

Hot day:

$$\theta = \frac{312.61 - \lambda_{ssl} H_P}{T_{ssl}} \qquad \textit{eq 8.52}$$

Where:

θ	- Temperature ratio
λ_{ssl}	- Standard sea level temperature lapse rate, 0.00196 °K/ft
T_{ssl}	- Standard sea level temperature, 288.15 °K
H_P	- Pressure altitude.

$$\delta = \left[1 - \left(\frac{\lambda_{ssl}\ H_P}{T_{ssl}}\right)\right]^{\frac{g_{ssl}}{g_c\ \lambda_{ssl}\ R}} \qquad \textit{eq 8.44}$$

$$\sigma = \frac{\delta}{\theta} \qquad \textit{eq 8.45}$$

$$V_i = 60\sqrt{\frac{W_T}{2\ \rho_{ssl}\ \sigma\ A_D}} \qquad \textit{eq 8.50}$$

$$V' = \frac{\left(ESHP_{clim\ b} - ESHP_{lf}\right)\ (33{,}000)}{W_T} \qquad \textit{eq 8.46}$$

$$V_v = \frac{\left(\frac{V_v}{V'}\right)}{V'} \qquad \textit{eq 8.37}$$

Use normalized climb correction factors (Figure 8.18), engine power available and engine power required for various true airspeeds to calculate specific condition climb performance capabilities as follows: (Note: Hold either weight or pressure altitude constant to simplify the process.)

Parameter	Notation	Formula	Units	Remarks
(1) Weight	W		lb	Select several weights as desired
(2) Pressure altitude	H_P		ft	Select several pressure altitudes
(3) Temperature ratio	θ	eq 8.51 or eq 8.52		Compute for each pressure altitude and atmospheric model chosen
(4) Pressure ratio	δ	eq 8.44		Compute for each pressure altitude and

				atmospheric model chosen
(5) Density ratio	σ	eq 8.45		Compute for each pressure altitude and atmospheric model chosen
(6) Rotor Speed	N_R		%	Select a main rotor speed
(7) True airspeed	V_T		KTAS	Select a range of true airspeeds
(8) Engine shaft horsepower	$ESHP_A$		hp	From Figure 7.17 for each V_T
(9) Engine shaft horsepower required for level flight	$ESHP_{lf}$		hp	From Figure 7.17 for each V_T
(10) Induced velocity	v_i	eq 8.50	ft/min	For each weight
(11) Predicted rate of climb	V'	eq 8.46		For each V_T
(12) Normalized forward flight speed	V_f / v_i	(7) (101.34)/ (10)		
(13) Normalized predicted rate of climb	V'/v_i	(11)/(10)		
(14) Climb correction factor	V_V/V'			From Figure 8.18 at each V_f / v_i and V'/v_i
(15) Actual rate of climb	V_V	eq 8.37		For each V_T

Plot actual rate of climb (V_V) versus true airspeed (V_T). The results should be similar to Figure 8.21 below.

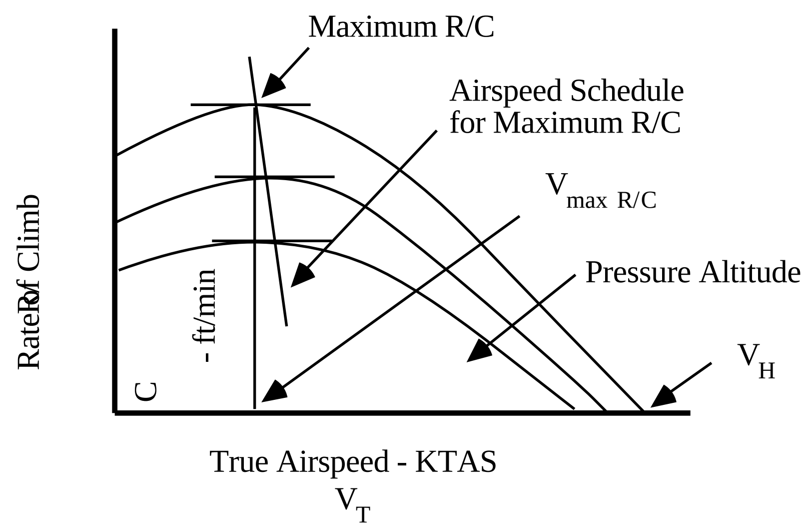

Figure 8.21
Specific Climb Performance

Many questions may be answered with this figure. If pressure altitude were introduced as a variable, the climb schedule for the continuous climb tests can be read directly for each altitude computed. The airspeed for maximum climb rate is easily obtained as well as an indication of the sensitivity of climb rate to off optimum airspeed conditions. A plot of maximum rate of climb versus pressure altitude at various altitudes will result in Figure 8.22.

The rate of climb corresponding to each of the commonly evaluated ceilings is noted on the figure. This figure is only accurate for the specific gross weight, main rotor speed, configuration and atmospheric model used in the computations.

Climb rate capabilities evaluated at specific conditions is now available to determine contract guarantees and/or specification compliance.

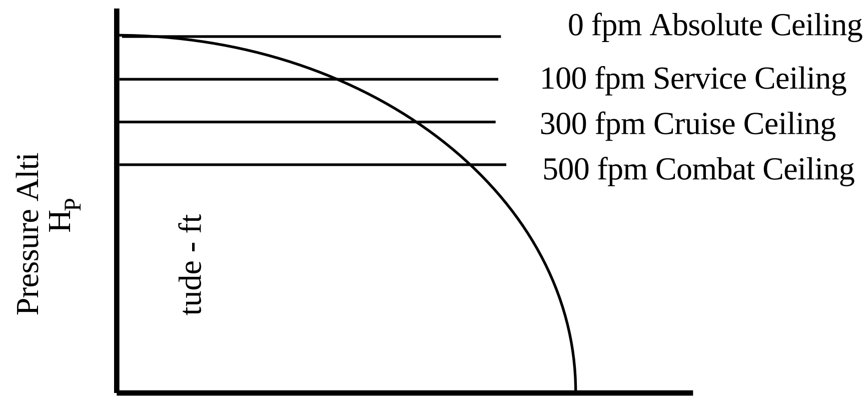

Figure 8.22
Selected Ceiling Determination

8.6.3 Specific Conditions for Autorotative Descent Performance

Normally autorotative descent performance is not generalized. The test data collected are presented for the conditions of interest. On the surface, this may seem overly restrictive, but in reality only a limited number of conditions are of interest. The minimum rate of descent airspeed, best glide airspeed and rates of descent at these speeds are of primary interest, especially at the maximum gross weight for which an autorotation is possible. Stores jettison is used sometimes in establishing this criteria. Rotor speed variations and out of trim conditions are of interest. Once the full range of required data is assembled, a matrix of test conditions can be established to ensure that all critical handbook and contractual/specification requirements are evaluated.

Data analysis begins with a plot of the rate of descent (corrected for nonstandard temperature) versus true airspeed. The items of primary interest are depicted in Figure 8.23.

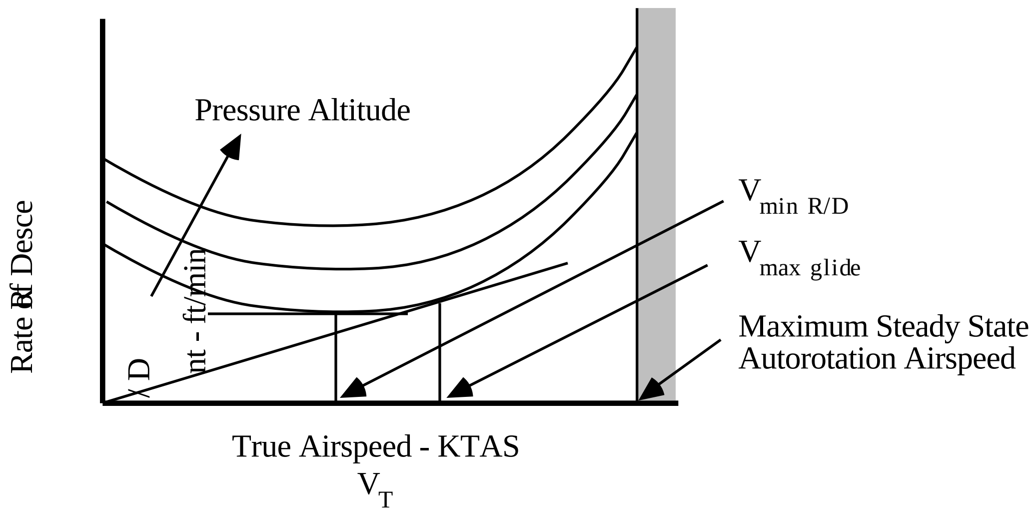

Figure 8.23
Specific Autorotative Performance

8.7 MISSION SUITABILITY

8.7.1 Climb Performance

Primarily the climb performance test is flown to determine the airspeed for maximum rate of climb for use in establishing the test climb schedule. However, there are several items that should be considered before recommending a climb airspeed. Of primary importance is how difficult is it to stabilize at the recommended climb airspeed? Secondly, how much airspeed tolerance is available without sacrificing a significant amount of climb performance?

The answer to the first question must come from a qualitative evaluation of aircraft handling qualities. The second question can be answered by examining the shape of the curve, rate of climb versus airspeed. A general rule of thumb is to accept a one percent loss in performance for flight path stability as shown graphically in Figure 8.24.

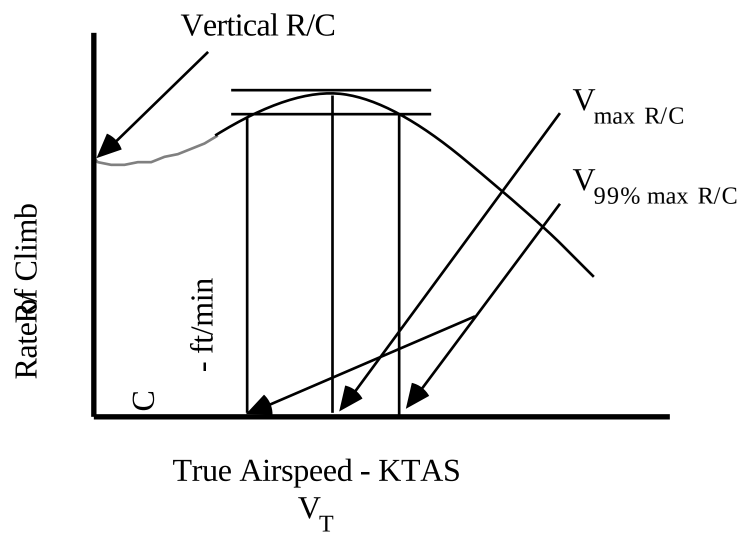

Figure 8.24
Climb Performance Mission Suitability

A line drawn at 99 percent maximum rate of climb will yield two climb airspeeds at which a one percent performance loss will occur. These two airspeeds will constitute the allowable airspeed variation that will still maintain near maximum rate of climb performance. In most cases the higher airspeed is selected by the pilot. This airspeed is on the front side of the power curve and is generally more stable, and thus easier to fly.

Additionally, the following items should always be considered when recommending a maximum rate of climb airspeed schedule:

1. Field of view.
2. Vibrations.
3. Aircraft attitude.
4. Best main rotor speed.

8.7.2 Autorotative Performance

An evaluation of the autorotative performance of a helicopter should answer the following questions:

1. What is the best airspeed for minimum rate of descent?

2. What is the minimum rate of descent?
3. What is the airspeed for maximum glide distance?
4. What is the glide capability?
5. What are the effects of airspeed and main rotor speed on glide capability?

The same consideration to flight path stability should be given to the minimum rate of descent airspeed and maximum glide distance airspeed as was given to the maximum rate of climb airspeed. Again, a general rule of thumb is to accept a one percent loss in performance for flight path stability during the autorotation, as shown in Figure 8.25.

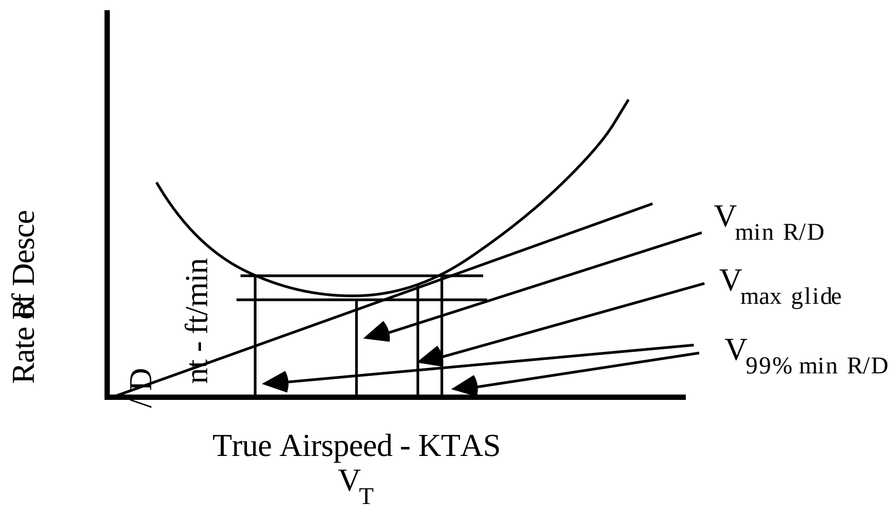

Figure 8.25
Autorotative Performance Mission Suitability

The airspeed band which will result in a one percent increase in the minimum rate of descent is shown graphically in Figure 8.25. The maximum glide distance airspeed is computed by constructing a tangent to the curve through the origin. This point is used to calculate the best glide ratio, (R/D)/airspeed.

As in climb performance, give the same consideration to field of view, vibrations, attitude, and best rpm when recommending autorotation airspeeds. One additional consideration must be given to rpm control during entry to the autorotation and during a steady state autorotation. This aspect is particularly important if the rpm has a tendency to overspeed or there is a substantial change in performance with variations in rotor speed.

Once the best speed for minimum rate of descent is established, a short evaluation of the effects of rpm on rate of descent should be accomplished at this airspeed to determine the variation of rate of descent with variations in rotor speed. Likewise, the effects of rpm on glide angle should be evaluated.

The last service suitability evaluation should be that of determining the turning performance in an autorotation, in terms of turn rate, turn radius, and rate of descent during the turn.

8.8 SPECIFICATION COMPLIANCE

Generally, there are few specification compliance items for forward flight climb or autorotative descent performance. The excess power available to establish a forward flight climb is normally evaluated with a vertical climb rate requirement as discussed in Chapter 6. When a forward flight climb requirement is stated it is usually evaluated at sea level, standard day conditions. This will require data extrapolation due to the safe altitude requirements used in gathering the data. Check the aircraft detail specification to ensure that all requirements are evaluated.

8.9 GLOSSARY

8.9.1 Notations

A_D	Rotor disc area	ft^2
A_0, A_1	Constants	
C_D	Coefficient of drag	
$\overline{C}_{d_0}$	Average blade element profile drag coefficient	
C_{D_p}	Parasite drag coefficient	
C_L	Coefficient of lift	
D	Drag	lb
dh/dt	Slope of test data time history	ft/min
E_0, E_1,E_2,E_3,E_4	Constants	
ESGW	Engine start gross weight	lb
ESHP	Engine shaft horsepower	hp

$ESHP_A$	Engine shaft horsepower available	hp
$ESHP_{climb}$	Engine shaft horsepower in climb	hp
$ESHP_{descent}$	Engine shaft horsepower in a descent	hp
$ESHP_{lf}$	Engine shaft horsepower in level flight	hp
$ESHP_{ref}$	Referred engine shaft horsepower	hp
$ESHP_{req}$	Engine shaft horsepower required	hp
$ESHP_T$	Test engine shaft horsepower	hp
$\Delta ESHP$	Excess engine shaft horsepower	hp
F	Force	lb
FC	Fuel counts	cts
FU	Fuel used	lb
g_c	32.17 lbm/ slug	
g_{ssl}	Standard sea level gravity	32.17 ft/s^2
GW	Gross weight	lb
h	Height	ft
H_P	Pressure altitude	ft
H_{P_c}	Calibrated pressure altitude	ft
H_{P_o}	Observed pressure altitude	ft
$\Delta H_{P_{ic}}$	Altimeter instrument correction	ft
ΔH_{P_c}	Change in calibrated pressure altitude	ft
$\Delta H_P/\Delta t$	Flight test rate of climb	ft/min
ΔH_{pos}	Altimeter position error correction	ft
K_{Hp}	Temperature correction factor	
K_P	Power correction factor	
K_Q	Engine torque constant (aircraft unique)	
K_T	Temperature recovery factor	
K_W	Weight correction factor	
L	Lift	lb
MRP	Military rated power	
N_R	Main rotor speed	%, rpm

$N_{R\ climb}$	Main rotor speed in a climb	%, rpm
$N_{R\ descent}$	Main rotor speed in a descent	%, rpm
$N_{R\ lf}$	Main rotor speed in level flight	%, rpm
N_{R_S}	Standard main rotor speed	%, rpm
N_{R_T}	Test main rotor speed	%, rpm
OAT	Outside air temperature (total temperature)	°C
P	Power	hp
PE	Potential energy	ft-lbf
P_i	Induced power	hp
$P_{MR\ climb}$	Main rotor power in a climb	hp
$P_{MR\ lf}$	Main rotor power in level flight	hp
P_0	Profile power	hp
P_p	Parasite power	hp
Q	Engine torque	%, psi
Q_{climb}	Engine torque in a climb	%, psi
$Q_{descent}$	Engine torque in a descent	%, psi
Q_{lf}	Engine torque in level flight	%, psi
r	Radius to blade element	ft
R	Engineering gas constant for air	96.03 ft - lb/lb_m- °K
	Rotor radius	ft
	Resultant aerodynamic force	lb
R/C	Rate of climb	ft/min
R/D	Rate of descent	ft/min
(R/D)/ω	Altitude loss to turn rate ratio	
SHP	Shaft horsepower	hp
ΔSHP	Change in shaft horsepower	hp
t	Time	min, s
Δt	Change in time	s
T_a	Ambient temperature	°C

T_o	Observed temperature	°C
T_S	Standard temperature (at that altitude)	°C
T_{ssl}	Standard sea level temperature	288.15 °K
T_T	Test temperature	°C
ΔT_{ic}	Temperature instrument correction	°C
T	Thrust	lb
v_i	Induced velocity at hover	ft/min
v_{i_v}	Induced velocity in a vertical climb	ft/min
V'	Predicted rate of climb or descent	ft/min
V'/v_i	Normalized predicted rate of climb	
V	Velocity	kn
V_c	Calibrated airspeed	KCAS
V_f	Forward flight velocity	kn
V_f/v_i	Normalized forward flight velocity	
V_H	Maximum level flight airspeed	kn
$V_{max\ glide}$	Airspeed for maximum autorotation glide range	kn
$V_{max\ R/C}$	Airspeed for maximum rate of climb	kn
$V_{99\%\ max\ R/C}$	Airspeed for 99% maximum rate of climb	kn
$V_{min\ R/D}$	Airspeed for minimum rate of descent	kn
$V_{99\%\ min\ R/D}$	Airspeed for 99% minimum rate of descent	kn
V_o	Observed airspeed	KOAS
V_R	Resultant velocity	kn
V_T	True airspeed	KTAS
$V_{T_{ref}}$	Referred true airspeed	KTAS
V_V	Actual rate of climb or descent	ft/min
V_V/V'	Climb correction factor	
V_V/v_i	Normalized actual rate of climb or descent	
ΔV_{ic}	Airspeed instrument correction	kn
ΔV_{pos}	Airspeed position error	kn

W	Weight	lb
W_{ref}	Referred weight	lb
W_T	Test weight	lb

8.9.2 Greek Symbols

α (Alpha)	Rotor angle of attack	deg
β (Beta)	Sideslip angle	deg
β_o (Beta)	Observed sideslip angle	deg
δ (Delta)	Pressure ratio	
λ_{ssl}	Standard sea level lapse rate	0.00198 °K/ft
ρ (Rho)	Density	slug/ft^3
ρ_{ssl}	Standard sea level air density	0.0023769 slug/ft^3
σ (Sigma)	Density ratio	
σ_R	Rotor solidity ratio	
σ_T	Test density ratio	
θ (Theta)	Collective blade angle	deg
	Temperature ratio	
	Heading change	deg
Ω (Omega)	Rotor angular velocity	rad/s
ω (Omega)	Turn rate	deg/s
Ωr	Blade element rotational velocity	ft/s
ΩR	Blade rotational velocity	ft/s
ϕ (Phi)	Bank angle	deg
μ (Mu)	Advance ratio	
φ	Inflow angle	deg
φ'	Angle between resultant and lift vectors	deg

8.10 REFERENCES

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APPENDIX I

GLOSSARY

APPENDIX I

GLOSSARY

NOTATIONS

a	Speed of sound	kn, ft/min
a_{ssl}	Standard sea level speed of sound	661.23 kn
A	Area	ft^2
A_0, A_1, A_2, A_3	Constants	
A_f	Frontal area	ft^2
A_D	Rotor disc area	ft^2
AGL	Above ground level	
APU	Auxillary power unit	
ASW	Antisubmarine Warfare	
b	Number of blades	
	Wing span	ft
B_0, B_1, B_2, B_3	Constants	
c	Blade chord	ft
C_0, C_1, C_2, C_3	Constants	
CG	Center of gravity	in
C_D	Drag coefficient	
C_{d_0}	Blade element profile drag coefficient	
$\bar{C}_{d_0}$	Average blade element profile drag coefficient	
C_{D_0}	Profile drag coefficient	
C_{d_i}	Blade element induced drag coefficient	
$\bar{c}_{d_i}$	Average blade element induced drag coefficient	
C_{D_p}	Parasite drag coefficient	
C_ℓ	Blade element lift coefficient	

$\overline{C}_\ell$	Average blade element lift coefficient	
C_L	Lift coefficient	
C_P	Power coefficient	
C_T	Thust coefficient	
d	Horizontal distance	ft, in
dD_0	Blade element profile drag	lb
dF	Blade element torque force	lb
dh/dt	Slope of test day time history	ft/min
dL	Blade element lift	lb
dP	Blade element power required	lb
dP_0	Blade element profile power	lb
dr	Blade element radius	in
dR	Blade element resultant aerodynamic force	lb
dS	Blade element area	ft^2
dT	Blade element thust	lb
D	Course length	ft
	Diameter	ft
	Drag	lb
DT IIA	Developmental Test IIA	
D_0, D_1, D_2, D_3	Constants	
D_{eff}	Effective drag	lb
D_p	Parasite drag	lb
E_0, E_1, E_2, E_3	Constants	
ESGW	Engine start gross weight	lb
ESHP	Engine shaft horsepower	hp
$ESHP_A$	Engine shaft horsepower available	hp
$ESHP_{climb}$	Engine shaft horsepower in climb	hp
$ESHP_{corr}$	Corrected engine shaft horsepower	hp
$ESHP_{descent}$	Engine shaft horsepower in a descent	hp
$ESHP_h$	Engine shaft horsepower in a hover	hp
$ESHP_{lf}$	Engine shaft horsepower in level flight	hp
$ESHP_{max}$	Maximum permissible engine shaft horsepower	hp
$ESHP_{ref}$	Referred engine shaft horsepower	hp

$ESHP_{req}$	Engine shaft horsepower required	hp
$ESHP_S$	Standard engine shaft horsepower	hp
$ESHP_T$	Test engine shaft horsepower	hp
$\Delta ESHP$	Excess engine shaft horsepower	hp
f	Equivalent flat plate area	ft^2
F	Force	lb
dF	Blade element torque force	lb
FAR	Federal Aviation Regulations	
FC	Fuel counts	cts
FOD	Foreign object damage	
FTM	Flight Test Manual	
FU	Fuel used	lb, gal
g_c	Gravity conversion constant	32.17 lb - mass / slug
g_{ssl}	Standard sea level gravity	32.17 ft/ s^2
GW	Gross weight	lb
h	Height	ft
$h_{a/c}$	Aircraft height above ground	ft
h_{ref}	Reference altimeter height AGL	ft
HHMD	Hover height measuring device	
HOGE	Hover out-of-ground effect	
H_2	Profile drag caused by spanwise flow	lb
H_D	Density altitude	ft
H_G	Gear height AGL	ft
H_P	Pressure altitude	ft
H_{P_c}	Calibrated pressure altitude	ft
$H_{P_{c\,twr}}$	Tower calibrated pressure altitude	ft
H_{P_h}	Pressure altitude at a hover	ft
$H_{P_{ic}}$	Instrument corrected pressure altitude	ft
$H_{P_{ic\,ref}}$	Reference instrument corrected pressure altitude	ft
$H_{P_{ic\,ship}}$	Ship instrument corrected pressure altitude	ft

H_{P_o}	Observed pressure altitude	ft
$H_{P_{o\,ref}}$	Reference observed pressure altitude	ft
$H_{P_{o\,ship}}$	Ship observed pressure altitude	ft
$H_{P_{o\,twr}}$	Tower observed pressure altitude	ft
Δh	Aircraft height above tower	ft
$\Delta h/\Delta t$	Flight test rate of climb	ft/min
ΔH_P	Change in pressure altitude	ft
$\Delta H_{P_{ic}}$	Altimeter instrument correction	ft
$\Delta H_{P_{ic\,ref}}$	Reference altimeter instrument correction	ft
$\Delta H_{P_{ic\,ship}}$	Ship altimeter instrument correction	ft
$\Delta H_{P_{ic\,twr}}$	Tower altimeter instrument correction	ft
ΔH_{pos}	Altimeter position error	ft
$\Delta H_{pos\,ref}$	Reference altimeter position error	ft
$\Delta H_{pos\,ship}$	Ship altimeter position error	ft
IGE	In-ground effect	
IFR	Instrument Flight Rules	
ILS	Instrument landing system	
K_{GR}	Gear ratio (aircraft unique)	
K_{H_p}	Temperature correction factor	
K_P	Power correction factor	
K_Q	Engine torque constant (aircraft unique)	
K_T	Temperature recovery factor	
K_W	Weight correction factor	
KCAS	Knots calibrated airspeed	
KEAS	Knots equivalent airspeed	
KIAS	Knots indicated airspeed	
KOAS	Knots observed airspeed	

KTAS	Knots true airspeed	
ℓ_{TR}	Moment arm of the tail rotor	ft
L	Lift	lb
dL	Blade element lift	lb
m	Mass	lb-mass
M	Mach number	
M_2	Inlet Mach number	
M_B	Advancing blade Mach number	
M_{TIP}	Blade tip Mach number	
M'	Figure of merit	
MilSpec	Military Specification	
MRP	Military rated power	
NAWC	Naval Air Warfare Center	
NTE	Navy Technical Evaluation	
N_1, N_g	Engine gas generator speed	%, rpm
$N_{1_{corr}}$	Corrected engine gas generator speed	%, rpm
$N_{1_{max}}$	Maximum permissible engine gas generator speed	%, rpm
N_2, N_f	Engine power turbine speed	%, rpm
N_f, N_2	Engine power turbine speed	%, rpm
N_g, N_1	Engine gas generator speed	%, rpm
N_R	Main rotor speed	%, rpm
$N_{R\ climb}$	Main rotor speed in a climb	%, rpm
$N_{R\ descent}$	Main rotor speed in a descent	%, rpm
$N_{R\ h}$	Main rotor speed in a hover	%, rpm
$N_{R\ lf}$	Main rotor speed in level flight	%, rpm
N_{R_S}	Standard main rotor speed	%, rpm
N_{R_T}	Test main rotor speed	%, rpm
$N_R / \sqrt{\theta}$	Main rotor speed to the square root of the temperature ratio	%, rpm
OAT	Outside air temperature (total temperature)	°C

OGE	Out-of-ground effect	
P	Power	hp
P_0	Profile power	hp
P_{0_h}	Profile power at hover	hp
$P_{0_{H2}}$	Profile power due to spanwise flow	hp
P_{0_v}	Profile power in a vertical climb	hp
P_A	Power available	hp
P_{A_h}	Power available at hover	hp
P_{A_v}	Power available in a vertical climb	hp
P_{Acc}	Accessory power required	hp
P_b	Power required for b blades	hp
P_i	Induced power	hp
P_{i_h}	Induced power at hover	hp
$P_{i_{lf}}$	Induced power in level flight	hp
P_{i_v}	Induced power in a vertical climb	hp
P_{loss}	Power loss	hp
P_m	Miscellaneous power	hp
P_{MR}	Main rotor power required	hp
$P_{MR_{climb}}$	Main rotor power required in a climb	hp
$P_{MR_{lf}}$	Main rotor power required in level flight	hp
P_p	Parasite power	hp
P_{p_h}	Parasite power at hover	hp
P_{p_v}	Parasite power in a vertical climb	hp
P_{req}	Power required	hp
P_{Total}	Total power required	hp
P_{TR}	Tail rotor power required	hp
$P_{\Delta PE}$	Power required to change potential energy	hp

ΔP	Excess power	hp
ΔP_h	Excess power at hover	hp
ΔP_v	Excess power in a vertical climb	hp
P	Pressure	psi, psf
P_a	Ambient pressure	psi, psf
P_s	Static pressure	psi, psf
$P_{s\ ref}$	Reference static pressure	in Hg
$P_{s\ ic\ ship}$	Ship instrument corrected static pressure	in Hg
P_{s_2}	Inlet static pressure (at engine compressor face)	psi
$\bar{P}_{s_2}$	Average inlet static pressure (at engine compressor face)	psi
P_{ssl}	Standard sea level pressure	14.7psi, 2116.2 psf
P_T	Total pressure (free stream)	psi, psf
P_{T_d}	Downstream total pressure	psi, psf
P_{T_u}	Upstream total pressure	psi, psf
P_{T_2}	Inlet total pressure (at engine compressor face)	psi
$\bar{P}_{T_2}$	Average inlet total pressure (at engine compressor face)	psi
ΔP	Change in pressure	psi
$\Delta P_{s\ ship}$	Ship static pressure error	in Hg
PE	Potential energy	ft-lbf
q_c	Dynamic pressure	psi, psf
q_{ci}	Indicated dynamic pressure	psi, psf
q_e	Blade element dynamic pressure	psi, psf
q_w	Wake dynamic pressure	psi, psf
Q	Engine torque	%, psi, lb-in
Q_{climb}	Engine torque in a climb	%, psi, lb-in
$Q_{descent}$	Engine torque in a descent	%, psi, lb-in
Q_h	Engine torque at hover	%, psi, lb-in

Q_{lf}	Engine torque in level flight	%, psi, lb-in
Q_{TR}	Torque developed by the tail rotor	%, psi, lb-in
r	Radius to blade element	ft
R	Rotor radius	ft
	Resultant aerodynamic force	lb
R	Engineering gas constant for air	96.03 ft - lb/ lb_m- °K
R_n	Reynolds number	
dR	Blade element resultant aerodynamic force	lb
R/C	Rate of climb	ft/min
(R/C) / ω	Altitude loss to turn rate ratio	
R/D	Rate of descent	ft/min
RSHP	Rotor shaft horsepower	hp
$RSHP_A$	Rotor shaft horsepower available	hp
$RSHP_{req}$	Rotor shaft horsepower required	hp
$RSHP_S$	Standard rotor shaft horsepower	hp
$RSHP_T$	Test rotor shaft horsepower	hp
ΔRSHP	Excess rotor shaft horsepower	hp
S	Wing area	ft^2
dS	Blade element area	ft^2
SAS	Stability augmentation system	
SFC	Specific fuel consumption	lb/h/hp
SHP	Shaft horsepower	hp
ΔSHP	Change in shaft horsepower	hp
SR	Specific range	nm/lb fuel
t	Time	h, min, s
Δt	Elapsed time	h, min, s
T	Temperature	°C
T_5	Power turbine inlet temperature	°C
$T_{5_{corr}}$	Corrected power turbine inlet temperature	°C

$T_{5_{max}}$	Maximum permissible power turbine inlet tempeature	°C
T_a	Ambient temperature	°C
T_{a_H}	Hot day ambient temperature	°C
T_{a_S}	Standard day ambient temperature	°C
T_{ic}	Instrument corrected temperature	°C
T_o	Observed temperature	°C
T_s	Static temperature	°C
T_{s_2}	Inlet static temperature (at engine compressor face)	°C
T_{ssl}	Standard sea level temperature	15°C, 288.15°K
T_S	Standard temperature	°C
T_T	Test temperature	°C
T_T	Total temperature	°C
T_{T_2}	Inlet total temperature (at engine compressor face)	°C
ΔT_{ic}	Temperature instrument correction	°C
T	Thrust	lb
	Cable tension	lb
T_b	Thrust of b blades	lb
T_h	Thrust at hover	lb
T_f	Thrust in forward flight	lb
T_{TR}	Tail rotor thrust	lb
T_v	Thrust in a vertical climb	lb
dT	Blade element thrust	lb
USNTPS	United States Naval Test Pilot School	
v_e	Blade element velocity	ft/min, ft/s
v_i	Induced velocity at hover	ft/min, ft/s
v_{i_f}	Induced velocity in forward flight	ft/min, ft/s

v_{ih}	Horizontal component of v_i	ft/min, ft/s
v_{i_v}	Induced velocity in a vertical climb	ft/min, ft/s
v_w	Ultimate wake velocity	ft/min, ft/s
V	Velocity	kn
V_c	Calibrated airspeed	KCAS
V_{cruise}	Cruise airspeed	kn
V_e	Equivalent airspeed	KEAS
V_f	Forward flight velocity	kn
V_f/v_i	Normalized forward flight velocity	
V_G	Ground speed	kn
V_H	Maximum level flight airspeed	kn
V_i	Indicated airspeed	KIAS
V_{ic}	Instrument corrected airspeed	KIAS
$V_{ic\ ship}$	Ship instrument corrected airspeed	KIAS
V_{max}	Airframe limited level flight airspeed	kn
$V_{max\ end}$	Maximum endurance airspeed	kn
$V_{max\ glide}$	Airspeed for maximum autorotation glide range	kn
$V_{max\ range}$	Maximum range airspeed	kn
$V_{max\ R/C}$	Maximum rate of climb airspeed	kn
$V_{99\%\ max\ R/C}$	Airspeed for 99% maximum rate of climb	kn
$V_{min\ R/D}$	Minimum rate of descent airspeed	kn
$V_{99\%\ min\ R/D}$	Airspeed for 99% minimum rate of descent	kn
V_{NE}	Never exceed airspeed	kn
$V_{(P/L)max}$	Airframe maximum range airspeed	kn
V_o	Observed airspeed	KOAS
$V_{o\ ref}$	Reference observed airspeed	KOAS
$V_{o\ ship}$	Ship observed airspeed	KOAS
V_R	Resultant velocity (through rotor)	ft/s
V_T	True airspeed	KTAS
$V_{T_{ref}}$	Referred true airspeed	KTAS
V_{TIP}	Advancing blade tip speed	kn, ft/s
V_v	Actual vertical velocity	kn

V'	Predicted vertical velocity	kn
V_V/V'	Climb correction factor	
V_V/v_i	Normalized actual rate of climb / descent	
V'/v_i	Normalized predicted rate of climb / descent	
ΔV	Change in velocity	kn, ft/s
ΔV_{ic}	Airspeed instrument correction	kn
$\Delta V_{ic\ ref}$	Reference airspeed instrument correction	kn
$\Delta V_{ic\ ship}$	Ship airspeed instrument correction	kn
ΔV_{pos}	Airspeed position error	kn
$\Delta V_{pos\ ref}$	Reference airspeed position error	kn
$\Delta V_{pos\ ship}$	Ship airspeed position error	kn
W	Weight	lb
W_{ref}	Referred weight	lb
W_S	Standard weight	lb
W_T	Test weight	lb
W/δ	Weight to pressure ratio	lb
W/σ	Weight to density ratio	lb
W_f	Fuel flow	lb/h
$W_{f_{corr}}$	Corrected fuel flow	lb/h
$W_{f_{max}}$	Maximum permissible fuel flow	lb/h

GREEK SYMBOLS

α (Alpha)	Angle of attack	deg
α_A	Airframe angle of attack	deg
α_i	Angle of incidence	deg
α_{ic}	Instrument corrected angle of attack	deg
α_o	Observed angle of attack	deg
α_{o_h}	Observed angle of attack at hover	deg
α_{o_v}	Observed angle of attack in a vertical climb	deg
α_p	Probe angle of attack	deg
α_R	Rotor angle of attack	deg
$\Delta\alpha_{ic}$	Angle of attack instrument correction	deg
β (Beta)	Sideslip angle	deg
β_A	Airframe sideslip angle	deg
β_i	Sideslip angle of incidence	deg
β_{ic}	Instrument corrected sideslip angle	deg
β_o	Observed sideslip angle	deg
β_p	Probe sideslip angle	deg
$\Delta\beta_{ic}$	Sideslip angle instrument correction	deg
δ (Delta)	Pressure ratio, $\delta = P/P_{ssl}$	
δ_a	Ambient pressure ratio, $\delta_a = P_a / P_{ssl}$	
δ_{s_2}	Static pressure ratio, $\delta_{s_2} = P_{s_2} / P_{ssl}$	
δ_{T_2}	Inlet pressure ratio, $\delta_{T_2} = P_{T_2} / P_{ssl}$	

γ (Gamma)	Ratio of specific heats (Air: $\gamma = 1.4$)	
	Flight path angle	deg
λ (Lambda)	Lag error constant	
	Lapse rate	°K/ft
λ_S	Static pressure lag error constant	
λ_T	Total pressure lag error constant	
λ_{ssl}	Standard sea level lapse rate	0.00198 °K/ft
η_m (Eta)	Mechanical efficiency, η_m = RSHP/ESHP	
θ (Theta)	Temperature ratio, $\theta = T/T_{ssl}$	
θ_a	Ambient temperature ratio, $\theta_a = T_a / T_{ssl}$	
θ_{s_2}	Static temperature ratio, $\theta_{s_2} = T_{s_2} / T_{ssl}$	
θ_{T_2}	Inlet temperature ratio, $\theta_{T_2} = T_{T_2} / T_{ssl}$	
θ	Angle	deg
	Collective blade angle	deg
	Heading change	deg
μ (Mu)	Advance ratio	
π (Pi)	Numerical constant	
ρ (Rho)	Density	slug/ft^3
ρ_a	Ambient air density	slug/ft^3
ρ_{ssl}	Standard sea level air density	0.0023769 slug/ft^3
ρ_S	Stsndard air density	slug/ft^3
ρ_T	Test air density	slug/ft^3
σ (Sigma)	Density ratio, $\sigma = \rho / \rho_{ssl}$, $\sigma = \delta / \theta$	
σ_T	Test density ratio	

σ_i	Induced angle	deg
σ_R	Rotor solidity ratio, $\sigma_R = bc/\pi R$	
ϕ (Phi)	Bank angle	deg
φ	Inflow angle	deg
φ'	Angle between resultant and lift vectors	deg
ψ (Psi)	Blade azimuth angle	deg
ω (Omega)	Turn rate	deg/s
Ω (Omega)	Rotor angular velocity	rad/s
Ωr	Blade element rotational velocity	ft/s
ΩR	Blade rotational velocity	ft/s
ΩR_S	Standard blade rotational velocity	ft/s
ΩR_T	Test blade rotational velocity	ft/s

SUBSCRIPTS

a	Ambient
a/c	Aircraft
A	Airframe
	Available
Acc	Accessory
B	Advancing blade
c	Calibrated
	Dynamic
ci	Indicated dynamic
corr	Corrected
d	Element drag
D	Disc
	Drag
	Density
e	Blade element

	Equivalent
eff	Effective
end	Endurance
f	Forward flight
	Power turbine
G	Ground
GR	Gear ratio
h	Hover
i	Incidence
	Indicated
	Induced
ic	Instrument corrected
ℓ	Element lift
lf	Level flight
loss	Loss
L	Lift
m	Mechanical
	Miscellaneous
max	Maximum
min	Minimum
MR	Main rotor
NE	Never exceed
o (lower case)	Observed
p (lower case)	Parasite
	Probe
pos	Position error
P (Upper case)	Power
	Pressure
ref	Referred
req	Required
R	Resultant
	Rotor

R/C	Rate of climb
R/D	Rate of descent
$_s$ (lower case)	static
ssl	Standard sea level
$_S$ (Upper case)	Standard
T	Test
	Temperature
	Thust
	Total
	True
TIP	Tip
TR	Tail rotor
$_v$ (lower case)	Vertical
$_V$ (Upper case)	Velocity
$_w$ (lower case)	Wake
$_W$ (Upper case)	Weight
$_0$ (Zero)	Initial
	Profile

TERMS

Flight idle glide	Minimum power descent.
Flight idle stop	The minimum fuel flow which can be delivered with the throttle in the fly position.
Static droop	The change in N_R, from initial to final steady state, caused by a change in power demand.
Transient droop	The change in N_R, from minimum to final steady state, caused by the rate of change in power demand.
Inlet pressure recovery ratio	$\frac{\bar{P}_{T_2}}{P_a}$, i.e. the average total pressure at the compressor face divided by the free stream ambient pressure.
Inlet temperature difference	$T_{T_2} - T_a$, i.e. the total temperature at the compressor face minus the free stream ambient pressure.

APPENDIX II

REFERENCES

APPENDIX II

REFERENCES

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APPENDIX III

FIGURES

APPENDIX III

FIGURES

CHAPTER 4

CHAPTER 5

CHAPTER 6

CHAPTER 7

CHAPTER 8

APPENDIX IV

TABLES

APPENDIX IV

TABLES

APPENDIX V

EQUATIONS

APPENDIX V

EQUATIONS

CHAPTER 2

$$\Delta H_{pos} = f(P_s - P_a)$$ *eq 2.1*

$$V_T^2 = \frac{2\gamma}{\gamma - 1}\frac{P_a}{\rho_a}\left[\left(\frac{P_T - P_a}{P_a} + 1\right)^{\frac{\gamma - 1}{\gamma}} - 1\right]$$ *eq 2.2*

$$V_c^2 = \frac{2\gamma}{\gamma - 1}\frac{P_{ssl}}{\rho_{ssl}}\left[\left(\frac{P_T - P_a}{P_{ssl}} + 1\right)^{\frac{\gamma - 1}{\gamma}} - 1\right]$$ *eq 2.3*

$$V_c = f(P_T - P_a) = f(q_c)$$ *eq 2.4*

$$V_i = f(P_T - P_s) = f(q_{ci})$$ *eq 2.5*

$$\Delta V_{pos} = f(P_s - P_a), q_c$$ *eq 2.6*

$$P_s + \frac{\rho V^2}{2} = \text{Constant}$$ *eq 2.7*

$$P_a + \frac{\rho V^2}{2} = P_T$$ *eq 2.8*

$$V = \sqrt{\frac{2(P_T - P_a)}{\rho}}$$ *eq 2.9*

$$V_e = \sqrt{\frac{2(P_T - P_a)}{0.0023769}}$$ *eq 2.10*

$$V_T = \frac{V_e}{\sqrt{\sigma}}$$ *eq 2.11*

$$H_{P_{ic}} = H_{P_o} + \Delta H_{P_{ic}}$$ *eq 2.12*

$$H_{P_{ic\ ref}} = H_{P_{o\ ref}} + \Delta H_{P_{ic\ ref}}$$ *eq 2.13*

$$H_{P_c} = H_{P_{ic\ ref}} - h_{ref} + h_{a/c}$$ *eq 2.14*

$$\Delta H_{pos} = H_{P_c} - H_{P_{ic}}$$ *eq 2.15*

$$V_{ic} = V_o + \Delta V_{ic}$$ *eq 2.16*

$$V_{G1} = 0.5917\left(\frac{D}{\Delta t_1}\right)$$ *eq 2.17*

$$V_{G2} = 0.5917\left(\frac{D}{\Delta t_2}\right)$$ *eq 2.18*

$$V_T = \frac{V_{G1} + V_{G2}}{2}$$ *eq 2.19*

$$T_a = T_o + \Delta T_{ic}$$ *eq 2.20*

$$V_c = \frac{V_T}{\sqrt{\sigma}}$$ *eq 2.21*

$$\Delta V_{pos} = V_c - V_{ic} \qquad \textit{eq 2.22}$$

$$\alpha_{ic} = \alpha_o + \Delta\alpha_{ic} \qquad \textit{eq 2.23}$$

$$\beta_{ic} = \beta_o + \Delta\beta_{ic} \qquad \textit{eq 2.24}$$

$$H_{P_c} = H_{P_{o\ ref}} + \Delta H_{P_{ic\ ref}} + \Delta H_{pos\ ref} \qquad \textit{eq 2.25}$$

$$H_{P_{ic\ ship}} = H_{P_{o\ ship}} + \Delta H_{P_{ic\ ship}} \qquad \textit{eq 2.26}$$

$$\Delta H_{pos\ ship} = H_{P_c} - H_{P_{ic\ ship}} \qquad \textit{eq 2.27}$$

$$V_c = V_{o\ ref} + \Delta V_{ic\ ref} + \Delta V_{pos\ ref} \qquad \textit{eq 2.28}$$

$$V_{ic\ ship} = V_{o\ ship} + \Delta V_{ic\ ship} \qquad \textit{eq 2.29}$$

$$\Delta V_{pos\ ship} = V_c - V_{ic\ ship} \qquad \textit{eq 2.30}$$

$$H_{P_{c\ twr}} = H_{P_{o\ twr}} + \Delta H_{P_{ic\ twr}} \qquad \textit{eq 2.31}$$

$$\Delta h = d \tan\theta \qquad \textit{eq 2.32}$$

$$H_{P_c} = H_{P_{ctwr}} + \Delta h \qquad \textit{eq 2.33}$$

$$\Delta P_{s\ ship} = P_{s\ ref} - P_{s_{ic\ ship}} \qquad \textit{eq 2.34}$$

$$T_a\ (°K) = T_a\ (°C) + 273.15 \qquad \textit{eq 2.35}$$

$$a = 38.9678\ \sqrt{T_a\ (°K)} \qquad \textit{eq 2.36}$$

$$\Delta V_{pos} = \left[\frac{\Delta P_{s\ ship}\ (a^2)}{41.888\ (V_{ic})}\right]\left[\frac{2.5}{1+0.2\left(\frac{V_{ic}}{a}\right)^2}\right]$$ *eq 2.37*

$$V_c = V_{ic} + \Delta V_{pos}$$ *eq 2.38*

CHAPTER 4

$$\delta = \frac{P}{P_{ssl}}$$ *eq 4.1*

$$\theta = \frac{T}{T_{ssl}}$$ *eq 4.2*

$$\delta_a = \frac{P_a}{P_{ssl}}$$ *eq 4.3*

$$\theta_a = \frac{T_a}{T_{ssl}}$$ *eq 4.4*

$$\delta_{s_2} = \frac{P_{s_2}}{P_{ssl}}$$ *eq 4.5*

$$\theta_{s_2} = \frac{T_{s_2}}{T_{ssl}}$$ *eq 4.6*

$$\delta_{T_2} = \frac{P_{T_2}}{P_{ssl}}$$ *eq 4.7*

$$\theta_{T_2} = \frac{T_{T_2}}{T_{ssl}}$$ *eq 4.8*

$$\delta_{T_2} = \frac{P_{T_2}}{P_{ssl}} = f\left(V_c,\ \delta_a\right)$$ *eq 4.9*

$$\theta_{T_2} = \frac{T_{T_2}}{T_{ssl}} = f\left(V_c, \theta_a\right)$$ *eq 4.10*

$$\frac{ESHP}{\delta\sqrt{\theta}} = f\left(\frac{N_1}{\sqrt{\theta}},\ \frac{N_2}{\sqrt{\theta}},\ \frac{V_T}{\sqrt{\theta}},\ R_n\right)$$ *eq 4.11*

$$\frac{W_f}{\delta\sqrt{\theta}} = \left(\frac{N_1}{\sqrt{\theta}},\ \frac{N_2}{\sqrt{\theta}},\ \frac{V_T}{\sqrt{\theta}},\ R_n\right)$$ *eq 4.12*

$$ESHP_{corr} = \frac{ESHP}{\delta_{T_2}\sqrt{\theta_{T_2}}}$$ *eq 4.13*

$$W_{f_{corr}} = \frac{W_f}{\delta_{T_2}\sqrt{\theta_{T_2}}}$$ *eq 4.14*

$$T_{5_{corr}} = \frac{T_5}{\theta_{T_2}}$$ *eq 4.15*

$$N_{1_{corr}} = \frac{N_1}{\sqrt{\theta_{T_2}}}$$ *eq 4.16*

$$ESHP_{corr} = f(N_{1_{corr}})$$ *eq 4.17*

$$W_{f_{corr}} = f(N_{1_{corr}})$$ *eq 4.18*

$$T_{5_{corr}} = f(N_{1_{corr}})$$ *eq 4.19*

$$W_{f_{corr}} = f(ESHP_{corr})$$ *eq 4.20*

$$SFC = \frac{W_{f_{corr}}}{ESHP_{corr}}$$ *eq 4.21*

$$\bar{P}_{s_2} = \frac{\sum P_{s_2}}{n}$$ *eq 4.22*

$$\bar{P}_{T_2} = \frac{\sum P_{T_2}}{n}$$ *eq 4.23*

$$M_2 = \sqrt{\left(\frac{2}{\gamma - 1}\right)\left[\left(\frac{\bar{P}_{T_2}}{\bar{P}_{s_2}}\right)^{\frac{\gamma - 1}{\gamma}} - 1\right]}$$ *eq 4.24*

$$T_{T_2} = T_{s_2}\left[1 + \left(\frac{\gamma - 1}{2}\right)M_2^2\right]$$ *eq 4.25*

$$\delta_{T_2} = \frac{\bar{P}_{T_2}}{P_{ssl}} = \left(\frac{\bar{P}_{T_2}}{P_a}\right)\left(\frac{P_a}{P_{ssl}}\right) = \left(\frac{\bar{P}_{T_2}}{P_a}\right)\delta_a$$ *eq 4.26*

$$\theta_{T_2} = \frac{T_{T_2}}{T_{ssl}} = \frac{\left(T_{T_2} - T_a\right)}{T_{ssl}} + \frac{T_a}{T_{ssl}} = \frac{\left(T_{T_2} - T_a\right)}{T_{ssl}} + \theta_a$$ *eq 4.26a*

$$V_c = V_o + \Delta V_{ic} + \Delta V_{pos}$$ *eq 4.27*

$$ESHP = (K_Q)(Q)(N_R)$$ *eq 4.28*

$$\left(N_{1_{corr}}\right)_{max} = \frac{N_{1_{max}}}{\sqrt{\theta_{T_2}}}$$ *eq 4.29*

$$\left(T_{5_{corr}}\right)_{max} = \frac{T_{5_{max}}}{\theta_{T_2}}$$ *eq 4.30*

$$\left(W_{f_{corr}}\right)_{max} = \frac{W_{f_{max}}}{\delta_{T_2}\sqrt{\theta_{T_2}}}$$ *eq 4.31*

$$ESHP_A = \left(\frac{ESHP}{\delta_{T_2}\sqrt{\theta_{T_2}}}\right)_{max} \left(\delta_{T_2}\sqrt{\theta_{T_2}}\right)$$ *eq 4.32*

$$\frac{P_{T_2}}{P_a} = f(V_c)$$ *eq 4.33*

$$\left(T_{T_2} - T_a\right) = f(V_c)$$ *eq 4.34*

$$\frac{P_{T_2}}{P_a} = A_0 + A_1 V_c + A_2 V_c^2 + A_3 V_c^3$$ *eq 4.35*

$$\left(T_{T_2} - T_a\right) = B_0 + B_1 V_c + B_2 V_c^2 + B_3 V_c^3$$ *eq 4.36*

$$\frac{P_a}{P_{ssl}} = \left[1 - \left(\frac{\lambda_{ssl} H_P}{T_{ssl}}\right)\right]^{\frac{g_{ssl}}{g_c \lambda_{ssl} R}}$$ *eq 4.37*

$$T_a = \frac{OAT + 273.15}{1 + 0.2K_T\left[\dfrac{\left(\dfrac{V_c^2}{\delta}\right)}{a_{ssl}^2}\right]}$$ *eq 4.38*

$$P_{T_2} = \left(\frac{P_{T_2}}{P_a}\right)\left(P_a\right)$$ *eq 4.39*

$$T_{T_2} = (T_{T_2} - T_a) + T_a$$ *eq 4.40*

$$ESHP_{corr} = C_0 + C_1(N_{1_{corr}}) + C_2(N_{1_{corr}})^2 + C_3(N_{1_{corr}})^3$$ *eq 4.41*

$$W_{f_{corr}} = D_0 + D_1(N_{1_{corr}}) + D_2(N_{1_{corr}})^2 + D_3(N_{1_{corr}})^3$$ *eq 4.42*

$$T_{5_{corr}} = E_0 + E_1(N_{1_{corr}}) + E_2(N_{1_{corr}})^2 + E_3(N_{1_{corr}})^3$$ *eq 4.43*

CHAPTER 5

$$dR = dL + dD_0$$ *eq 5.1*

$$dR = dT + dF$$ *eq 5.2*

$$dT = dL\cos\alpha_i - dD_0\sin\alpha_i$$ *eq 5.3*

$$dF = dL\sin\alpha_i + dD_0\cos\alpha_i$$ *eq 5.4*

$$dL = C_\ell q_e\, dS$$ *eq 5.5*

EQUATIONS

$$dD_0 = C_{d_0}\, q_e\, dS \qquad \textit{eq 5.6}$$

$$dT = \frac{\rho(\Omega r)^2}{2\cos^2 \alpha_i}\, c\, dr\, (C_\ell \cos\alpha_i - C_{d_0} \sin\alpha_i) \qquad \textit{eq 5.7}$$

$$dT = \frac{1}{2}\, \rho\, (\Omega r)^2\, c\, dr\, C_\ell \qquad \textit{eq 5.8}$$

$$T_{perblade} = \int dT = \int \frac{1}{2}\, \rho\, (\Omega r)^2\, c\, dr\, C_\ell \qquad \textit{eq 5.9}$$

$$T_b = \frac{1}{6}\, \sigma_R\, \overline{C}_\ell\, \rho\, A_D\, (\Omega R)^2 \qquad \textit{eq 5.10}$$

$$dF = \frac{\rho\, (\Omega r)^2}{2\cos^2 \alpha_i}\, c\, dr\, (C_\ell \sin\alpha_i + C_{d_0} \cos\alpha_i) \qquad \textit{eq 5.11}$$

$$dF = \frac{1}{2}\rho\, (\Omega r)^2\, c\, dr\, (C_{d_i} + C_{d_0}) \qquad \textit{eq 5.12}$$

$$dP = dF\, (r)\, \Omega \qquad \textit{eq 5.13}$$

$$P_b = \frac{1}{8}\, \sigma_R\, (\overline{C}_{d_0} + \overline{C}_{d_i})\, \rho\, A_D\, (\Omega R)^3 \qquad \textit{eq 5.14}$$

$$P_0 = \frac{1}{8}\, \sigma_R\, \overline{C}_{d_0}\, \rho\, A_D\, (\Omega R)^3 \qquad \textit{eq 5.15}$$

$$P_{T_u} = P_a = C_1 \qquad \textit{eq 5.16}$$

$$P_{T_d} = P_a + q_w = C_2 \qquad \textit{eq 5.17}$$

$$\Delta P = P_{T_d} - P_{T_u} = q_w$$ *eq 5.18*

$$\frac{T}{A_D} = \Delta P = \frac{1}{2}\rho\, v_w^2$$ *eq 5.19*

$$F = \frac{dm}{dt}\ \Delta V$$ *eq 5.20*

$$(\Delta P)\, A_D = \rho\, A_D\, (v_w)\, v_i$$ *eq 5.21*

$$T = \frac{dm}{dt}\ \Delta D = 2\,\rho\, A_D\, v_i^2$$ *eq 5.22*

$$v_i = \sqrt{\frac{T}{2\,\rho\, A_D}} = \sqrt{\frac{W}{2\,\rho\, A_D}}$$ *eq 5.23*

$$P_i = Tv_i = \sqrt{\frac{T^3}{2\,\rho\, A_D}} = \sqrt{\frac{W^3}{2\,\rho\, A_D}}$$ *eq 5.24*

$$\eta_m = \frac{RSHP}{ESHP}$$ *eq 5.25*

$$P_{MR} = P_i + P_0$$ *eq 5.26*

$$P_{MR} = \sqrt{\frac{T^3}{2\,\rho\, A_D}} + \frac{1}{8}\,\sigma_R\, \bar{C}_{d_0}\,\rho\, A_D(\Omega R)^3$$ *eq 5.27*

$$P_{TOTAL} = P_i + P_0 + P_{TR} + P_{Acc} + P_{loss}$$ *eq 5.28*

$$C_T = \frac{T}{\rho_a\, A_D\, (\Omega R)^2} = f\left(\frac{T}{\sigma}, N_R\right)$$ *eq 5.29*

$$C_P = \frac{P}{\rho_a A_D (\Omega R)^3} = f\left(\frac{P}{\sigma}, N_R\right)$$ *eq 5.30*

$$C_T = \frac{1}{6} \sigma_R \bar{C}_1$$ *eq 5.31*

$$C_P = \frac{1}{8} \sigma_R \left(\bar{C}_{d_0} + \bar{C}_{d_i}\right)$$ *eq 5.32*

$$C_T = f\left(W, H_D, \frac{1}{N_R}\right)$$ *eq 5.33*

$$C_P = f\left(RSHP, H_D, \frac{1}{N_R}\right)$$ *eq 5.34*

$$C_P = \sqrt{\frac{C_T^3}{2}} = \frac{1}{8} \sigma_R \bar{C}_{d_0}$$ *eq 5.35*

$$M' = \frac{T v_i}{P} = \frac{\text{Minimum Power Required}}{\text{Actual Power Required}}$$ *eq 5.36*

$$M' = \frac{T^{\frac{3}{2}}}{P} \sqrt{\frac{1}{2 \rho A_D}}$$ *eq 5.37*

$$M' = \frac{\dfrac{C_T^{\frac{3}{2}}}{\sqrt{2}}}{\dfrac{C_T^{\frac{3}{2}}}{\sqrt{2}} + \dfrac{1}{8} \sigma_R \bar{C}_{d_0}}$$ *eq 5.38*

$$M' = \frac{\left(\dfrac{C_T^{\frac{3}{2}}}{\sqrt{2}}\right)}{C_P}$$ *eq 5.39*

$$C_P = C_1 C_T^{\frac{3}{2}} + C_2$$ *eq 5.40*

$$W_S = C_T \rho_S A_D (\Omega R_S)^2 \qquad \textit{eq 5.41}$$

$$RSHP_S = C_P \rho_S A_D (\Omega R_S)^3 \qquad \textit{eq 5.42}$$

$$W_S = W_T \frac{\rho_S}{\rho_T}\left(\frac{\Omega R_S}{\Omega R_T}\right)^2 \qquad \textit{eq 5.43}$$

$$ESHP_S = ESHP_T \frac{\rho_S}{\rho_T}\left(\frac{\Omega R_S}{\Omega R_T}\right)^3 \qquad \textit{eq 5.44}$$

$$W_{ref} = \frac{W_T}{\sigma_T}\left(\frac{N_{R_S}}{N_{R_T}}\right)^2 \qquad \textit{eq 5.45}$$

$$ESHP_{ref} = \frac{ESHP_T}{\sigma_T}\left(\frac{N_{R_S}}{N_{R_T}}\right)^3 \qquad \textit{eq 5.46}$$

$$H_{P_c} = H_{P_o} + \Delta H_{P_{ic}} + \Delta H_{pos} \qquad \textit{eq 5.47}$$

$$T_a = T_o + \Delta T_{ic} \qquad \textit{eq 5.48}$$

$$\sigma_T = \frac{\delta}{\theta} = \frac{\delta}{\left(\frac{T_a}{T_{ssl}}\right)} \qquad \textit{eq 5.49}$$

$$ESHP_T = K_Q (Q) (N_{R_T}) \qquad \textit{eq 5.50}$$

$$GW = ESGW - FU \qquad \textit{eq 5.51}$$

$$GW = ESGW - FU + \text{Cable Tension} \qquad \textit{eq 5.52}$$

$$\Omega R = K_{GR}\left(N_{R_T}\right)\left(\frac{2\pi}{60}\right)(R)$$ *eq 5.53*

$$ESHP_{ref} = f(W_{ref})$$ *eq 5.54*

$$ESHP_{ref} = A_0 + A_1\, W_{ref}^{\frac{3}{2}}$$ *eq 5.55*

$$\delta = \frac{P_a}{P_{ssl}} = \left(1 - \frac{\lambda_{ssl}\, H_P}{T_{ssl}}\right)^{\frac{g_{ssl}}{g_c\, \lambda_{ssl}\, R}}$$ *eq 5.56*

$$\theta = \frac{T_a}{T_{ssl}} = \frac{OAT + 273.15}{288.15}$$ *eq 5.57*

$$\sigma = \frac{\rho}{\rho_{ssl}} = \frac{\delta}{\theta}$$ *eq 5.58*

$$ESHP = \frac{ESHP_{ref}\ (\sigma)}{\left(\frac{N_{R_S}}{N_{R_T}}\right)^3}$$ *eq 5.59*

$$W_{ref} = \frac{W_S}{\sigma}$$ *eq 5.60*

$$ESHP = ESHP_{ref}(\sigma)$$ *eq 5.61*

CHAPTER 6

$$\Delta RSHP = RSHP_A - RSHP_{req}$$ *eq 6.1*

$$P_{\Delta PE} = W\left(\frac{dh}{dt}\right)$$ *eq 6.2*

$$V' = \frac{\Delta RSHP(33{,}000)}{W}$$ *eq 6.3*

$$V' = \frac{\Delta ESHP\left(\eta_m\right)(33{,}000)}{W}$$ *eq 6.4*

$$F = \frac{dm}{dt}\Delta V$$ *eq 6.5*

$$T = \rho\ A_D\ (V_V + v_{i_v})\ (2v_{i_v})$$ *eq 6.6*

$$\frac{T}{2\ \rho\ A_D} = v_{i_v}\ (V_V + v_{i_v})$$ *eq 6.7*

$$v_i^2 = v_{i_v}^2 + v_{i_v} V_V$$ *eq 6.8*

$$v_{i_v} = \sqrt{\left(\frac{V_V}{2}\right)^2 + v_i^2} - \frac{V_V}{2}$$ *eq 6.9*

$$P_{A_h} = P_{A_v}$$ *eq 6.10*

$$P_{p_h} + P_{0_h} + P_{i_h} + \Delta P_h = P_{p_v} + P_{0_v} + P_{i_v} + \Delta P_v$$ *eq 6.11*

$$W\left(v_i + V'\right) = W\left(v_{i_v} + V_v\right)$$ *eq 6.12*

$$\frac{V_v}{V'} = 1 + \frac{1}{\frac{V'}{v_i} + 1}$$ *eq 6.13*

$$v_i = \sqrt{\frac{W}{2\rho A_D}}$$ *eq 6.14*

$$H_{P_c} = H_{P_o} + \Delta H_{P_{ic}} + \Delta H_{pos}$$ *eq 6.15*

$$T_a = T_o + \Delta T_{ic}$$ *eq 6.16*

$$\sigma_T = \frac{\delta}{\theta} = \frac{\delta}{\left(\frac{T_a}{T_{ssl}}\right)}$$ *eq 6.17*

$$ESHP_h = K_Q\,(Q_h)\,(N_{R_h})$$ *eq 6.18*

$$ESHP_{climb} = K_Q(Q_{climb})\,(N_{R_{climb}})$$ *eq 6.19*

$$\Delta ESHP = ESHP_{climb} - ESHP_h$$ *eq 6.20*

$$GW = ESGW - FU$$ *eq 6.21*

$$V' = \frac{(33{,}000)(\Delta ESHP)}{GW}$$ *eq 6.22*

$$K_{H_P} = \frac{T_a + 273.15}{T_S + 273.15}$$ *eq 6.23*

$$V_v = K_{Hp}\left(\frac{dh}{dt}\right)$$ *eq 6.24*

$$v_i = 60\sqrt{\frac{GW}{2\,\rho\,A_D}}$$ *eq 6.25*

$$V' = \frac{\left(ESHP_A - ESHP_h\right)(33{,}000)}{W}$$ *eq 6.26*

$$V_V = \left(\frac{V_V}{V'}\right) V'$$ *eq 6.27*

$$ESHP_{ref} = A_0 + A_1\, W_{ref}^{\frac{3}{2}}$$ *eq 6.28*

$$\theta = \frac{T_a}{T_{ssl}} = \frac{OAT + 273.15}{288.15}$$ *eq 6.29*

$$\delta = \left[1 - \left(\frac{\lambda_{ssl}\ H_P}{T_{ssl}}\right)\right]^{\frac{g_{ssl}}{g_c\ \lambda_{ssl}\ R}}$$ *eq 6.30*

$$\theta = \frac{\delta}{\theta}$$ *eq 6.31*

$$V' = \frac{\left(ESHP_{climb} - ESHP_h\right)(33{,}000)}{W_T}$$ *eq 6.32*

$$\frac{T_T}{T_S} = \frac{OAT + 273.15}{T_{ssl} - \lambda_{ssl}\ H_{P_h}}$$ *eq 6.33*

$$V_V = 60\left(\frac{T_T}{T_S}\right)\left(\frac{\Delta H_P}{\Delta t}\right)$$ *eq 6.34*

$$v_i = 60\sqrt{\frac{W_T}{2\rho_{ssl}\ \sigma\ A_D}}$$ *eq 6.35*

$$\theta = 1 - \left(\frac{\lambda_{ssl}}{T_{ssl}}\right) H_P$$ *eq 6.36*

$$\theta = \frac{312.61 - \lambda_{ssl}\ H_P}{T_{ssl}}$$ *eq 6.37*

$$W_{ref} = \frac{W_T}{\sigma_T}\left(\frac{N_{R_S}}{N_{R_T}}\right)^2$$ *eq 6.38*

$$ESHP_h = ESHP_{ref}\sigma_S\left(\frac{N_R}{N_{R_S}}\right)^3$$ *eq 6.39*

$$ESHP_{A_2} = ESHP_{A_1}\left(\frac{100 - \text{margin}}{100}\right)$$ *eq 6.40*

CHAPTER 7

$$T = \frac{dm}{dt}\Delta V$$ *eq 7.1*

$$T_h = 2\,\rho\,A_D\,v_i^2$$ *eq 7.2*

$$T_v = \rho\,A_D\,(2\,v_{i_v})\,(v_{i_v} + V_V)$$ *eq 7.3*

$$T_f = \rho\,A_D\,V_R\,(2\,v_{i_f})$$ *eq 7.4*

$$V_R^2 = V_f^2 + v_{i_f}^2 + \left[(2V_f)\left(v_{i_f}\right)(\sin a)\right]$$ *eq 7.5*

$$V_R = V_f$$ *eq 7.6*

$$P_{i_f} = \frac{T^2}{2\,\rho\,A_D\,V_R} = \frac{W^2}{2\,\rho\,A_D\,V_f}$$ *eq 7.7*

$$dP_0 = dD_0\,v_e = \tfrac{1}{2}C_{d_0}\,\rho\left(\Omega r + V_f\,\sin\,\psi\right)^3\,c\,dr$$ *eq 7.8*

$$P_0 = \frac{b}{2\pi}\iint\tfrac{1}{2}\,C_{d_0}\,\rho\left(\Omega r + V_f\,\sin\,\psi\right)^3\,c\,dr\,d\psi$$ *eq 7.9*

$$P_0 = \frac{1}{8}\ \sigma_R\ \overline{C}_{d_0}\ \rho\ A_D\ (\Omega R)^3\ \left(1 + 3\mu^2\right)$$ *eq 7.10*

$$P_{0_{H2}} = \frac{1}{8}\ \sigma_R\ \overline{C}_{d_0}\ \rho\ A_D\ (\Omega R)^3\ \left(1 + 1.65\mu^2\right)$$ *eq 7.11*

$$P_0 = \frac{1}{8}\ \sigma_R\ \overline{C}_{d_0}\ \rho\ A_D\ (\Omega R)^3\ \left(1 + 4.65\mu^2\right)$$ *eq 7.12*

$$D_p = \frac{1}{2}\rho\ C_{D_p}\ A_f\ V^2$$ *eq 7.13*

$$P_p = \frac{1}{2}\rho\ C_{D_p}\ A_f\ V_f^3 = \frac{1}{2}\rho\ f\ V_f^3$$ *eq 7.14*

$$P_p = \frac{1}{2}\rho\ f\ (\Omega R)^3\ \mu^3$$ *eq 7.15*

$$T_{TR} = \frac{Q_{MR}}{\ell_{TR}}$$ *eq 7.16*

$$P_{TR} = \left(\frac{T_{TR}^2}{2\ \rho\ A_D\ V_f}\right)_{TR} + \left[\frac{1}{8}\sigma_R\ \overline{C}_{d_0}\ \rho\ A_D\ (\Omega R)^3 \left(1 + 4.65\mu^2\right)\right]_{TR}$$ *eq 7.17*

$$P_{TOTAL} = P_i + P_0 + P_p + P_m$$ *eq 7.18*

$$P_{MR} = \left(\frac{W^2}{2\ \rho\ A_D\ V_f}\right)_{MR} + \left[\frac{1}{8}\sigma_R\ \overline{C}_{d_0}\ \rho\ A_D\ (\Omega R)^3 \left(1 + 4.65\mu^2\right)\right]_{MR} + \left(\frac{1}{2}\ \rho\ V_f^3\ f\right)$$

eq 7.19

$$C_P = \frac{C_T^2}{2\mu} + \left[\frac{1}{8}\sigma_R\ \overline{C}_{d_0}\ \left(1 + 4.65\mu^2\right)\right] + \left(\frac{1}{2}\ C_{D_p}\ \mu^3\right)$$ *eq 7.20*

$$C_T = \frac{\left(\dfrac{W}{\sigma_T}\right)}{\rho_{ssl}\ A_D\ (\Omega R)^2}$$ *eq 7.21*

$$r = -\mu\ R\ \sin\psi$$ *eq 7.22*

$$P_0 = \frac{1}{8}\sigma_R\ \bar{C}_{d_0}\ \rho A_D\ (\Omega R)^3 \left(1 + 4.65\mu^2 - \frac{3}{8}\mu^4\right)$$ *eq 7.23*

$$RSHP_{ref} = \frac{C_P\ \rho_{ssl}\ A_D \left(\Omega R_S\right)^3}{550} = \left(\frac{RSHP_T}{\sigma_T}\right)\left(\frac{N_{R_S}}{N_{R_T}}\right)^3$$ *eq 7.24*

$$W_{ref} = C_T\ \rho_{ssl}\ A_D \left(\Omega R_S\right)^2 = \frac{W_T}{\sigma_T}\left(\frac{N_{R_S}}{N_{R_T}}\right)^2$$ *eq 7.25*

$$V_{T_{ref}} = \mu\left(\Omega R_T\right) = V_T\left(\frac{N_{R_S}}{N_{R_T}}\right)$$ *eq 7.26*

$$ESHP_{ref} = \frac{ESHP_T}{\sigma_T}\left(\frac{N_{R_S}}{N_{R_T}}\right)^3$$ *eq 7.27*

$$P = D_{eff}\ V_f$$ *eq 7.28*

$$H_{P_c} = H_{P_o} + \Delta H_{P_{ic}} + \Delta H_{pos}$$ *eq 7.29*

$$T_a = T_o + \Delta T_{ic}$$ *eq 7.30*

$$T_a(^\circ K) = T_a(^\circ C) + 273.15$$ *eq 7.31*

$$\sigma_T = \frac{\delta}{\theta} = \frac{\delta}{\left(\frac{T_a}{T_{ssl}}\right)}$$ *eq 7.32*

$$ESHP_T = K_Q\,(Q)\,(N_{R_T}) \qquad \textit{eq 7.33}$$

$$W_T = ESGW - FU \qquad \textit{eq 7.34}$$

$$W_{ref} = \frac{W_T}{\sigma_T}\left(\frac{N_{R_S}}{N_{R_T}}\right)^2 \qquad \textit{eq 7.35}$$

$$\Omega R = K_{GR}\left(N_{R_T}\right)\left(\frac{2\pi}{60}\right)(R) \qquad \textit{eq 7.36}$$

$$C_P = \frac{P}{\rho_a\,(A_D)\,(\Omega R)^3} \qquad \textit{eq 7.37}$$

$$C_T = \frac{T}{\rho_a\,(A_D)\,(\Omega R)^2} \qquad \textit{eq 7.38}$$

$$V_c = V_o + \Delta V_{ic} + \Delta V_{pos} \qquad \textit{eq 7.39}$$

$$V_T = \frac{V_c}{\sqrt{\sigma}} \qquad \textit{eq 7.40}$$

$$V_{T_{ref}} = V_T\left(\frac{N_{R_S}}{N_{R_T}}\right) \qquad \textit{eq 7.41}$$

$$\mu = 1.69\left(\frac{V_T}{\Omega R}\right) \qquad \textit{eq 7.42}$$

$$a = 38.967\sqrt{T_a\,(°K)} \qquad \textit{eq 7.43}$$

$$V_{TIP} = \left(\frac{\Omega R}{1.69}\right) + V_T \qquad \textit{eq 7.44}$$

$$M_{TIP} = \frac{V_{TIP}}{a}$$ *eq 7.45*

$$ESHP_{ref} = f\left(W_{ref}\ ,\ V_{T_{ref}} \right)$$ *eq 7.46*

$$ESHP_{ref} = E_0 + E_1\ V_{T_{ref}} + E_2\ V^2_{T_{ref}} + E_3\ V^3_{T_{ref}} + E_4\ V^4_{T_{ref}}$$ *eq 7.47*

$$\delta = \frac{P_a}{P_{ssl}} = \left[1 - \left(\frac{\lambda_{ssl}\ H_P}{T_{ssl}} \right) \right]^{\frac{g_{ssl}}{g_c\ \lambda_{ssl}\ R}}$$ *eq 7.48*

$$\theta = \frac{T_a}{T_{ssl}} = \frac{OAT + 273.15}{288.15}$$ *eq 7.49*

$$\sigma = \frac{\rho}{\rho_{ssl}} = \frac{\delta}{\theta}$$ *eq 7.50*

$$V_T = \frac{V_c}{\sqrt{\sigma_T}}$$ *eq 7.51*

$$M_{TIP} = 1.68781 V_T + \left(\frac{\Omega R_T}{\sqrt{\gamma\ g_c\ R\ T_{ssl}\ \theta}} \right)$$ *eq 7.52*

$$ESHP_{ref} = A_0 + A_1\ W_{ref}^{\frac{3}{2}}$$ *eq 7.53*

$$W_{f_{corr}} = B_0 + B_1\ ESHP_{corr} + B_2\ ESHP^2_{corr} + B_3\ ESHP^3_{corr}$$ *eq 7.54*

$$\frac{P_{T_2}}{P_a} = C_0 + C_1\ V_c + C_2\ V_c^2 + C_3\ V_c^3$$ *eq 7.55*

$$T_{T_2} - T_a = D_0 + D_1\ V_c + D_2\ V_c^2 + D_3\ V_c^3$$ *eq 7.56*

$$SR = \frac{V_T}{1.05\ W_f}$$ *eq 7.57*

$$W = \frac{W_{ref}\ (\sigma)}{\left(\frac{N_{R_S}}{N_R}\right)^2}$$ *eq 7.58*

$$V_T = \frac{V_{T_{ref}}}{\left(\frac{N_{R_S}}{N_R}\right)}$$ *eq 7.59*

$$ESHP = \frac{ESHP_{ref}\ (\sigma)}{\left(\frac{N_{R_S}}{N_R}\right)}$$ *eq 7.60*

$$V_c = V_T \sqrt{\sigma}$$ *eq 7.61*

$$ESHP_{corr} = \frac{ESHP}{\delta_{T_2}\sqrt{\theta_{T_2}}}$$ *eq 7.62*

$$W_f = W_{f_{corr}} \left(\delta_{T_2}\right)\sqrt{\theta_{T_2}}$$ *eq 7.63*

$$\text{Endurance} = \frac{\text{Fuel available}}{1.05\ W_f}$$ *eq 7.64*

CHAPTER 8

$$R/C = \frac{(ESHP_A - ESHP_{req})\,(33{,}000)}{W}$$ *eq 8.1*

$$\frac{Power_C - Power_A}{Power_B - Power_A} = \frac{(R/C)_C}{(R/C)_B}$$ *eq 8.2*

$$W = \rho\, A_D\, V_R\, (2v_{i_v})$$ *eq 8.3*

$$V_R = \sqrt{V_f^2 + (V_V + v_{i_v})^2}$$ *eq 8.4*

$$v_i = \sqrt{\frac{W}{2\,\rho\, A_D}}$$ *eq 8.5*

$$v_i^2 = v_{i_v}\sqrt{V_f^2 + (V_V + v_{i_v})^2}$$ *eq 8.6*

$$\frac{v_i}{v_{i_v}} = \sqrt{\left(\frac{V_f}{v_i}\right)^2 + \left(\frac{V_V}{v_i} + \frac{v_{i_v}}{v_i}\right)^2}$$ *eq 8.7*

$$P_{climb} \approx \left(P_i + P_o + P_p\right)_{level\ flight} + \frac{WV'}{33{,}000}$$ *eq 8.8*

$$V' = \frac{\left(P_{climb} - P_{level\ flight}\right)}{W}\,(33{,}000)$$ *eq 8.9*

$$\tan\varphi = \frac{V_V}{\Omega r}$$ *eq 8.10*

$$\varphi = \tan^{-1}\left(\frac{V_V}{\Omega r}\right)$$ *eq 8.11*

$$\tan \varphi' = \frac{D}{L} = \frac{C_D}{C_L}$$ *eq 8.12*

$$\varphi' = \tan^{-1}\left(\frac{C_D}{C_L}\right)$$ *eq 8.13*

$$\alpha = \theta + \varphi$$ *eq 8.14*

$$\varphi' = f(\alpha)$$ *eq 8.15*

$$P = \frac{d(PE)}{dt}$$ *eq 8.16*

$$P = W\left(\frac{dh}{dt}\right)$$ *eq 8.17*

$$P = W V_V$$ *eq 8.18*

$$K_{H_P} = \frac{T_a + 273.15}{T_S + 273.15}$$ *eq 8.19*

$$H_{P_c} = H_{P_o} + \Delta H_{P_{ic}} + \Delta H_{pos}$$ *eq 8.20*

$$T_a = T_o + \Delta T_{ic}$$ *eq 8.21*

$$\sigma_T = \frac{\delta}{\theta} = \frac{\delta}{\left(\frac{T_a}{T_{ssl}}\right)}$$ *eq 8.22*

$$ESHP_{lf} = K_Q\ Q_{lf}\ N_{R_{lf}}$$ *eq 8.23*

$$ESHP_{climb} = K_Q \, Q_{climb} \, N_{R_{climb}}$$ *eq 8.24*

$$ESHP_{descent} = K_Q \, Q_{descent} \, N_{R_{descent}}$$ *eq 8.25*

$$\Delta ESHP = ESHP_{climb} - ESHP_{lf}$$ *eq 8.26*

$$\Delta ESHP = ESHP_{descent} - ESHP_{lf}$$ *eq 8.27*

$$GW = ESGW - FU$$ *eq 8.28*

$$V' = \frac{33{,}000 \, (\Delta ESHP)}{GW}$$ *eq 8.29*

$$V_V = \left(\frac{dh}{dt}\right) K_{H_P}$$ *eq 8.30*

$$v_i = 60 \sqrt{\frac{GW}{2 \, \rho \, A_D}}$$ *eq 8.31*

$$V_c = V_o + \Delta V_{ic} + \Delta V_{pos}$$ *eq 8.32*

$$V_T = \frac{V_c}{\sqrt{\sigma_T}}$$ *eq 8.33*

$$\omega = \frac{\theta}{\Delta t}$$ *eq 8.34*

$$R/D = \frac{\Delta H_{P_c}}{\Delta t} \qquad \textit{eq 8.35}$$

$$V' = \frac{(ESHP_A - ESHP_{req})\,(33{,}000)}{W} \qquad \textit{eq 8.36}$$

$$V_v = \frac{\left(\frac{V_v}{V'}\right)}{V'} \qquad \textit{eq 8.37}$$

$$ESHP_{ref} = E_0 + E_1\, V_{T_{ref}} + E_2\, V^2_{T_{ref}} + E_3\, V^3_{T_{ref}} + E_4\, V^4_{T_{ref}} \qquad \textit{eq 8.38}$$

$$ESHP_{ref} = \frac{ESHP_T}{\sigma_T}\left(\frac{N_{R_S}}{N_{R_T}}\right)^3 \qquad \textit{eq 8.39}$$

$$W_{ref} = \frac{W_T}{\sigma_T}\left(\frac{N_{R_S}}{N_{R_T}}\right)^2 \qquad \textit{eq 8.40}$$

$$ESHP_T = K_Q\,(Q)\,(N_{R_T}) \qquad \textit{eq 8.41}$$

$$ESHP_{ref} = A_0 + A_1\, W_{ref}^{\frac{3}{2}} \qquad \textit{eq 8.42}$$

$$\theta = \frac{T_a}{T_{ssl}} = \frac{OAT + 273.15}{288.15} \qquad \textit{eq 8.43}$$

$$\delta = \left[1 - \left(\frac{\lambda_{ssl}\, H_P}{T_{ssl}}\right)\right]^{\frac{g_{ssl}}{g_c\, \lambda_{ssl}\, R}} \qquad \textit{eq 8.44}$$

$$\sigma = \frac{\delta}{\theta}$$ *eq 8.45*

$$V' = \frac{\left(ESHP_{clim\ b} - ESHP_{lf}\right)(33{,}000)}{W_T}$$ *eq 8.46*

$$V' = \frac{\left(ESHP_{descent} - ESHP_{lf}\right)(33{,}000)}{W_T}$$ *eq 8.47*

$$\frac{T_T}{T_S} = \frac{OAT + 273.15}{T_{ssl} - \lambda_{ssl} H_p}$$ *eq 8.48*

$$V_V = 60 \left(\frac{T_T}{T_S}\right)\left(\frac{\Delta H_p}{\Delta t}\right)$$ *eq 8.49*

$$V_i = 60 \sqrt{\frac{W_T}{2\ \rho_{ssl}\ \sigma\ A_D}}$$ *eq 8.50*

$$\theta = 1 - \left(\frac{\lambda_{ssl}}{T_{ssl}}\right) H_P$$ *eq 8.51*

$$\theta = \frac{312.61 - \lambda_{ssl} H_p}{T_{ssl}}$$ *eq 8.52*

APPENDIX VI

STANDARD ATMOSPHERE

APPENDIX VI

STANDARD ATMOSPHERE

1. The following constants and equations are provided for reference on atmospheric calculations:

δ = Ambient pressure ratio, P_a / P_{ssl}
θ = Ambient temperature ratio, T_a / T_{ssl}
σ = Ambient density ratio, ρ_a / ρ_{ssl}
$\sigma = \delta / \theta$ (non-dimensional equation of state)

Where:

P_{ssl} - Standard day ambient pressure at sea level, 2116.22 lb/ft^2
T_{ssl} - Standard day temperature at sea level, 288.15 ˚K (15 ˚C), 518.67 ˚R (59 ˚F)
ρ_{ssl} - Standard day density at sea level, 0.023769 $slugs/ft^3$

In addition:

a_{ssl} - Standard day speed of sound at sea level, 661.483 knots, 1116.45 ft/sec

2. The following expressions and constants pertain to the 1962 US and ICAO Standard Atmospheres. (note: The 1962 US Standard Atmosphere and ICAO Standard Atmosphere are identical up to 65,617 ft.)

A. The standard atmosphere is defined in terms of geopotential altitude. The relationship between geopotential height, h, and geometric height, z, is given by:

$$h = \frac{R_e z}{R_e + z}$$

Where:

R_e = Radius of the Earth, 20,925,780 ft

B. The variation of standard day temperature from sea level to the temperature from sea level to the tropopause (36,089 ft) is given by:

$$\theta = 1 - \frac{a_{lapse}}{T_{ssl}} h$$

Where:

a_{lapse} = Standard day temperature lapse rate, 0.0019812 ˚K (or ˚C) or 0.0035661 ˚R (or ˚F) per foot.

Hence, on a standard day:

$$\theta = 1 - 6.875585 \times 10^{-6} h$$

C. The pressure ratio on a standard day is defined by:

$$\delta = \left[1 - \frac{a_{lapse}}{T_{ssl}} h\right]^{\frac{g_{ssl}}{g_c} \frac{1}{a_{lapse} R}}$$

Where:

g_{ssl} - Standard sea level gravitational acceleration, 32.1741 ft /sec^2

g_c - 32.1741 lb mass/slug

R - Engineering gas constant for standard day air 53.35205 $\frac{ft - lb\ f}{lbm\ °R}$, or 96.03368 $\frac{ft - lb\ f}{lbm\ °K}$

Hence, on a standard day:

$$\delta = \left[1 - 6.875585 \times 10^{6} h\right]$$

D. The density ratio on a standard day is defined by:

$$\sigma = \delta / \theta$$

or

$$\sigma = \left[1 - \frac{a_{lapse}}{T_{ssl}} h\right]^{\frac{g_{ssl}}{g_c} \frac{1}{a_{lapse} R} - 1}$$

Which is equivalent to:

$$\sigma = \left[1 - 6.8755856 \times 10^{-6} h\right]^{4.255863}$$

E. The speed of sound in a gas is defined by:

$$a = (\gamma g_c R T_a)^{1/2} \text{ (ft/sec)}$$

Where:

$$\gamma = \text{ratio of specific heats (Air: } \gamma = 1.4)$$

Hence, for air:

$$a = 49.022 T_a^{1/2} \text{ (ft/sec)}$$
$$\text{for } T_a \text{ measured in } {}^\circ R$$

or:

$$a = 65.770 T_a^{1/2} \text{ (ft/sec)}$$
$$\text{for } T_a \text{ measured in } {}^\circ K$$

3. The following tabulation presents the atmospheric properties of the 1962 US and ICAO Standard Atmospheres. The 1962 US Standard Atmosphere are identified up to 65.617 ft.

H Geopotential Altitude	θ Temperature Ratio	T Standard Temperature	δ Pressure Ratio	P Standard Pressure	σ Density Ratio	ρ Standard Density	a Speed of Sound
ft		deg C		in Hg		slug/ft^3	kn
-2000	1.01375	18.96	1.07441	32.15	1.05984	0.0025191	666.0
-1980	1.01361	18.92	1.07363	32.12	1.05921	0.0025176	666.0
-1960	1.01348	18.88	1.07291	32.10	1.05864	0.0025163	665.9
-1940	1.01334	18.84	1.07213	32.08	1.05802	0.0025148	665.9
-1920	1.01320	18.80	1.07135	32.05	1.05739	0.0025133	665.8
-1900	1.01306	18.76	1.07058	32.03	1.05678	0.0025119	665.8
-1880	1.01293	18.73	1.06985	32.01	1.05619	0.0025105	665.7
-1860	1.01279	18.69	1.06908	31.99	1.05558	0.0025090	665.7
-1840	1.01265	18.65	1.06830	31.96	1.05495	0.0025075	665.7
-1820	1.01251	18.60	1.06753	31.94	1.05434	0.0025061	665.6
-1800	1.01238	18.57	1.06680	31.92	1.05375	0.0025047	665.6
-1780	1.01224	18.53	1.06603	31.90	1.05314	0.0025032	665.5
-1760	1.01210	18.49	1.06526	31.87	1.05252	0.0025017	665.5
-1740	1.01196	18.45	1.06448	31.85	1.05190	0.0025003	665.4
-1720	1.01183	18.41	1.06376	31.83	1.05132	0.0024989	665.4
-1700	1.01169	18.37	1.06299	31.80	1.05071	0.0024974	665.3
-1680	1.01155	18.33	1.06222	31.78	1.05009	0.0024960	665.3
-1660	1.01141	18.29	1.06144	31.76	1.04947	0.0024945	665.2
-1640	1.01128	18.25	1.06073	31.74	1.04890	0.0024931	665.2
-1620	1.01114	18.21	1.05996	31.71	1.04828	0.0024917	665.2
-1600	1.01100	18.17	1.05918	31.69	1.04766	0.0024902	665.1
-1580	1.01086	18.13	1.05841	31.67	1.04704	0.0024887	665.1
-1560	1.01073	18.09	1.05770	31.65	1.04647	0.0024874	665.0
-1540	1.01059	18.05	1.05693	31.62	1.04585	0.0024859	665.0
-1520	1.01045	18.01	1.05616	31.60	1.04524	0.0024844	664.9
-1500	1.01031	17.97	1.05539	31.58	1.04462	0.0024830	664.9
-1480	1.01018	17.93	1.05468	31.56	1.04405	0.0024816	664.8
-1460	1.01004	17.89	1.05391	31.53	1.04343	0.0024801	664.8
-1440	1.00990	17.85	1.05314	31.51	1.04282	0.0024787	664.7
-1420	1.00976	17.81	1.05237	31.49	1.04220	0.0024772	664.7
-1400	1.00963	17.77	1.05166	31.47	1.04163	0.0024759	664.7
-1380	1.00949	17.73	1.05090	31.44	1.04102	0.0024744	664.6
-1360	1.00935	17.69	1.05013	31.42	1.04040	0.0024729	664.6
-1340	1.00921	17.65	1.04936	31.40	1.03978	0.0024715	664.5
-1320	1.00908	17.62	1.04865	31.38	1.03921	0.0024701	664.5

H Geopotential Altitude	θ Temperature Ratio	T Standard Temperature	δ Pressure Ratio	P Standard Pressure	σ Density Ratio	ρ Standard Density	a Speed of Sound
ft		deg C		in Hg		slug/ft^3	kn
-1300	1.00894	17.58	1.04789	31.35	1.03860	0.0024686	664.4
-1280	1.00880	17.54	1.04713	31.33	1.03800	0.0024672	664.4
-1260	1.00866	17.50	1.04636	31.31	1.03738	0.0024657	664.3
-1240	1.00853	17.46	1.04565	31.29	1.03681	0.0024644	664.3
-1220	1.00839	17.42	1.04489	31.26	1.03620	0.0024629	664.3
-1200	1.00825	17.38	1.04413	31.24	1.03559	0.0024615	664.2
-1180	1.00811	17.34	1.04337	31.22	1.03498	0.0024600	664.2
-1160	1.00798	17.30	1.04266	31.20	1.03441	0.0024587	664.1
-1140	1.00784	17.26	1.04190	31.17	1.03380	0.0024572	664.1
-1120	1.00770	17.22	1.04114	31.15	1.03318	0.0024558	664.0
-1100	1.00756	17.18	1.04038	31.13	1.03257	0.0024543	664.0
-1080	1.00743	17.14	1.03967	31.11	1.03200	0.0024530	663.9
-1060	1.00729	17.10	1.03891	31.08	1.03139	0.0024515	663.9
-1040	1.00715	17.06	1.03816	31.06	1.03079	0.0024501	663.8
-1020	1.00701	17.02	1.03740	31.04	1.03018	0.0024486	663.8
-1000	1.00688	16.98	1.03669	31.02	1.02961	0.0024473	663.8
-980	1.00674	16.94	1.03594	31.00	1.02900	0.0024458	663.7
-960	1.00660	16.90	1.03518	30.97	1.02839	0.0024444	663.7
-940	1.00646	16.86	1.03442	30.95	1.02778	0.0024429	663.6
-920	1.00633	16.82	1.03372	30.93	1.02722	0.0024416	663.6
-900	1.00619	16.78	1.03297	30.91	1.02662	0.0024402	663.5
-880	1.00605	16.74	1.03221	30.88	1.02600	0.0024387	663.5
-860	1.00591	16.70	1.03146	30.86	1.02540	0.0024373	663.4
-840	1.00578	16.67	1.03075	30.84	1.02483	0.0024359	663.4
-820	1.00564	16.63	1.03000	30.82	1.02422	0.0024345	663.3
-800	1.00550	16.58	1.02925	30.80	1.02362	0.0024330	663.3
-780	1.00536	16.54	1.02849	30.77	1.02301	0.0024316	663.3
-760	1.00523	16.51	1.02780	30.75	1.02245	0.0024303	663.2
-740	1.00509	16.47	1.02704	30.73	1.02184	0.0024288	663.2
-720	1.00495	16.43	1.02629	30.71	1.02123	0.0024274	663.1
-700	1.00481	16.39	1.02554	30.68	1.02063	0.0024259	663.1
-680	1.00468	16.35	1.02484	30.66	1.02007	0.0024246	663.0
-660	1.00454	16.31	1.02409	30.64	1.01946	0.0024232	663.0
-640	1.00440	16.27	1.02334	30.62	1.01886	0.0024217	662.9
-620	1.00426	16.23	1.02259	30.60	1.01825	0.0024203	662.9

H Geopotential Altitude ft	θ Temperature Ratio	T Standard Temperature deg C	δ Pressure Ratio	P Standard Pressure in Hg	σ Density Ratio	ρ Standard Density slug/ft^3	a Speed of Sound kn
-600	1.00413	16.19	1.02190	30.58	1.01770	0.0024190	662.8
-580	1.00399	16.15	1.02115	30.55	1.01709	0.0024175	662.8
-560	1.00385	16.11	1.02040	30.53	1.01649	0.0024161	662.8
-540	1.00371	16.07	1.01965	30.51	1.01588	0.0024146	662.7
-520	1.00358	16.03	1.01896	30.49	1.01533	0.0024133	662.7
-500	1.00344	15.99	1.01821	30.46	1.01472	0.0024119	662.6
-480	1.00330	15.95	1.01747	30.44	1.01412	0.0024105	662.6
-460	1.00316	15.91	1.01672	30.42	1.01352	0.0024090	662.5
-440	1.00303	15.87	1.01603	30.40	1.01296	0.0024077	662.5
-420	1.00289	15.83	1.01528	30.38	1.01235	0.0024063	662.4
-400	1.00275	15.79	1.01454	30.36	1.01176	0.0024049	662.4
-380	1.00261	15.75	1.01379	30.33	1.01115	0.0024034	662.3
-360	1.00248	15.71	1.01310	30.31	1.01059	0.0024021	662.3
-340	1.00234	15.67	1.01236	30.29	1.01000	0.0024007	662.3
-320	1.00220	15.63	1.01162	30.27	1.00940	0.0023992	662.2
-300	1.00206	15.59	1.01087	30.25	1.00879	0.0023978	662.2
-280	1.00193	15.56	1.01019	30.22	1.00824	0.0023965	662.1
-260	1.00179	15.52	1.00944	30.20	1.00764	0.0023951	662.1
-240	1.00165	15.48	1.00870	30.18	1.00704	0.0023936	662.0
-220	1.00151	15.44	1.00796	30.16	1.00644	0.0023922	662.0
-200	1.00138	15.40	1.00727	30.14	1.00588	0.0023909	661.9
-180	1.00124	15.36	1.00653	30.12	1.00528	0.0023895	661.9
-160	1.00110	15.32	1.00579	30.09	1.00468	0.0023880	661.8
-140	1.00096	15.28	1.00506	30.07	1.00410	0.0023866	661.8
-120	1.00083	15.24	1.00437	30.05	1.00354	0.0023853	661.8
-100	1.00069	15.20	1.00363	30.03	1.00294	0.0023839	661.7
-80	1.00055	15.16	1.00289	30.01	1.00234	0.0023825	661.7
-60	1.00041	15.12	1.00216	29.98	1.00175	0.0023811	661.6
-40	1.00028	15.08	1.00147	29.96	1.00119	0.0023797	661.6
-20	1.00014	15.04	1.00074	29.94	1.00060	0.0023783	661.5
0	1.00000	15.00	1.00000	29.92	1.00000	0.0023769	661.5
20	0.99986	14.96	0.99926	29.90	0.99940	0.0023755	661.4
40	0.99972	14.92	0.99853	29.88	0.99881	0.0023741	661.4
60	0.99959	14.88	0.99785	29.86	0.99826	0.0023728	661.3
80	0.99945	14.84	0.99711	29.83	0.99766	0.0023713	661.3

H Geopotential Altitude	θ Temperature Ratio	T Standard Temperature	δ Pressure Ratio	P Standard Pressure	σ Density Ratio	ρ Standard Density	a Speed of Sound
ft		deg C		in Hg		slug/ft^3	kn
100	0.99931	14.80	0.99638	29.81	0.99707	0.0023699	661.3
120	0.99917	14.76	0.99565	29.79	0.99648	0.0023685	661.2
140	0.99904	14.72	0.99496	29.77	0.99592	0.0023672	661.2
160	0.99890	14.68	0.99423	29.75	0.99532	0.0023658	661.1
180	0.99876	14.64	0.99350	29.73	0.99473	0.0023644	661.1
200	0.99862	14.60	0.99277	29.70	0.99414	0.0023630	661.0
220	0.99849	14.56	0.99209	29.68	0.99359	0.0023617	661.0
240	0.99835	14.52	0.99136	29.66	0.99300	0.0023603	660.9
260	0.99821	14.48	0.99063	29.64	0.99241	0.0023589	660.9
280	0.99807	14.44	0.98990	29.62	0.99181	0.0023574	660.8
300	0.99794	14.41	0.98922	29.60	0.99126	0.0023561	660.8
320	0.99780	14.37	0.98849	29.58	0.99067	0.0023547	660.8
340	0.99766	14.33	0.98776	29.55	0.99008	0.0023533	660.7
360	0.99752	14.29	0.98703	29.53	0.98948	0.0023519	660.7
380	0.99739	14.25	0.98636	29.51	0.98894	0.0023506	660.6
400	0.99725	14.21	0.98563	29.49	0.98835	0.0023492	660.6
420	0.99711	14.17	0.98490	29.47	0.98775	0.0023478	660.5
440	0.99697	14.13	0.98418	29.45	0.98717	0.0023464	660.5
460	0.99684	14.09	0.98350	29.43	0.98662	0.0023451	660.4
480	0.99670	14.05	0.98278	29.40	0.98603	0.0023437	660.4
500	0.99656	14.01	0.98205	29.38	0.98544	0.0023423	660.3
520	0.99642	13.97	0.98133	29.36	0.98486	0.0023409	660.3
540	0.99629	13.93	0.98065	29.34	0.98430	0.0023396	660.3
560	0.99615	13.89	0.97993	29.32	0.98372	0.0023382	660.2
580	0.99601	13.85	0.97921	29.30	0.98313	0.0023368	660.2
600	0.99587	13.81	0.97848	29.28	0.98254	0.0023354	660.1
620	0.99574	13.77	0.97781	29.26	0.98199	0.0023341	660.1
640	0.99560	13.73	0.97709	29.23	0.98141	0.0023327	660.0
660	0.99546	13.69	0.97637	29.21	0.98082	0.0023313	660.0
680	0.99532	13.65	0.97565	29.19	0.98024	0.0023299	659.9
700	0.99519	13.61	0.97498	29.17	0.97969	0.0023286	659.9
720	0.99505	13.57	0.97426	29.15	0.97911	0.0023272	659.8
740	0.99491	13.53	0.97354	29.13	0.97852	0.0023258	659.8
760	0.99477	13.49	0.97282	29.11	0.97793	0.0023244	659.8
780	0.99464	13.46	0.97215	29.09	0.97739	0.0023232	659.7

H Geopotential Altitude ft	θ Temperature Ratio	T Standard Temperature deg C	δ Pressure Ratio	P Standard Pressure in Hg	σ Density Ratio	ρ Standard Density slug/ft^3	a Speed of Sound kn
800	0.99450	13.42	0.97143	29.07	0.97680	0.0023218	659.7
820	0.99436	13.37	0.97071	29.04	0.97622	0.0023204	659.6
840	0.99422	13.33	0.96999	29.02	0.97563	0.0023190	659.6
860	0.99409	13.30	0.96933	29.00	0.97509	0.0023177	659.5
880	0.99395	13.26	0.96861	28.98	0.97451	0.0023163	659.5
900	0.99381	13.22	0.96789	28.96	0.97392	0.0023149	659.4
920	0.99367	13.18	0.96718	28.94	0.97334	0.0023135	659.4
940	0.99354	13.14	0.96651	28.92	0.97279	0.0023122	659.3
960	0.99340	13.10	0.96580	28.90	0.97222	0.0023109	659.3
980	0.99326	13.06	0.96508	28.88	0.97163	0.0023095	659.3
1000	0.99312	13.02	0.96437	28.85	0.97105	0.0023081	659.2
1020	0.99299	12.98	0.96370	28.83	0.97050	0.0023068	659.2
1040	0.99285	12.94	0.96299	28.81	0.96992	0.0023054	659.1
1060	0.99271	12.90	0.96227	28.79	0.96934	0.0023040	659.1
1080	0.99257	12.86	0.96156	28.77	0.96876	0.0023026	659.0
1100	0.99244	12.82	0.96090	28.75	0.96822	0.0023014	659.0
1120	0.99230	12.78	0.96019	28.73	0.96764	0.0023000	658.9
1140	0.99216	12.74	0.95948	28.71	0.96706	0.0022986	658.9
1160	0.99202	12.70	0.95876	28.69	0.96647	0.0022972	658.8
1180	0.99189	12.66	0.95810	28.67	0.96593	0.0022959	658.8
1200	0.99175	12.62	0.95739	28.65	0.96535	0.0022945	658.7
1220	0.99161	12.58	0.95668	28.62	0.96477	0.0022932	658.7
1240	0.99147	12.54	0.95597	28.60	0.96419	0.0022918	658.7
1260	0.99134	12.50	0.95532	28.58	0.96367	0.0022905	658.6
1280	0.99120	12.46	0.95461	28.56	0.96309	0.0022892	658.6
1300	0.99106	12.42	0.95390	28.54	0.96250	0.0022878	658.5
1320	0.99092	12.38	0.95319	28.52	0.96192	0.0022864	658.5
1340	0.99079	12.35	0.95253	28.50	0.96138	0.0022851	658.4
1360	0.99065	12.31	0.95183	28.48	0.96081	0.0022837	658.4
1380	0.99051	12.27	0.95112	28.46	0.96023	0.0022824	658.3
1400	0.99037	12.23	0.95041	28.44	0.95965	0.0022810	658.3
1420	0.99024	12.19	0.94976	28.42	0.95912	0.0022797	658.2
1440	0.99010	12.15	0.94905	28.40	0.95854	0.0022784	658.2
1460	0.98996	12.11	0.94835	28.37	0.95797	0.0022770	658.2
1480	0.98982	12.07	0.94764	28.35	0.95739	0.0022756	658.1

H Geopotential Altitude ft	θ Temperature Ratio	T Standard Temperature deg C	δ Pressure Ratio	P Standard Pressure in Hg	σ Density Ratio	ρ Standard Density slug/ft^3	a Speed of Sound kn
1500	0.98969	12.03	0.94699	28.33	0.95686	0.0022744	658.1
1520	0.98955	11.99	0.94628	28.31	0.95627	0.0022730	658.0
1540	0.98941	11.95	0.94558	28.29	0.95570	0.0022716	658.0
1560	0.98927	11.91	0.94488	28.27	0.95513	0.0022702	657.9
1580	0.98914	11.87	0.94422	28.25	0.95459	0.0022690	657.9
1600	0.98900	11.83	0.94352	28.23	0.95401	0.0022676	657.8
1620	0.98886	11.79	0.94282	28.21	0.95344	0.0022662	657.8
1640	0.98872	11.75	0.94212	28.19	0.95287	0.0022649	657.7
1660	0.98859	11.71	0.94147	28.17	0.95234	0.0022636	657.7
1680	0.98845	11.67	0.94077	28.15	0.95176	0.0022622	657.7
1700	0.98831	11.63	0.94007	28.13	0.95119	0.0022609	657.6
1720	0.98817	11.59	0.93937	28.11	0.95062	0.0022595	657.6
1740	0.98804	11.55	0.93872	28.09	0.95008	0.0022582	657.5
1760	0.98790	11.51	0.93802	28.07	0.94951	0.0022569	657.5
1780	0.98776	11.47	0.93732	28.04	0.94893	0.0022555	657.4
1800	0.98762	11.43	0.93662	28.02	0.94836	0.0022542	657.4
1820	0.98749	11.40	0.93598	28.00	0.94784	0.0022529	657.3
1840	0.98735	11.35	0.93528	27.98	0.94726	0.0022515	657.3
1860	0.98721	11.31	0.93458	27.96	0.94669	0.0022502	657.2
1880	0.98707	11.27	0.93389	27.94	0.94612	0.0022488	657.2
1900	0.98694	11.24	0.93324	27.92	0.94559	0.0022476	657.1
1920	0.98680	11.20	0.93254	27.90	0.94501	0.0022462	657.1
1940	0.98666	11.16	0.93185	27.88	0.94445	0.0022449	657.1
1960	0.98652	11.12	0.93115	27.86	0.94387	0.0022435	657.0
1980	0.98639	11.08	0.93051	27.84	0.94335	0.0022422	657.0
2000	0.98625	11.04	0.92982	27.82	0.94278	0.0022409	656.9
2020	0.98611	11.00	0.92912	27.80	0.94221	0.0022395	656.9
2040	0.98597	10.96	0.92843	27.78	0.94164	0.0022382	656.8
2060	0.98584	10.92	0.92779	27.76	0.94112	0.0022369	656.8
2080	0.98570	10.88	0.92709	27.74	0.94054	0.0022356	656.7
2100	0.98556	10.84	0.92640	27.72	0.93997	0.0022342	656.7
2120	0.98542	10.80	0.92571	27.70	0.93941	0.0022329	656.6
2140	0.98529	10.76	0.92507	27.68	0.93888	0.0022316	656.6
2160	0.98515	10.72	0.92438	27.66	0.93831	0.0022303	656.6
2180	0.98501	10.68	0.92369	27.64	0.93775	0.0022289	656.5

H Geopotential Altitude	θ Temperature Ratio	T Standard Temperature	δ Pressure Ratio	P Standard Pressure	σ Density Ratio	ρ Standard Density	a Speed of Sound
ft		deg C		in Hg		slug/ft^3	kn
2200	0.98487	10.64	0.92300	27.62	0.93718	0.0022276	656.5
2220	0.98474	10.60	0.92236	27.60	0.93665	0.0022263	656.4
2240	0.98460	10.56	0.92167	27.58	0.93609	0.0022250	656.4
2260	0.98446	10.52	0.92098	27.56	0.93552	0.0022236	656.3
2280	0.98432	10.48	0.92029	27.54	0.93495	0.0022223	656.3
2300	0.98419	10.44	0.91965	27.52	0.93442	0.0022210	656.2
2320	0.98405	10.40	0.91897	27.50	0.93387	0.0022197	656.2
2340	0.98391	10.36	0.91828	27.47	0.93330	0.0022184	656.1
2360	0.98377	10.32	0.91759	27.45	0.93273	0.0022170	656.1
2380	0.98364	10.29	0.91695	27.44	0.93220	0.0022157	656.0
2400	0.98350	10.25	0.91627	27.41	0.93164	0.0022144	656.0
2420	0.98336	10.21	0.91558	27.39	0.93107	0.0022131	656.0
2440	0.98322	10.16	0.91490	27.37	0.93051	0.0022117	655.9
2460	0.98309	10.13	0.91426	27.35	0.92999	0.0022105	655.9
2480	0.98295	10.09	0.91358	27.33	0.92943	0.0022092	655.8
2500	0.98281	10.05	0.91290	27.31	0.92887	0.0022078	655.8
2520	0.98267	10.01	0.91221	27.29	0.92830	0.0022065	655.7
2540	0.98254	9.97	0.91158	27.27	0.92778	0.0022052	655.7
2560	0.98240	9.93	0.91090	27.25	0.92722	0.0022039	655.6
2580	0.98226	9.89	0.91021	27.23	0.92665	0.0022026	655.6
2600	0.98212	9.85	0.90953	27.21	0.92609	0.0022012	655.5
2620	0.98199	9.81	0.90890	27.19	0.92557	0.0022000	655.5
2640	0.98185	9.77	0.90822	27.17	0.92501	0.0021987	655.5
2660	0.98171	9.73	0.90754	27.15	0.92445	0.0021973	655.4
2680	0.98157	9.69	0.90686	27.13	0.92389	0.0021960	655.4
2700	0.98144	9.65	0.90623	27.11	0.92337	0.0021948	655.3
2720	0.98130	9.61	0.90555	27.09	0.92281	0.0021934	655.3
2740	0.98116	9.57	0.90487	27.07	0.92225	0.0021921	655.2
2760	0.98102	9.53	0.90419	27.05	0.92168	0.0021907	655.2
2780	0.98089	9.49	0.90356	27.03	0.92116	0.0021895	655.1
2800	0.98075	9.45	0.90288	27.01	0.92060	0.0021882	655.1
2820	0.98061	9.41	0.90221	26.99	0.92005	0.0021869	655.0
2840	0.98047	9.37	0.90153	26.97	0.91949	0.0021855	655.0
2860	0.98034	9.33	0.90090	26.95	0.91897	0.0021843	654.9
2880	0.98020	9.29	0.90023	26.93	0.91841	0.0021830	654.9

H Geopotential Altitude	θ Temperature Ratio	T Standard Temperature	δ Pressure Ratio	P Standard Pressure	σ Density Ratio	ρ Standard Density	a Speed of Sound
ft		deg C		in Hg		slug/ft^3	kn
2900	0.98006	9.25	0.89955	26.91	0.91785	0.0021816	654.9
2920	0.97992	9.21	0.89887	26.89	0.91729	0.0021803	654.8
2940	0.97979	9.18	0.89825	26.88	0.91678	0.0021791	654.8
2960	0.97965	9.14	0.89757	26.86	0.91621	0.0021777	654.7
2980	0.97951	9.10	0.89690	26.84	0.91566	0.0021764	654.7
3000	0.97937	9.06	0.89623	26.82	0.91511	0.0021751	654.6
3020	0.97924	9.02	0.89560	26.80	0.91459	0.0021739	654.6
3040	0.97910	8.98	0.89493	26.78	0.91403	0.0021726	654.5
3060	0.97896	8.94	0.89426	26.76	0.91348	0.0021713	654.5
3080	0.97882	8.90	0.89358	26.74	0.91292	0.0021699	654.4
3100	0.97869	8.86	0.89296	26.72	0.91240	0.0021687	654.4
3120	0.97855	8.82	0.89229	26.70	0.91185	0.0021674	654.4
3140	0.97841	8.78	0.89162	26.68	0.91129	0.0021660	654.3
3160	0.97827	8.74	0.89095	26.66	0.91074	0.0021647	654.3
3180	0.97814	8.70	0.89033	26.64	0.91023	0.0021635	654.2
3200	0.97800	8.66	0.88966	26.62	0.90967	0.0021622	654.2
3220	0.97786	8.62	0.88899	26.60	0.90912	0.0021609	654.1
3240	0.97772	8.58	0.88832	26.58	0.90856	0.0021596	654.1
3260	0.97759	8.54	0.88770	26.56	0.90805	0.0021583	654.0
3280	0.97745	8.50	0.88703	26.54	0.90749	0.0021570	654.0
3300	0.97731	8.46	0.88636	26.52	0.90694	0.0021557	653.9
3320	0.97717	8.42	0.88570	26.50	0.90639	0.0021544	653.9
3340	0.97704	8.38	0.88508	26.48	0.90588	0.0021532	653.8
3360	0.97690	8.34	0.88441	26.46	0.90532	0.0021519	653.8
3380	0.97676	8.30	0.88374	26.44	0.90477	0.0021505	653.8
3400	0.97662	8.26	0.88308	26.42	0.90422	0.0021492	653.7
3420	0.97649	8.23	0.88246	26.40	0.90371	0.0021480	653.7
3440	0.97635	8.19	0.88180	26.38	0.90316	0.0021467	653.6
3460	0.97621	8.14	0.88113	26.36	0.90260	0.0021454	653.6
3480	0.97607	8.10	0.88047	26.34	0.90206	0.0021441	653.5
3500	0.97594	8.07	0.87985	26.33	0.90154	0.0021429	653.5
3520	0.97580	8.03	0.87919	26.31	0.90099	0.0021416	653.4
3540	0.97566	7.99	0.87853	26.29	0.90045	0.0021403	653.4
3560	0.97552	7.95	0.87786	26.27	0.89989	0.0021389	653.3
3580	0.97539	7.91	0.87725	26.25	0.89938	0.0021377	653.3

H Geopotential Altitude ft	θ Temperature Ratio	T Standard Temperature deg C	δ Pressure Ratio	P Standard Pressure in Hg	σ Density Ratio	ρ Standard Density slug/ft^3	a Speed of Sound kn
3600	0.97525	7.87	0.87659	26.23	0.89884	0.0021365	653.2
3620	0.97511	7.83	0.87593	26.21	0.89829	0.0021351	653.2
3640	0.97497	7.79	0.87526	26.19	0.89773	0.0021338	653.2
3660	0.97484	7.75	0.87465	26.17	0.89722	0.0021326	653.1
3680	0.97470	7.71	0.87399	26.15	0.89668	0.0021313	653.1
3700	0.97456	7.67	0.87333	26.13	0.89613	0.0021300	653.0
3720	0.97442	7.63	0.87267	26.11	0.89558	0.0021287	653.0
3740	0.97429	7.59	0.87206	26.09	0.89507	0.0021275	652.9
3760	0.97415	7.55	0.87140	26.07	0.89452	0.0021262	652.9
3780	0.97401	7.51	0.87074	26.05	0.89397	0.0021249	652.8
3800	0.97387	7.47	0.87009	26.03	0.89344	0.0021236	652.8
3820	0.97374	7.43	0.86948	26.01	0.89293	0.0021224	652.7
3840	0.97360	7.39	0.86882	26.00	0.89238	0.0021211	652.7
3860	0.97346	7.35	0.86816	25.98	0.89183	0.0021198	652.6
3880	0.97332	7.31	0.86751	25.96	0.89129	0.0021185	652.6
3900	0.97319	7.27	0.86690	25.94	0.89078	0.0021173	652.6
3920	0.97305	7.23	0.86624	25.92	0.89023	0.0021160	652.5
3940	0.97291	7.19	0.86559	25.90	0.88969	0.0021147	652.5
3960	0.97277	7.15	0.86493	25.88	0.88914	0.0021134	652.4
3980	0.97264	7.12	0.86433	25.86	0.88864	0.0021122	652.4
4000	0.97250	7.08	0.86367	25.84	0.88809	0.0021109	652.3
4020	0.97236	7.04	0.86302	25.82	0.88755	0.0021096	652.3
4040	0.97222	7.00	0.86237	25.80	0.88701	0.0021083	652.2
4060	0.97209	6.96	0.86176	25.78	0.88650	0.0021071	652.2
4080	0.97195	6.92	0.86111	25.76	0.88596	0.0021058	652.1
4100	0.97181	6.88	0.86046	25.74	0.88542	0.0021046	652.1
4120	0.97167	6.84	0.85981	25.73	0.88488	0.0021033	652.0
4140	0.97154	6.80	0.85920	25.71	0.88437	0.0021021	652.0
4160	0.97140	6.76	0.85855	25.69	0.88383	0.0021008	652.0
4180	0.97126	6.72	0.85790	25.67	0.88329	0.0020995	651.9
4200	0.97112	6.68	0.85725	25.65	0.88274	0.0020982	651.9
4220	0.97099	6.64	0.85665	25.63	0.88224	0.0020970	651.8
4240	0.97085	6.60	0.85600	25.61	0.88170	0.0020957	651.8
4260	0.97071	6.56	0.85535	25.59	0.88116	0.0020944	651.7
4280	0.97057	6.52	0.85470	25.57	0.88062	0.0020931	651.7

H Geopotential Altitude ft	θ Temperature Ratio	T Standard Temperature deg C	δ Pressure Ratio	P Standard Pressure in Hg	σ Density Ratio	ρ Standard Density slug/ft^3	a Speed of Sound kn
4300	0.97043	6.48	0.85405	25.55	0.88007	0.0020918	651.6
4320	0.97030	6.44	0.85345	25.54	0.87957	0.0020906	651.6
4340	0.97016	6.40	0.85281	25.52	0.87904	0.0020894	651.5
4360	0.97002	6.36	0.85216	25.50	0.87850	0.0020881	651.5
4380	0.96988	6.32	0.85151	25.48	0.87795	0.0020868	651.4
4400	0.96975	6.28	0.85091	25.46	0.87745	0.0020856	651.4
4420	0.96961	6.24	0.85027	25.44	0.87692	0.0020844	651.4
4440	0.96947	6.20	0.84962	25.42	0.87638	0.0020831	651.3
4460	0.96933	6.16	0.84898	25.40	0.87584	0.0020818	651.3
4480	0.96920	6.12	0.84838	25.38	0.87534	0.0020806	651.2
4500	0.96906	6.08	0.84774	25.36	0.87481	0.0020793	651.2
4520	0.96892	6.04	0.84709	25.34	0.87426	0.0020780	651.1
4540	0.96878	6.00	0.84645	25.33	0.87373	0.0020768	651.1
4560	0.96865	5.97	0.84585	25.31	0.87323	0.0020756	651.0
4580	0.96851	5.93	0.84521	25.29	0.87269	0.0020743	651.0
4600	0.96837	5.89	0.84457	25.27	0.87216	0.0020730	650.9
4620	0.96823	5.85	0.84393	25.25	0.87162	0.0020718	650.9
4640	0.96810	5.81	0.84333	25.23	0.87112	0.0020706	650.8
4660	0.96796	5.77	0.84269	25.21	0.87058	0.0020693	650.8
4680	0.96782	5.73	0.84205	25.19	0.87005	0.0020680	650.8
4700	0.96768	5.69	0.84141	25.17	0.86951	0.0020667	650.7
4720	0.96755	5.65	0.84082	25.16	0.86902	0.0020656	650.7
4740	0.96741	5.61	0.84018	25.14	0.86848	0.0020643	650.6
4760	0.96727	5.57	0.83954	25.12	0.86795	0.0020630	650.6
4780	0.96713	5.53	0.83890	25.10	0.86741	0.0020617	650.5
4800	0.96700	5.49	0.83831	25.08	0.86692	0.0020606	650.5
4820	0.96686	5.45	0.83767	25.06	0.86638	0.0020593	650.4
4840	0.96672	5.41	0.83703	25.04	0.86585	0.0020580	650.4
4860	0.96658	5.37	0.83640	25.03	0.86532	0.0020568	650.3
4880	0.96645	5.33	0.83580	25.01	0.86481	0.0020556	650.3
4900	0.96631	5.29	0.83517	24.99	0.86429	0.0020543	650.2
4920	0.96617	5.25	0.83453	24.97	0.86375	0.0020530	650.2
4940	0.96603	5.21	0.83390	24.95	0.86322	0.0020518	650.2
4960	0.96590	5.17	0.83331	24.93	0.86273	0.0020506	650.1
4980	0.96576	5.13	0.83267	24.91	0.86219	0.0020493	650.1

H Geopotential Altitude	θ Temperature Ratio	T Standard Temperature	δ Pressure Ratio	P Standard Pressure	σ Density Ratio	ρ Standard Density	a Speed of Sound
ft		deg C		in Hg		slug/ft^3	kn
5000	0.96562	5.09	0.83204	24.89	0.86166	0.0020481	650.0
5020	0.96548	5.05	0.83141	24.88	0.86114	0.0020468	650.0
5040	0.96535	5.02	0.83082	24.86	0.86064	0.0020457	649.9
5060	0.96521	4.98	0.83018	24.84	0.86010	0.0020444	649.9
5080	0.96507	4.93	0.82955	24.82	0.85957	0.0020431	649.8
5100	0.96493	4.89	0.82892	24.80	0.85905	0.0020419	649.8
5120	0.96480	4.86	0.82833	24.78	0.85855	0.0020407	649.7
5140	0.96466	4.82	0.82770	24.76	0.85802	0.0020394	649.7
5160	0.96452	4.78	0.82707	24.75	0.85749	0.0020382	649.6
5180	0.96438	4.74	0.82644	24.73	0.85697	0.0020369	649.6
5200	0.96425	4.70	0.82585	24.71	0.85647	0.0020357	649.6
5220	0.96411	4.66	0.82522	24.69	0.85594	0.0020345	649.5
5240	0.96397	4.62	0.82459	24.67	0.85541	0.0020332	649.5
5260	0.96383	4.58	0.82396	24.65	0.85488	0.0020320	649.4
5280	0.96370	4.54	0.82338	24.64	0.85439	0.0020308	649.4
5300	0.96356	4.50	0.82275	24.62	0.85386	0.0020295	649.3
5320	0.96342	4.46	0.82212	24.60	0.85333	0.0020283	649.3
5340	0.96328	4.42	0.82150	24.58	0.85282	0.0020271	649.2
5360	0.96315	4.38	0.82091	24.56	0.85232	0.0020259	649.2
5380	0.96301	4.34	0.82029	24.54	0.85180	0.0020246	649.1
5400	0.96287	4.30	0.81966	24.52	0.85127	0.0020234	649.1
5420	0.96273	4.26	0.81903	24.51	0.85074	0.0020221	649.0
5440	0.96260	4.22	0.81845	24.49	0.85025	0.0020210	649.0
5460	0.96246	4.18	0.81783	24.47	0.84973	0.0020197	648.9
5480	0.96232	4.14	0.81720	24.45	0.84920	0.0020185	648.9
5500	0.96218	4.10	0.81658	24.43	0.84868	0.0020172	648.9
5520	0.96205	4.06	0.81600	24.41	0.84819	0.0020161	648.8
5540	0.96191	4.02	0.81537	24.40	0.84766	0.0020148	648.8
5560	0.96177	3.98	0.81475	24.38	0.84714	0.0020136	648.7
5580	0.96163	3.94	0.81413	24.36	0.84661	0.0020123	648.7
5600	0.96150	3.91	0.81355	24.34	0.84613	0.0020112	648.6
5620	0.96136	3.87	0.81293	24.32	0.84560	0.0020099	648.6
5640	0.96122	3.83	0.81230	24.30	0.84507	0.0020086	648.5
5660	0.96108	3.79	0.81168	24.29	0.84455	0.0020074	648.5
5680	0.96095	3.75	0.81111	24.27	0.84407	0.0020063	648.4

H Geopotential Altitude	θ Temperature Ratio	T Standard Temperature	δ Pressure Ratio	P Standard Pressure	σ Density Ratio	ρ Standard Density	a Speed of Sound
ft		deg C		in Hg		slug/ft^3	kn
5700	0.96081	3.71	0.81049	24.25	0.84355	0.0020050	648.4
5720	0.96067	3.67	0.80986	24.23	0.84302	0.0020038	648.3
5740	0.96053	3.63	0.80924	24.21	0.84249	0.0020025	648.3
5760	0.96040	3.59	0.80867	24.20	0.84201	0.0020014	648.3
5780	0.96026	3.55	0.80805	24.18	0.84149	0.0020001	648.2
5800	0.96012	3.51	0.80743	24.16	0.84097	0.0019989	648.2
5820	0.95998	3.47	0.80681	24.14	0.84044	0.0019976	648.1
5840	0.95985	3.43	0.80624	24.12	0.83996	0.0019965	648.1
5860	0.95971	3.39	0.80562	24.10	0.83944	0.0019953	648.0
5880	0.95957	3.35	0.80500	24.09	0.83892	0.0019940	648.0
5900	0.95943	3.31	0.80439	24.07	0.83840	0.0019928	647.9
5920	0.95930	3.27	0.80381	24.05	0.83791	0.0019916	647.9
5940	0.95916	3.23	0.80320	24.03	0.83740	0.0019904	647.8
5960	0.95902	3.19	0.80258	24.01	0.83688	0.0019892	647.8
5980	0.95888	3.15	0.80197	23.99	0.83636	0.0019879	647.7
6000	0.95875	3.11	0.80139	23.98	0.83587	0.0019868	647.7
6020	0.95861	3.07	0.80078	23.96	0.83536	0.0019856	647.6
6040	0.95847	3.03	0.80016	23.94	0.83483	0.0019843	647.6
6060	0.95833	2.99	0.79955	23.92	0.83432	0.0019831	647.6
6080	0.95820	2.96	0.79898	23.91	0.83383	0.0019819	647.5
6100	0.95806	2.91	0.79837	23.89	0.83332	0.0019807	647.5
6120	0.95792	2.87	0.79775	23.87	0.83279	0.0019795	647.4
6140	0.95778	2.83	0.79714	23.85	0.83228	0.0019782	647.4
6160	0.95765	2.80	0.79657	23.83	0.83180	0.0019771	647.3
6180	0.95751	2.76	0.79596	23.82	0.83128	0.0019759	647.3
6200	0.95737	2.72	0.79535	23.80	0.83077	0.0019747	647.2
6220	0.95723	2.68	0.79474	23.78	0.83025	0.0019734	647.2
6240	0.95710	2.64	0.79417	23.76	0.82977	0.0019723	647.1
6260	0.95696	2.60	0.79356	23.74	0.82925	0.0019710	647.1
6280	0.95682	2.56	0.79295	23.73	0.82873	0.0019698	647.0
6300	0.95668	2.52	0.79234	23.71	0.82822	0.0019686	647.0
6320	0.95655	2.48	0.79178	23.69	0.82775	0.0019675	647.0
6340	0.95641	2.44	0.79117	23.67	0.82723	0.0019662	646.9
6360	0.95627	2.40	0.79056	23.65	0.82671	0.0019650	646.9
6380	0.95613	2.36	0.78995	23.64	0.82620	0.0019638	646.8

H Geopotential Altitude ft	θ Temperature Ratio	T Standard Temperature deg C	δ Pressure Ratio	P Standard Pressure in Hg	σ Density Ratio	ρ Standard Density slug/ft^3	a Speed of Sound kn
6400	0.95600	2.32	0.78939	23.62	0.82572	0.0019627	646.8
6420	0.95586	2.28	0.78878	23.60	0.82520	0.0019614	646.7
6440	0.95572	2.24	0.78817	23.58	0.82469	0.0019602	646.7
6460	0.95558	2.20	0.78756	23.56	0.82417	0.0019590	646.6
6480	0.95545	2.16	0.78700	23.55	0.82370	0.0019579	646.6
6500	0.95531	2.12	0.78640	23.53	0.82319	0.0019566	646.5
6520	0.95517	2.08	0.78579	23.51	0.82267	0.0019554	646.5
6540	0.95503	2.04	0.78519	23.49	0.82216	0.0019542	646.4
6560	0.95490	2.00	0.78462	23.48	0.82168	0.0019531	646.4
6580	0.95476	1.96	0.78402	23.46	0.82117	0.0019518	646.3
6600	0.95462	1.92	0.78342	23.44	0.82066	0.0019506	646.3
6620	0.95448	1.88	0.78281	23.42	0.82014	0.0019494	646.3
6640	0.95435	1.85	0.78225	23.40	0.81967	0.0019483	646.2
6660	0.95421	1.81	0.78165	23.39	0.81916	0.0019471	646.2
6680	0.95407	1.77	0.78105	23.37	0.81865	0.0019458	646.1
6700	0.95393	1.72	0.78044	23.35	0.81813	0.0019446	646.1
6720	0.95380	1.69	0.77988	23.33	0.81766	0.0019435	646.0
6740	0.95366	1.65	0.77928	23.32	0.81715	0.0019423	646.0
6760	0.95352	1.61	0.77868	23.30	0.81664	0.0019411	645.9
6780	0.95338	1.57	0.77808	23.28	0.81613	0.0019399	645.9
6800	0.95325	1.53	0.77752	23.26	0.81565	0.0019387	645.8
6820	0.95311	1.49	0.77692	23.25	0.81514	0.0019375	645.8
6840	0.95297	1.45	0.77632	23.23	0.81463	0.0019363	645.7
6860	0.95283	1.41	0.77573	23.21	0.81413	0.0019351	645.7
6880	0.95270	1.37	0.77517	23.19	0.81366	0.0019340	645.6
6900	0.95256	1.33	0.77457	23.18	0.81315	0.0019328	645.6
6920	0.95242	1.29	0.77397	23.16	0.81264	0.0019316	645.6
6940	0.95228	1.25	0.77337	23.14	0.81212	0.0019303	645.5
6960	0.95215	1.21	0.77282	23.12	0.81166	0.0019292	645.5
6980	0.95201	1.17	0.77222	23.10	0.81115	0.0019280	645.4
7000	0.95187	1.13	0.77163	23.09	0.81065	0.0019268	645.4
7020	0.95173	1.09	0.77103	23.07	0.81014	0.0019256	645.3
7040	0.95160	1.05	0.77048	23.05	0.80967	0.0019245	645.3
7060	0.95146	1.01	0.76988	23.03	0.80916	0.0019233	645.2
7080	0.95132	0.97	0.76929	23.02	0.80866	0.0019221	645.2

H Geopotential Altitude	θ Temperature Ratio	T Standard Temperature	δ Pressure Ratio	P Standard Pressure	σ Density Ratio	ρ Standard Density	a Speed of Sound
ft		deg C		in Hg		slug/ft^3	kn
7100	0.95118	0.93	0.76869	23.00	0.80814	0.0019209	645.1
7120	0.95105	0.90	0.76814	22.98	0.80768	0.0019198	645.1
7140	0.95091	0.85	0.76754	22.96	0.80716	0.0019185	645.0
7160	0.95077	0.81	0.76695	22.95	0.80666	0.0019174	645.0
7180	0.95063	0.77	0.76636	22.93	0.80616	0.0019162	644.9
7200	0.95050	0.74	0.76581	22.91	0.80569	0.0019150	644.9
7220	0.95036	0.70	0.76521	22.90	0.80518	0.0019138	644.9
7240	0.95022	0.66	0.76462	22.88	0.80468	0.0019126	644.8
7260	0.95008	0.62	0.76403	22.86	0.80417	0.0019114	644.8
7280	0.94995	0.58	0.76348	22.84	0.80371	0.0019103	644.7
7300	0.94981	0.54	0.76289	22.83	0.80320	0.0019091	644.7
7320	0.94967	0.50	0.76230	22.81	0.80270	0.0019079	644.6
7340	0.94953	0.46	0.76171	22.79	0.80220	0.0019067	644.6
7360	0.94940	0.42	0.76116	22.77	0.80173	0.0019056	644.5
7380	0.94926	0.38	0.76057	22.76	0.80122	0.0019044	644.5
7400	0.94912	0.34	0.75998	22.74	0.80072	0.0019032	644.4
7420	0.94898	0.30	0.75939	22.72	0.80022	0.0019020	644.4
7440	0.94885	0.26	0.75885	22.70	0.79976	0.0019009	644.3
7460	0.94871	0.22	0.75826	22.69	0.79925	0.0018997	644.3
7480	0.94857	0.18	0.75767	22.67	0.79875	0.0018985	644.2
7500	0.94843	0.14	0.75708	22.65	0.79825	0.0018974	644.2
7520	0.94830	0.10	0.75654	22.64	0.79779	0.0018963	644.2
7540	0.94816	0.06	0.75595	22.62	0.79728	0.0018951	644.1
7560	0.94802	0.02	0.75536	22.60	0.79678	0.0018939	644.1
7580	0.94788	-0.02	0.75478	22.58	0.79628	0.0018927	644.0
7600	0.94775	-0.06	0.75423	22.57	0.79581	0.0018916	644.0
7620	0.94761	-0.10	0.75365	22.55	0.79532	0.0018904	643.9
7640	0.94747	-0.14	0.75306	22.53	0.79481	0.0018892	643.9
7660	0.94733	-0.18	0.75248	22.51	0.79432	0.0018880	643.8
7680	0.94720	-0.21	0.75194	22.50	0.79386	0.0018869	643.8
7700	0.94706	-0.25	0.75135	22.48	0.79335	0.0018857	643.7
7720	0.94692	-0.30	0.75077	22.46	0.79285	0.0018845	643.7
7740	0.94678	-0.34	0.75019	22.45	0.79236	0.0018834	643.6
7760	0.94665	-0.37	0.74964	22.43	0.79189	0.0018822	643.6
7780	0.94651	-0.41	0.74906	22.41	0.79139	0.0018811	643.5

H Geopotential Altitude	θ Temperature Ratio	T Standard Temperature	δ Pressure Ratio	P Standard Pressure	σ Density Ratio	ρ Standard Density	a Speed of Sound
ft		deg C		in Hg		slug/ft^3	kn
7800	0.94637	-0.45	0.74848	22.39	0.79090	0.0018799	643.5
7820	0.94623	-0.49	0.74790	22.38	0.79040	0.0018787	643.5
7840	0.94610	-0.53	0.74736	22.36	0.78994	0.0018776	643.4
7860	0.94596	-0.57	0.74678	22.34	0.78944	0.0018764	643.4
7880	0.94582	-0.61	0.74620	22.33	0.78895	0.0018753	643.3
7900	0.94568	-0.65	0.74562	22.31	0.78845	0.0018741	643.3
7920	0.94555	-0.69	0.74508	22.29	0.78799	0.0018730	643.2
7940	0.94541	-0.73	0.74450	22.28	0.78749	0.0018718	643.2
7960	0.94527	-0.77	0.74392	22.26	0.78699	0.0018706	643.1
7980	0.94513	-0.81	0.74334	22.24	0.78649	0.0018694	643.1
8000	0.94500	-0.85	0.74280	22.22	0.78603	0.0018683	643.0
8020	0.94486	-0.89	0.74222	22.21	0.78553	0.0018671	643.0
8040	0.94472	-0.93	0.74165	22.19	0.78505	0.0018660	642.9
8060	0.94458	-0.97	0.74107	22.17	0.78455	0.0018648	642.9
8080	0.94445	-1.01	0.74053	22.16	0.78409	0.0018637	642.8
8100	0.94431	-1.05	0.73996	22.14	0.78360	0.0018625	642.8
8120	0.94417	-1.09	0.73938	22.12	0.78310	0.0018614	642.8
8140	0.94403	-1.13	0.73880	22.10	0.78260	0.0018602	642.7
8160	0.94390	-1.17	0.73827	22.09	0.78215	0.0018591	642.7
8180	0.94376	-1.21	0.73769	22.07	0.78165	0.0018579	642.6
8200	0.94362	-1.25	0.73712	22.05	0.78116	0.0018567	642.6
8220	0.94348	-1.29	0.73654	22.04	0.78066	0.0018556	642.5
8240	0.94335	-1.32	0.73601	22.02	0.78021	0.0018545	642.5
8260	0.94321	-1.36	0.73544	22.00	0.77972	0.0018533	642.4
8280	0.94307	-1.40	0.73486	21.99	0.77922	0.0018521	642.4
8300	0.94293	-1.44	0.73429	21.97	0.77873	0.0018510	642.3
8320	0.94280	-1.48	0.73376	21.95	0.77828	0.0018499	642.3
8340	0.94266	-1.52	0.73319	21.94	0.77779	0.0018487	642.2
8360	0.94252	-1.56	0.73261	21.92	0.77729	0.0018475	642.2
8380	0.94238	-1.60	0.73204	21.90	0.77680	0.0018464	642.1
8400	0.94225	-1.64	0.73151	21.89	0.77634	0.0018453	642.1
8420	0.94211	-1.68	0.73094	21.87	0.77585	0.0018441	642.1
8440	0.94197	-1.72	0.73037	21.85	0.77536	0.0018430	642.0
8460	0.94183	-1.76	0.72980	21.84	0.77487	0.0018418	642.0
8480	0.94170	-1.80	0.72927	21.82	0.77442	0.0018407	641.9

H Geopotential Altitude ft	θ Temperature Ratio	T Standard Temperature deg C	δ Pressure Ratio	P Standard Pressure in Hg	σ Density Ratio	ρ Standard Density slug/ft^3	a Speed of Sound kn
8500	0.94156	-1.84	0.72870	21.80	0.77393	0.0018396	641.9
8520	0.94142	-1.88	0.72813	21.79	0.77344	0.0018384	641.8
8540	0.94128	-1.92	0.72756	21.77	0.77295	0.0018372	641.8
8560	0.94114	-1.96	0.72699	21.75	0.77246	0.0018361	641.7
8580	0.94101	-2.00	0.72647	21.74	0.77201	0.0018350	641.7
8600	0.94087	-2.04	0.72590	21.72	0.77152	0.0018338	641.6
8620	0.94073	-2.08	0.72533	21.70	0.77103	0.0018327	641.6
8640	0.94059	-2.12	0.72476	21.68	0.77054	0.0018315	641.5
8660	0.94046	-2.16	0.72424	21.67	0.77009	0.0018304	641.5
8680	0.94032	-2.20	0.72367	21.65	0.76960	0.0018293	641.4
8700	0.94018	-2.24	0.72310	21.64	0.76911	0.0018281	641.4
8720	0.94004	-2.28	0.72254	21.62	0.76863	0.0018270	641.3
8740	0.93991	-2.31	0.72201	21.60	0.76817	0.0018259	641.3
8760	0.93977	-2.36	0.72145	21.59	0.76769	0.0018247	641.3
8780	0.93963	-2.40	0.72088	21.57	0.76720	0.0018236	641.2
8800	0.93949	-2.44	0.72032	21.55	0.76671	0.0018224	641.2
8820	0.93936	-2.47	0.71980	21.54	0.76627	0.0018213	641.1
8840	0.93922	-2.51	0.71923	21.52	0.76577	0.0018202	641.1
8860	0.93908	-2.55	0.71867	21.50	0.76529	0.0018190	641.0
8880	0.93894	-2.59	0.71811	21.49	0.76481	0.0018179	641.0
8900	0.93881	-2.63	0.71758	21.47	0.76435	0.0018168	640.9
8920	0.93867	-2.67	0.71702	21.45	0.76387	0.0018156	640.9
8940	0.93853	-2.71	0.71646	21.44	0.76339	0.0018145	640.8
8960	0.93839	-2.75	0.71590	21.42	0.76290	0.0018133	640.8
8980	0.93826	-2.79	0.71538	21.40	0.76245	0.0018123	640.7
9000	0.93812	-2.83	0.71482	21.39	0.76197	0.0018111	640.7
9020	0.93798	-2.87	0.71425	21.37	0.76148	0.0018100	640.6
9040	0.93784	-2.91	0.71369	21.35	0.76099	0.0018088	640.6
9060	0.93771	-2.95	0.71318	21.34	0.76055	0.0018078	640.5
9080	0.93757	-2.99	0.71262	21.32	0.76007	0.0018066	640.5
9100	0.93743	-3.03	0.71206	21.30	0.75959	0.0018055	640.5
9120	0.93729	-3.07	0.71150	21.29	0.75910	0.0018043	640.4
9140	0.93716	-3.11	0.71098	21.27	0.75865	0.0018032	640.4
9160	0.93702	-3.15	0.71042	21.26	0.75817	0.0018021	640.3
9180	0.93688	-3.19	0.70986	21.24	0.75769	0.0018010	640.3

H Geopotential Altitude ft	θ Temperature Ratio	T Standard Temperature deg C	δ Pressure Ratio	P Standard Pressure in Hg	σ Density Ratio	ρ Standard Density slug/ft^3	a Speed of Sound kn
9200	0.93674	-3.23	0.70931	21.22	0.75721	0.0017998	640.2
9220	0.93661	-3.27	0.70879	21.21	0.75676	0.0017987	640.2
9240	0.93647	-3.31	0.70823	21.19	0.75628	0.0017976	640.1
9260	0.93633	-3.35	0.70768	21.17	0.75580	0.0017965	640.1
9280	0.93619	-3.39	0.70712	21.16	0.75532	0.0017953	640.0
9300	0.93606	-3.42	0.70660	21.14	0.75487	0.0017943	640.0
9320	0.93592	-3.46	0.70605	21.13	0.75439	0.0017931	639.9
9340	0.93578	-3.50	0.70549	21.11	0.75391	0.0017920	639.9
9360	0.93564	-3.55	0.70494	21.09	0.75343	0.0017908	639.8
9380	0.93551	-3.58	0.70442	21.08	0.75298	0.0017898	639.8
9400	0.93537	-3.62	0.70387	21.06	0.75250	0.0017886	639.8
9420	0.93523	-3.66	0.70332	21.04	0.75203	0.0017875	639.7
9440	0.93509	-3.70	0.70276	21.03	0.75154	0.0017863	639.7
9460	0.93496	-3.74	0.70225	21.01	0.75110	0.0017853	639.6
9480	0.93482	-3.78	0.70170	20.99	0.75063	0.0017842	639.6
9500	0.93468	-3.82	0.70115	20.98	0.75015	0.0017830	639.5
9520	0.93454	-3.86	0.70059	20.96	0.74966	0.0017819	639.5
9540	0.93441	-3.90	0.70008	20.95	0.74922	0.0017808	639.4
9560	0.93427	-3.94	0.69953	20.93	0.74875	0.0017797	639.4
9580	0.93413	-3.98	0.69898	20.91	0.74827	0.0017786	639.3
9600	0.93399	-4.02	0.69843	20.90	0.74779	0.0017774	639.3
9620	0.93386	-4.06	0.69792	20.88	0.74735	0.0017764	639.2
9640	0.93372	-4.10	0.69737	20.87	0.74687	0.0017752	639.2
9660	0.93358	-4.14	0.69682	20.85	0.74640	0.0017741	639.1
9680	0.93344	-4.18	0.69627	20.83	0.74592	0.0017730	639.1
9700	0.93331	-4.22	0.69576	20.82	0.74548	0.0017719	639.0
9720	0.93317	-4.26	0.69521	20.80	0.74500	0.0017708	639.0
9740	0.93303	-4.30	0.69467	20.78	0.74453	0.0017697	638.9
9760	0.93289	-4.34	0.69412	20.77	0.74405	0.0017685	638.9
9780	0.93276	-4.38	0.69361	20.75	0.74361	0.0017675	638.9
9800	0.93262	-4.42	0.69306	20.74	0.74313	0.0017663	638.8
9820	0.93248	-4.46	0.69252	20.72	0.74266	0.0017652	638.8
9840	0.93234	-4.50	0.69197	20.70	0.74219	0.0017641	638.7
9860	0.93221	-4.53	0.69146	20.69	0.74174	0.0017630	638.7
9880	0.93207	-4.57	0.69092	20.67	0.74127	0.0017619	638.6

H Geopotential Altitude	θ Temperature Ratio	T Standard Temperature	δ Pressure Ratio	P Standard Pressure	σ Density Ratio	ρ Standard Density	a Speed of Sound
ft		deg C		in Hg		slug/ft^3	kn
9900	0.93193	-4.61	0.69037	20.66	0.74080	0.0017608	638.6
9920	0.93179	-4.65	0.68983	20.64	0.74033	0.0017597	638.5
9940	0.93166	-4.69	0.68932	20.62	0.73988	0.0017586	638.5
9960	0.93152	-4.73	0.68878	20.61	0.73942	0.0017575	638.4
9980	0.93138	-4.77	0.68823	20.59	0.73894	0.0017564	638.4
10000	0.93124	-4.81	0.68769	20.58	0.73847	0.0017553	638.3
10020	0.93111	-4.85	0.68718	20.56	0.73802	0.0017542	638.3
10040	0.93097	-4.89	0.68664	20.54	0.73755	0.0017531	638.2
10060	0.93083	-4.93	0.68610	20.53	0.73708	0.0017520	638.2
10080	0.93069	-4.97	0.68556	20.51	0.73661	0.0017508	638.1
10100	0.93056	-5.01	0.68505	20.50	0.73617	0.0017498	638.1
10120	0.93042	-5.05	0.68451	20.48	0.73570	0.0017487	638.1
10140	0.93028	-5.09	0.68397	20.46	0.73523	0.0017476	638.0
10160	0.93014	-5.13	0.68343	20.45	0.73476	0.0017465	638.0
10180	0.93001	-5.17	0.68293	20.43	0.73433	0.0017454	637.9
10200	0.92987	-5.21	0.68239	20.42	0.73386	0.0017443	637.9
10220	0.92973	-5.25	0.68185	20.40	0.73338	0.0017432	637.8
10240	0.92959	-5.29	0.68131	20.38	0.73291	0.0017421	637.8
10260	0.92946	-5.33	0.68081	20.37	0.73248	0.0017410	637.7
10280	0.92932	-5.37	0.68027	20.35	0.73201	0.0017399	637.7
10300	0.92918	-5.41	0.67973	20.34	0.73154	0.0017388	637.6
10320	0.92904	-5.45	0.67919	20.32	0.73107	0.0017377	637.6
10340	0.92891	-5.48	0.67869	20.31	0.73063	0.0017366	637.5
10360	0.92877	-5.52	0.67816	20.29	0.73017	0.0017355	637.5
10380	0.92863	-5.57	0.67762	20.27	0.72970	0.0017344	637.4
10400	0.92849	-5.61	0.67708	20.26	0.72923	0.0017333	637.4
10420	0.92836	-5.64	0.67658	20.24	0.72879	0.0017323	637.3
10440	0.92822	-5.68	0.67605	20.23	0.72833	0.0017312	637.3
10460	0.92808	-5.72	0.67551	20.21	0.72786	0.0017301	637.3
10480	0.92794	-5.76	0.67498	20.20	0.72740	0.0017290	637.2
10500	0.92781	-5.80	0.67448	20.18	0.72696	0.0017279	637.2
10520	0.92767	-5.84	0.67395	20.16	0.72650	0.0017268	637.1
10540	0.92753	-5.88	0.67341	20.15	0.72603	0.0017257	637.1
10560	0.92739	-5.92	0.67288	20.13	0.72556	0.0017246	637.0
10580	0.92726	-5.96	0.67238	20.12	0.72513	0.0017236	637.0

H Geopotential Altitude	θ Temperature Ratio	T Standard Temperature	δ Pressure Ratio	P Standard Pressure	σ Density Ratio	ρ Standard Density	a Speed of Sound
ft		deg C		in Hg		$slug/ft^3$	kn
10600	0.92712	-6.00	0.67185	20.10	0.72466	0.0017224	636.9
10620	0.92698	-6.04	0.67131	20.09	0.72419	0.0017213	636.9
10640	0.92684	-6.08	0.67078	20.07	0.72373	0.0017202	636.8
10660	0.92671	-6.12	0.67029	20.06	0.72330	0.0017192	636.8
10680	0.92657	-6.16	0.66976	20.04	0.72284	0.0017181	636.7
10700	0.92643	-6.20	0.66922	20.02	0.72236	0.0017170	636.7
10720	0.92629	-6.24	0.66869	20.01	0.72190	0.0017159	636.6
10740	0.92616	-6.28	0.66820	19.99	0.72147	0.0017149	636.6
10760	0.92602	-6.32	0.66767	19.98	0.72101	0.0017138	636.5
10780	0.92588	-6.36	0.66714	19.96	0.72055	0.0017127	636.5
10800	0.92574	-6.40	0.66661	19.94	0.72008	0.0017116	636.4
10820	0.92561	-6.44	0.66612	19.93	0.71966	0.0017106	636.4
10840	0.92547	-6.48	0.66559	19.91	0.71919	0.0017094	636.4
10860	0.92533	-6.52	0.66506	19.90	0.71873	0.0017083	636.3
10880	0.92519	-6.56	0.66453	19.88	0.71826	0.0017072	636.3
10900	0.92506	-6.59	0.66404	19.87	0.71783	0.0017062	636.2
10920	0.92492	-6.63	0.66351	19.85	0.71737	0.0017051	636.2
10940	0.92478	-6.67	0.66298	19.84	0.71691	0.0017040	636.1
10960	0.92464	-6.71	0.66246	19.82	0.71645	0.0017029	636.1
10980	0.92451	-6.75	0.66197	19.81	0.71602	0.0017019	636.0
11000	0.92437	-6.79	0.66144	19.79	0.71556	0.0017008	636.0
11020	0.92423	-6.83	0.66091	19.77	0.71509	0.0016997	635.9
11040	0.92409	-6.87	0.66039	19.76	0.71464	0.0016986	635.9
11060	0.92396	-6.91	0.65990	19.74	0.71421	0.0016976	635.8
11080	0.92382	-6.95	0.65937	19.73	0.71374	0.0016965	635.8
11100	0.92368	-6.99	0.65885	19.71	0.71329	0.0016954	635.7
11120	0.92354	-7.03	0.65832	19.70	0.71282	0.0016943	635.7
11140	0.92341	-7.07	0.65784	19.68	0.71240	0.0016933	635.6
11160	0.92327	-7.11	0.65731	19.67	0.71194	0.0016922	635.6
11180	0.92313	-7.15	0.65679	19.65	0.71148	0.0016911	635.6
11200	0.92299	-7.19	0.65627	19.64	0.71103	0.0016900	635.5
11220	0.92286	-7.23	0.65578	19.62	0.71060	0.0016890	635.5
11240	0.92272	-7.27	0.65526	19.61	0.71014	0.0016879	635.4
11260	0.92258	-7.31	0.65474	19.59	0.70968	0.0016868	635.4
11280	0.92244	-7.35	0.65421	19.57	0.70922	0.0016857	635.3

H Geopotential Altitude ft	θ Temperature Ratio	T Standard Temperature deg C	δ Pressure Ratio	P Standard Pressure in Hg	σ Density Ratio	ρ Standard Density slug/ft^3	a Speed of Sound kn
11300	0.92231	-7.39	0.65373	19.56	0.70880	0.0016847	635.3
11320	0.92217	-7.43	0.65321	19.54	0.70834	0.0016837	635.2
11340	0.92203	-7.47	0.65269	19.53	0.70788	0.0016826	635.2
11360	0.92189	-7.51	0.65217	19.51	0.70743	0.0016815	635.1
11380	0.92176	-7.54	0.65168	19.50	0.70700	0.0016805	635.1
11400	0.92162	-7.59	0.65116	19.48	0.70654	0.0016794	635.0
11420	0.92148	-7.63	0.65064	19.47	0.70608	0.0016783	635.0
11440	0.92134	-7.67	0.65012	19.45	0.70562	0.0016772	634.9
11460	0.92121	-7.70	0.64964	19.44	0.70520	0.0016762	634.9
11480	0.92107	-7.74	0.64912	19.42	0.70475	0.0016751	634.8
11500	0.92093	-7.78	0.64860	19.41	0.70429	0.0016740	634.8
11520	0.92079	-7.82	0.64809	19.39	0.70384	0.0016730	634.7
11540	0.92066	-7.86	0.64761	19.38	0.70342	0.0016720	634.7
11560	0.92052	-7.90	0.64709	19.36	0.70296	0.0016709	634.7
11580	0.92038	-7.94	0.64657	19.35	0.70250	0.0016698	634.6
11600	0.92024	-7.98	0.64605	19.33	0.70205	0.0016687	634.6
11620	0.92011	-8.02	0.64557	19.32	0.70162	0.0016677	634.5
11640	0.91997	-8.06	0.64506	19.30	0.70118	0.0016666	634.5
11660	0.91983	-8.10	0.64454	19.28	0.70072	0.0016655	634.4
11680	0.91969	-8.14	0.64403	19.27	0.70027	0.0016645	634.4
11700	0.91956	-8.18	0.64355	19.26	0.69985	0.0016635	634.3
11720	0.91942	-8.22	0.64303	19.24	0.69939	0.0016624	634.3
11740	0.91928	-8.26	0.64252	19.22	0.69894	0.0016613	634.2
11760	0.91914	-8.30	0.64201	19.21	0.69849	0.0016602	634.2
11780	0.91901	-8.34	0.64153	19.19	0.69807	0.0016592	634.1
11800	0.91887	-8.38	0.64102	19.18	0.69762	0.0016582	634.1
11820	0.91873	-8.42	0.64050	19.16	0.69716	0.0016571	634.0
11840	0.91859	-8.46	0.63999	19.15	0.69671	0.0016560	634.0
11860	0.91846	-8.50	0.63951	19.13	0.69629	0.0016550	633.9
11880	0.91832	-8.54	0.63900	19.12	0.69584	0.0016539	633.9
11900	0.91818	-8.58	0.63849	19.10	0.69539	0.0016529	633.8
11920	0.91804	-8.62	0.63798	19.09	0.69494	0.0016518	633.8
11940	0.91791	-8.65	0.63750	19.07	0.69451	0.0016508	633.8
11960	0.91777	-8.69	0.63699	19.06	0.69406	0.0016497	633.7
11980	0.91763	-8.73	0.63648	19.04	0.69361	0.0016486	633.7

H Geopotential Altitude	θ Temperature Ratio	T Standard Temperature	δ Pressure Ratio	P Standard Pressure	σ Density Ratio	ρ Standard Density	a Speed of Sound
ft		deg C		in Hg		slug/ft^3	kn
12000	0.91749	-8.78	0.63597	19.03	0.69316	0.0016476	633.6
12020	0.91736	-8.81	0.63550	19.01	0.69275	0.0016466	633.6
12040	0.91722	-8.85	0.63499	19.00	0.69230	0.0016455	633.5
12060	0.91708	-8.89	0.63448	18.98	0.69185	0.0016445	633.5
12080	0.91694	-8.93	0.63397	18.97	0.69140	0.0016434	633.4
12100	0.91681	-8.97	0.63350	18.95	0.69098	0.0016424	633.4
12120	0.91667	-9.01	0.63299	18.94	0.69053	0.0016413	633.3
12140	0.91653	-9.05	0.63248	18.92	0.69008	0.0016403	633.3
12160	0.91639	-9.09	0.63197	18.91	0.68963	0.0016392	633.2
12180	0.91626	-9.13	0.63150	18.89	0.68921	0.0016382	633.2
12200	0.91612	-9.17	0.63100	18.88	0.68877	0.0016371	633.1
12220	0.91598	-9.21	0.63049	18.86	0.68832	0.0016361	633.1
12240	0.91584	-9.25	0.62998	18.85	0.68787	0.0016350	633.0
12260	0.91571	-9.29	0.62951	18.83	0.68746	0.0016340	633.0
12280	0.91557	-9.33	0.62901	18.82	0.68701	0.0016330	632.9
12300	0.91543	-9.37	0.62850	18.80	0.68656	0.0016319	632.9
12320	0.91529	-9.41	0.62800	18.79	0.68612	0.0016308	632.8
12340	0.91516	-9.45	0.62753	18.78	0.68571	0.0016299	632.8
12360	0.91502	-9.49	0.62702	18.76	0.68525	0.0016288	632.8
12380	0.91488	-9.53	0.62652	18.75	0.68481	0.0016277	632.7
12400	0.91474	-9.57	0.62602	18.73	0.68437	0.0016267	632.7
12420	0.91461	-9.61	0.62555	18.72	0.68395	0.0016257	632.6
12440	0.91447	-9.65	0.62505	18.70	0.68351	0.0016246	632.6
12460	0.91433	-9.69	0.62454	18.69	0.68306	0.0016236	632.5
12480	0.91419	-9.73	0.62404	18.67	0.68262	0.0016225	632.5
12500	0.91406	-9.76	0.62357	18.66	0.68220	0.0016215	632.4
12520	0.91392	-9.80	0.62307	18.64	0.68176	0.0016205	632.4
12540	0.91378	-9.84	0.62257	18.63	0.68131	0.0016194	632.3
12560	0.91364	-9.88	0.62207	18.61	0.68087	0.0016184	632.3
12580	0.91351	-9.92	0.62161	18.60	0.68046	0.0016174	632.2
12600	0.91337	-9.96	0.62110	18.58	0.68001	0.0016163	632.2
12620	0.91323	-10.00	0.62060	18.57	0.67957	0.0016153	632.1
12640	0.91309	-10.04	0.62010	18.55	0.67912	0.0016142	632.1
12660	0.91296	-10.08	0.61964	18.54	0.67872	0.0016132	632.0
12680	0.91282	-10.12	0.61914	18.52	0.67827	0.0016122	632.0

H Geopotential Altitude	θ Temperature Ratio	T Standard Temperature	δ Pressure Ratio	P Standard Pressure	σ Density Ratio	ρ Standard Density	a Speed of Sound
ft		deg C		in Hg		slug/ft^3	kn
12700	0.91268	-10.16	0.61864	18.51	0.67783	0.0016111	631.9
12720	0.91254	-10.20	0.61814	18.49	0.67738	0.0016101	631.9
12740	0.91241	-10.24	0.61768	18.48	0.67698	0.0016091	631.8
12760	0.91227	-10.28	0.61718	18.47	0.67653	0.0016080	631.8
12780	0.91213	-10.32	0.61669	18.45	0.67610	0.0016070	631.8
12800	0.91199	-10.36	0.61619	18.44	0.67565	0.0016060	631.7
12820	0.91185	-10.40	0.61569	18.42	0.67521	0.0016049	631.7
12840	0.91172	-10.44	0.61523	18.41	0.67480	0.0016039	631.6
12860	0.91158	-10.48	0.61473	18.39	0.67436	0.0016029	631.6
12880	0.91144	-10.52	0.61424	18.38	0.67392	0.0016018	631.5
12900	0.91130	-10.56	0.61374	18.36	0.67348	0.0016008	631.5
12920	0.91117	-10.60	0.61328	18.35	0.67307	0.0015998	631.4
12940	0.91103	-10.64	0.61279	18.33	0.67263	0.0015988	631.4
12960	0.91089	-10.68	0.61229	18.32	0.67219	0.0015977	631.3
12980	0.91075	-10.72	0.61180	18.31	0.67175	0.0015967	631.3
13000	0.91062	-10.75	0.61134	18.29	0.67134	0.0015957	631.2
13020	0.91048	-10.80	0.61084	18.28	0.67090	0.0015947	631.2
13040	0.91034	-10.84	0.61035	18.26	0.67046	0.0015936	631.1
13060	0.91020	-10.88	0.60986	18.25	0.67003	0.0015926	631.1
13080	0.91007	-10.91	0.60940	18.23	0.66962	0.0015916	631.0
13100	0.90993	-10.95	0.60891	18.22	0.66918	0.0015906	631.0
13120	0.90979	-10.99	0.60842	18.20	0.66875	0.0015896	630.9
13140	0.90965	-11.03	0.60792	18.19	0.66830	0.0015885	630.9
13160	0.90952	-11.07	0.60747	18.18	0.66790	0.0015875	630.8
13180	0.90938	-11.11	0.60698	18.16	0.66747	0.0015865	630.8
13200	0.90924	-11.15	0.60648	18.15	0.66702	0.0015854	630.8
13220	0.90910	-11.19	0.60599	18.13	0.66658	0.0015844	630.7
13240	0.90897	-11.23	0.60554	18.12	0.66618	0.0015834	630.7
13260	0.90883	-11.27	0.60505	18.10	0.66575	0.0015824	630.6
13280	0.90869	-11.31	0.60456	18.09	0.66531	0.0015814	630.6
13300	0.90855	-11.35	0.60407	18.07	0.66487	0.0015803	630.5
13320	0.90842	-11.39	0.60362	18.06	0.66447	0.0015794	630.5
13340	0.90828	-11.43	0.60313	18.05	0.66404	0.0015784	630.4
13360	0.90814	-11.47	0.60264	18.03	0.66360	0.0015773	630.4
13380	0.90800	-11.51	0.60215	18.02	0.66316	0.0015763	630.3

H Geopotential Altitude	θ Temperature Ratio	T Standard Temperature	δ Pressure Ratio	P Standard Pressure	σ Density Ratio	ρ Standard Density	a Speed of Sound
ft		deg C		in Hg		slug/ft^3	kn
13400	0.90787	-11.55	0.60170	18.00	0.66276	0.0015753	630.3
13420	0.90773	-11.59	0.60121	17.99	0.66232	0.0015743	630.2
13440	0.90759	-11.63	0.60072	17.97	0.66188	0.0015732	630.2
13460	0.90745	-11.67	0.60024	17.96	0.66146	0.0015722	630.1
13480	0.90732	-11.71	0.59978	17.95	0.66105	0.0015712	630.1
13500	0.90718	-11.75	0.59930	17.93	0.66062	0.0015702	630.0
13520	0.90704	-11.79	0.59881	17.92	0.66018	0.0015692	630.0
13540	0.90690	-11.83	0.59833	17.90	0.65975	0.0015682	629.9
13560	0.90677	-11.86	0.59788	17.89	0.65935	0.0015672	629.9
13580	0.90663	-11.90	0.59739	17.87	0.65891	0.0015662	629.8
13600	0.90649	-11.94	0.59691	17.86	0.65848	0.0015651	629.8
13620	0.90635	-11.99	0.59642	17.84	0.65805	0.0015641	629.7
13640	0.90622	-12.02	0.59597	17.83	0.65764	0.0015631	629.7
13660	0.90608	-12.06	0.59549	17.82	0.65722	0.0015621	629.7
13680	0.90594	-12.10	0.59500	17.80	0.65678	0.0015611	629.6
13700	0.90580	-12.14	0.59452	17.79	0.65635	0.0015601	629.6
13720	0.90567	-12.18	0.59407	17.77	0.65595	0.0015591	629.5
13740	0.90553	-12.22	0.59359	17.76	0.65552	0.0015581	629.5
13760	0.90539	-12.26	0.59311	17.75	0.65509	0.0015571	629.4
13780	0.90525	-12.30	0.59263	17.73	0.65466	0.0015561	629.4
13800	0.90512	-12.34	0.59218	17.72	0.65426	0.0015551	629.3
13820	0.90498	-12.38	0.59170	17.70	0.65383	0.0015541	629.3
13840	0.90484	-12.42	0.59122	17.69	0.65340	0.0015531	629.2
13860	0.90470	-12.46	0.59074	17.67	0.65297	0.0015520	629.2
13880	0.90457	-12.50	0.59029	17.66	0.65256	0.0015511	629.1
13900	0.90443	-12.54	0.58981	17.65	0.65213	0.0015500	629.1
13920	0.90429	-12.58	0.58933	17.63	0.65170	0.0015490	629.0
13940	0.90415	-12.62	0.58885	17.62	0.65127	0.0015480	629.0
13960	0.90402	-12.66	0.58841	17.61	0.65088	0.0015471	628.9
13980	0.90388	-12.70	0.58793	17.59	0.65045	0.0015461	628.9
14000	0.90374	-12.74	0.58745	17.58	0.65002	0.0015450	628.8
14020	0.90360	-12.78	0.58697	17.56	0.64959	0.0015440	628.8
14040	0.90347	-12.82	0.58653	17.55	0.64920	0.0015431	628.7
14060	0.90333	-12.86	0.58605	17.53	0.64877	0.0015421	628.7
14080	0.90319	-12.90	0.58557	17.52	0.64834	0.0015410	628.6

H Geopotential Altitude	θ Temperature Ratio	T Standard Temperature	δ Pressure Ratio	P Standard Pressure	σ Density Ratio	ρ Standard Density	a Speed of Sound
ft		deg C		in Hg		slug/ft^3	kn
14100	0.90305	-12.94	0.58510	17.51	0.64792	0.0015400	628.6
14120	0.90292	-12.97	0.58465	17.49	0.64751	0.0015391	628.6
14140	0.90278	-13.01	0.58418	17.48	0.64709	0.0015381	628.5
14160	0.90264	-13.05	0.58370	17.46	0.64666	0.0015370	628.5
14180	0.90250	-13.09	0.58323	17.45	0.64624	0.0015360	628.4
14200	0.90237	-13.13	0.58278	17.44	0.64583	0.0015351	628.4
14220	0.90223	-13.17	0.58231	17.42	0.64541	0.0015341	628.3
14240	0.90209	-13.21	0.58183	17.41	0.64498	0.0015331	628.3
14260	0.90195	-13.25	0.58136	17.39	0.64456	0.0015321	628.2
14280	0.90182	-13.29	0.58092	17.38	0.64416	0.0015311	628.2
14300	0.90168	-13.33	0.58045	17.37	0.64374	0.0015301	628.1
14320	0.90154	-13.37	0.57997	17.35	0.64331	0.0015291	628.1
14340	0.90140	-13.41	0.57950	17.34	0.64289	0.0015281	628.0
14360	0.90127	-13.45	0.57906	17.33	0.64249	0.0015271	628.0
14380	0.90113	-13.49	0.57859	17.31	0.64207	0.0015261	627.9
14400	0.90099	-13.53	0.57812	17.30	0.64165	0.0015251	627.9
14420	0.90085	-13.57	0.57764	17.28	0.64122	0.0015241	627.8
14440	0.90072	-13.61	0.57721	17.27	0.64083	0.0015232	627.8
14460	0.90058	-13.65	0.57673	17.26	0.64040	0.0015222	627.7
14480	0.90044	-13.69	0.57626	17.24	0.63998	0.0015212	627.7
14500	0.90030	-13.73	0.57579	17.23	0.63955	0.0015201	627.6
14520	0.90017	-13.77	0.57536	17.21	0.63917	0.0015192	627.6
14540	0.90003	-13.81	0.57489	17.20	0.63875	0.0015182	627.5
14560	0.89989	-13.85	0.57442	17.19	0.63832	0.0015172	627.5
14580	0.89975	-13.89	0.57395	17.17	0.63790	0.0015162	627.5
14600	0.89962	-13.92	0.57351	17.16	0.63750	0.0015153	627.4
14620	0.89948	-13.96	0.57304	17.15	0.63708	0.0015143	627.4
14640	0.89934	-14.01	0.57257	17.13	0.63666	0.0015133	627.3
14660	0.89920	-14.05	0.57210	17.12	0.63623	0.0015123	627.3
14680	0.89907	-14.08	0.57167	17.10	0.63585	0.0015114	627.2
14700	0.89893	-14.12	0.57120	17.09	0.63542	0.0015103	627.2
14720	0.89879	-14.16	0.57073	17.08	0.63500	0.0015093	627.1
14740	0.89865	-14.20	0.57027	17.06	0.63459	0.0015084	627.1
14760	0.89852	-14.24	0.56983	17.05	0.63419	0.0015074	627.0
14780	0.89838	-14.28	0.56937	17.04	0.63377	0.0015064	627.0

H Geopotential Altitude ft	θ Temperature Ratio	T Standard Temperature deg C	δ Pressure Ratio	P Standard Pressure in Hg	σ Density Ratio	ρ Standard Density slug/ft^3	a Speed of Sound kn
14800	0.89824	-14.32	0.56890	17.02	0.63335	0.0015054	626.9
14820	0.89810	-14.36	0.56844	17.01	0.63294	0.0015044	626.9
14840	0.89797	-14.40	0.56800	16.99	0.63254	0.0015035	626.8
14860	0.89783	-14.44	0.56754	16.98	0.63212	0.0015025	626.8
14880	0.89769	-14.48	0.56707	16.97	0.63170	0.0015015	626.7
14900	0.89755	-14.52	0.56661	16.95	0.63129	0.0015005	626.7
14920	0.89742	-14.56	0.56618	16.94	0.63090	0.0014996	626.6
14940	0.89728	-14.60	0.56571	16.93	0.63047	0.0014986	626.6
14960	0.89714	-14.64	0.56525	16.91	0.63006	0.0014976	626.5
14980	0.89700	-14.68	0.56479	16.90	0.62964	0.0014966	626.5
15000	0.89687	-14.72	0.56436	16.89	0.62926	0.0014957	626.4
15020	0.89673	-14.76	0.56389	16.87	0.62883	0.0014947	626.4
15040	0.89659	-14.80	0.56343	16.86	0.62841	0.0014937	626.3
15060	0.89645	-14.84	0.56297	16.84	0.62800	0.0014927	626.3
15080	0.89632	-14.88	0.56254	16.83	0.62761	0.0014918	626.3
15100	0.89618	-14.92	0.56208	16.82	0.62720	0.0014908	626.2
15120	0.89604	-14.96	0.56162	16.80	0.62678	0.0014898	626.2
15140	0.89590	-15.00	0.56115	16.79	0.62635	0.0014888	626.1
15160	0.89577	-15.03	0.56073	16.78	0.62598	0.0014879	626.1
15180	0.89563	-15.07	0.56027	16.76	0.62556	0.0014869	626.0
15200	0.89549	-15.11	0.55981	16.75	0.62514	0.0014859	626.0
15220	0.89535	-15.15	0.55935	16.74	0.62473	0.0014849	625.9
15240	0.89522	-15.19	0.55892	16.72	0.62434	0.0014840	625.9
15260	0.89508	-15.23	0.55846	16.71	0.62392	0.0014830	625.8
15280	0.89494	-15.27	0.55800	16.70	0.62351	0.0014820	625.8
15300	0.89480	-15.31	0.55754	16.68	0.62309	0.0014810	625.7
15320	0.89467	-15.35	0.55712	16.67	0.62271	0.0014801	625.7
15340	0.89453	-15.39	0.55666	16.66	0.62229	0.0014791	625.6
15360	0.89439	-15.43	0.55620	16.64	0.62188	0.0014781	625.6
15380	0.89425	-15.47	0.55574	16.63	0.62146	0.0014771	625.5
15400	0.89412	-15.51	0.55532	16.62	0.62108	0.0014762	625.5
15420	0.89398	-15.55	0.55486	16.60	0.62066	0.0014752	625.4
15440	0.89384	-15.59	0.55441	16.59	0.62026	0.0014743	625.4
15460	0.89370	-15.63	0.55395	16.57	0.61984	0.0014733	625.3
15480	0.89357	-15.67	0.55353	16.56	0.61946	0.0014724	625.3

H Geopotential Altitude	θ Temperature Ratio	T Standard Temperature	δ Pressure Ratio	P Standard Pressure	σ Density Ratio	ρ Standard Density	a Speed of Sound
ft		deg C		in Hg		slug/ft^3	kn
15500	0.89343	-15.71	0.55307	16.55	0.61904	0.0014714	625.2
15520	0.89329	-15.75	0.55262	16.53	0.61863	0.0014704	625.2
15540	0.89315	-15.79	0.55216	16.52	0.61822	0.0014694	625.1
15560	0.89302	-15.83	0.55174	16.51	0.61784	0.0014685	625.1
15580	0.89288	-15.87	0.55128	16.49	0.61742	0.0014675	625.1
15600	0.89274	-15.91	0.55083	16.48	0.61701	0.0014666	625.0
15620	0.89260	-15.95	0.55038	16.47	0.61660	0.0014656	625.0
15640	0.89247	-15.98	0.54995	16.45	0.61621	0.0014647	624.9
15660	0.89233	-16.03	0.54950	16.44	0.61580	0.0014637	624.9
15680	0.89219	-16.07	0.54905	16.43	0.61540	0.0014627	624.8
15700	0.89205	-16.11	0.54860	16.41	0.61499	0.0014618	624.8
15720	0.89192	-16.14	0.54818	16.40	0.61461	0.0014609	624.7
15740	0.89178	-16.18	0.54772	16.39	0.61419	0.0014599	624.7
15760	0.89164	-16.22	0.54727	16.37	0.61378	0.0014589	624.6
15780	0.89150	-16.26	0.54682	16.36	0.61337	0.0014579	624.6
15800	0.89137	-16.30	0.54640	16.35	0.61299	0.0014570	624.5
15820	0.89123	-16.34	0.54595	16.33	0.61258	0.0014560	624.5
15840	0.89109	-16.38	0.54550	16.32	0.61217	0.0014551	624.4
15860	0.89095	-16.42	0.54505	16.31	0.61176	0.0014541	624.4
15880	0.89082	-16.46	0.54463	16.30	0.61138	0.0014532	624.3
15900	0.89068	-16.50	0.54418	16.28	0.61097	0.0014522	624.3
15920	0.89054	-16.54	0.54373	16.27	0.61056	0.0014512	624.2
15940	0.89040	-16.58	0.54328	16.25	0.61015	0.0014503	624.2
15960	0.89027	-16.62	0.54287	16.24	0.60978	0.0014494	624.1
15980	0.89013	-16.66	0.54242	16.23	0.60937	0.0014484	624.1
16000	0.88999	-16.70	0.54197	16.22	0.60896	0.0014474	624.0
16020	0.88985	-16.74	0.54152	16.20	0.60855	0.0014465	624.0
16040	0.88972	-16.78	0.54111	16.19	0.60818	0.0014456	623.9
16060	0.88958	-16.82	0.54066	16.18	0.60777	0.0014446	623.9
16080	0.88944	-16.86	0.54021	16.16	0.60736	0.0014436	623.8
16100	0.88930	-16.90	0.53977	16.15	0.60696	0.0014427	623.8
16120	0.88917	-16.94	0.53935	16.14	0.60658	0.0014418	623.8
16140	0.88903	-16.98	0.53890	16.12	0.60617	0.0014408	623.7
16160	0.88889	-17.02	0.53846	16.11	0.60577	0.0014399	623.7
16180	0.88875	-17.06	0.53801	16.10	0.60536	0.0014389	623.6

H Geopotential Altitude	θ Temperature Ratio	T Standard Temperature	δ Pressure Ratio	P Standard Pressure	σ Density Ratio	ρ Standard Density	a Speed of Sound
ft		deg C		in Hg		slug/ft^3	kn
16200	0.88862	-17.09	0.53760	16.08	0.60498	0.0014380	623.6
16220	0.88848	-17.13	0.53715	16.07	0.60457	0.0014370	623.5
16240	0.88834	-17.17	0.53671	16.06	0.60417	0.0014361	623.5
16260	0.88820	-17.22	0.53627	16.05	0.60377	0.0014351	623.4
16280	0.88807	-17.25	0.53585	16.03	0.60339	0.0014342	623.4
16300	0.88793	-17.29	0.53541	16.02	0.60299	0.0014332	623.3
16320	0.88779	-17.33	0.53497	16.01	0.60259	0.0014323	623.3
16340	0.88765	-17.37	0.53452	15.99	0.60217	0.0014313	623.2
16360	0.88752	-17.41	0.53411	15.98	0.60180	0.0014304	623.2
16380	0.88738	-17.45	0.53367	15.97	0.60140	0.0014295	623.1
16400	0.88724	-17.49	0.53323	15.95	0.60100	0.0014285	623.1
16420	0.88710	-17.53	0.53278	15.94	0.60059	0.0014275	623.0
16440	0.88697	-17.57	0.53237	15.93	0.60021	0.0014266	623.0
16460	0.88683	-17.61	0.53193	15.92	0.59981	0.0014257	622.9
16480	0.88669	-17.65	0.53149	15.90	0.59941	0.0014247	622.9
16500	0.88655	-17.69	0.53105	15.89	0.59901	0.0014238	622.8
16520	0.88642	-17.73	0.53064	15.88	0.59863	0.0014229	622.8
16540	0.88628	-17.77	0.53020	15.86	0.59823	0.0014219	622.7
16560	0.88614	-17.81	0.52976	15.85	0.59783	0.0014210	622.7
16580	0.88600	-17.85	0.52932	15.84	0.59743	0.0014200	622.6
16600	0.88587	-17.89	0.52891	15.82	0.59705	0.0014191	622.6
16620	0.88573	-17.93	0.52847	15.81	0.59665	0.0014182	622.5
16640	0.88559	-17.97	0.52803	15.80	0.59625	0.0014172	622.5
16660	0.88545	-18.01	0.52760	15.79	0.59586	0.0014163	622.4
16680	0.88532	-18.05	0.52719	15.77	0.59548	0.0014154	622.4
16700	0.88518	-18.09	0.52675	15.76	0.59508	0.0014144	622.3
16720	0.88504	-18.13	0.52631	15.75	0.59467	0.0014135	622.3
16740	0.88490	-18.17	0.52588	15.73	0.59428	0.0014125	622.3
16760	0.88477	-18.20	0.52547	15.72	0.59391	0.0014117	622.2
16780	0.88463	-18.24	0.52503	15.71	0.59350	0.0014107	622.2
16800	0.88449	-18.28	0.52460	15.70	0.59311	0.0014098	622.1
16820	0.88435	-18.32	0.52416	15.68	0.59271	0.0014088	622.1
16840	0.88422	-18.36	0.52376	15.67	0.59234	0.0014079	622.0
16860	0.88408	-18.40	0.52332	15.66	0.59194	0.0014070	622.0
16880	0.88394	-18.44	0.52288	15.64	0.59153	0.0014060	621.9

H Geopotential Altitude	θ Temperature Ratio	T Standard Temperature	δ Pressure Ratio	P Standard Pressure	σ Density Ratio	ρ Standard Density	a Speed of Sound
ft		deg C		in Hg		slug/ft^3	kn
16900	0.88380	-18.48	0.52245	15.63	0.59114	0.0014051	621.9
16920	0.88367	-18.52	0.52205	15.62	0.59077	0.0014042	621.8
16940	0.88353	-18.56	0.52161	15.61	0.59037	0.0014033	621.8
16960	0.88339	-18.60	0.52118	15.59	0.58998	0.0014023	621.7
16980	0.88325	-18.64	0.52074	15.58	0.58957	0.0014013	621.7
17000	0.88312	-18.68	0.52034	15.57	0.58921	0.0014005	621.6
17020	0.88298	-18.72	0.51991	15.56	0.58881	0.0013995	621.6
17040	0.88284	-18.76	0.51947	15.54	0.58841	0.0013986	621.5
17060	0.88270	-18.80	0.51904	15.53	0.58801	0.0013976	621.5
17080	0.88256	-18.84	0.51861	15.52	0.58762	0.0013967	621.4
17100	0.88243	-18.88	0.51821	15.50	0.58725	0.0013958	621.4
17120	0.88229	-18.92	0.51777	15.49	0.58685	0.0013949	621.3
17140	0.88215	-18.96	0.51734	15.48	0.58645	0.0013939	621.3
17160	0.88201	-19.00	0.51691	15.47	0.58606	0.0013930	621.2
17180	0.88188	-19.04	0.51651	15.45	0.58569	0.0013921	621.2
17200	0.88174	-19.08	0.51608	15.44	0.58530	0.0013912	621.1
17220	0.88160	-19.12	0.51565	15.43	0.58490	0.0013902	621.1
17240	0.88146	-19.16	0.51522	15.42	0.58451	0.0013893	621.0
17260	0.88133	-19.19	0.51482	15.40	0.58414	0.0013884	621.0
17280	0.88119	-19.24	0.51439	15.39	0.58374	0.0013875	620.9
17300	0.88105	-19.28	0.51396	15.38	0.58335	0.0013866	620.9
17320	0.88091	-19.32	0.51353	15.36	0.58295	0.0013856	620.8
17340	0.88078	-19.35	0.51313	15.35	0.58259	0.0013848	620.8
17360	0.88064	-19.39	0.51271	15.34	0.58220	0.0013838	620.8
17380	0.88050	-19.43	0.51228	15.33	0.58181	0.0013829	620.7
17400	0.88036	-19.47	0.51185	15.31	0.58141	0.0013820	620.7
17420	0.88023	-19.51	0.51145	15.30	0.58104	0.0013811	620.6
17440	0.88009	-19.55	0.51102	15.29	0.58065	0.0013801	620.6
17460	0.87995	-19.59	0.51060	15.28	0.58026	0.0013792	620.5
17480	0.87981	-19.63	0.51017	15.26	0.57986	0.0013783	620.5
17500	0.87968	-19.67	0.50977	15.25	0.57949	0.0013774	620.4
17520	0.87954	-19.71	0.50935	15.24	0.57911	0.0013765	620.4
17540	0.87940	-19.75	0.50892	15.23	0.57871	0.0013755	620.3
17560	0.87926	-19.79	0.50850	15.21	0.57833	0.0013746	620.3
17580	0.87913	-19.83	0.50810	15.20	0.57796	0.0013738	620.2

H Geopotential Altitude ft	θ Temperature Ratio	T Standard Temperature deg C	δ Pressure Ratio	P Standard Pressure in Hg	σ Density Ratio	ρ Standard Density slug/ft^3	a Speed of Sound kn
17600	0.87899	-19.87	0.50768	15.19	0.57757	0.0013728	620.2
17620	0.87885	-19.91	0.50725	15.18	0.57717	0.0013719	620.1
17640	0.87871	-19.95	0.50683	15.16	0.57679	0.0013710	620.1
17660	0.87858	-19.99	0.50643	15.15	0.57642	0.0013701	620.0
17680	0.87844	-20.03	0.50601	15.14	0.57603	0.0013692	620.0
17700	0.87830	-20.07	0.50559	15.13	0.57565	0.0013683	619.9
17720	0.87816	-20.11	0.50516	15.11	0.57525	0.0013673	619.9
17740	0.87803	-20.15	0.50477	15.10	0.57489	0.0013665	619.8
17760	0.87789	-20.19	0.50435	15.09	0.57450	0.0013655	619.8
17780	0.87775	-20.23	0.50392	15.08	0.57410	0.0013646	619.7
17800	0.87761	-20.27	0.50350	15.06	0.57372	0.0013637	619.7
17820	0.87748	-20.30	0.50311	15.05	0.57336	0.0013628	619.6
17840	0.87734	-20.34	0.50269	15.04	0.57297	0.0013619	619.6
17860	0.87720	-20.38	0.50227	15.03	0.57258	0.0013610	619.5
17880	0.87706	-20.43	0.50185	15.02	0.57220	0.0013601	619.5
17900	0.87693	-20.46	0.50145	15.00	0.57182	0.0013592	619.4
17920	0.87679	-20.50	0.50103	14.99	0.57144	0.0013583	619.4
17940	0.87665	-20.54	0.50061	14.98	0.57105	0.0013573	619.3
17960	0.87651	-20.58	0.50019	14.97	0.57066	0.0013564	619.3
17980	0.87638	-20.62	0.49980	14.95	0.57030	0.0013555	619.2
18000	0.87624	-20.66	0.49938	14.94	0.56991	0.0013546	619.2
18020	0.87610	-20.70	0.49896	14.93	0.56952	0.0013537	619.1
18040	0.87596	-20.74	0.49855	14.92	0.56915	0.0013528	619.1
18060	0.87583	-20.78	0.49816	14.90	0.56879	0.0013520	619.1
18080	0.87569	-20.82	0.49774	14.89	0.56840	0.0013510	619.0
18100	0.87555	-20.86	0.49732	14.88	0.56801	0.0013501	619.0
18120	0.87541	-20.90	0.49690	14.87	0.56762	0.0013492	618.9
18140	0.87528	-20.94	0.49652	14.86	0.56727	0.0013483	618.9
18160	0.87514	-20.98	0.49610	14.84	0.56688	0.0013474	618.8
18180	0.87500	-21.02	0.49568	14.83	0.56649	0.0013465	618.8
18200	0.87486	-21.06	0.49526	14.82	0.56610	0.0013456	618.7
18220	0.87473	-21.10	0.49488	14.81	0.56575	0.0013447	618.7
18240	0.87459	-21.14	0.49446	14.79	0.56536	0.0013438	618.6
18260	0.87445	-21.18	0.49405	14.78	0.56498	0.0013429	618.6
18280	0.87431	-21.22	0.49363	14.77	0.56459	0.0013420	618.5

H Geopotential Altitude	θ Temperature Ratio	T Standard Temperature	δ Pressure Ratio	P Standard Pressure	σ Density Ratio	ρ Standard Density	a Speed of Sound
ft		deg C		in Hg		slug/ft^3	kn
18300	0.87418	-21.26	0.49324	14.76	0.56423	0.0013411	618.5
18320	0.87404	-21.30	0.49283	14.75	0.56385	0.0013402	618.4
18340	0.87390	-21.34	0.49241	14.73	0.56346	0.0013393	618.4
18360	0.87376	-21.38	0.49200	14.72	0.56308	0.0013384	618.3
18380	0.87363	-21.41	0.49162	14.71	0.56273	0.0013376	618.3
18400	0.87349	-21.45	0.49120	14.70	0.56234	0.0013366	618.2
18420	0.87335	-21.49	0.49079	14.68	0.56196	0.0013357	618.2
18440	0.87321	-21.53	0.49037	14.67	0.56157	0.0013348	618.1
18460	0.87308	-21.57	0.48999	14.66	0.56122	0.0013340	618.1
18480	0.87294	-21.61	0.48958	14.65	0.56084	0.0013331	618.0
18500	0.87280	-21.65	0.48917	14.64	0.56046	0.0013322	618.0
18520	0.87266	-21.69	0.48875	14.62	0.56007	0.0013312	617.9
18540	0.87253	-21.73	0.48837	14.61	0.55972	0.0013304	617.9
18560	0.87239	-21.77	0.48796	14.60	0.55934	0.0013295	617.8
18580	0.87225	-21.81	0.48755	14.59	0.55896	0.0013286	617.8
18600	0.87211	-21.85	0.48714	14.58	0.55858	0.0013277	617.7
18620	0.87198	-21.89	0.48676	14.56	0.55822	0.0013268	617.7
18640	0.87184	-21.93	0.48634	14.55	0.55783	0.0013259	617.6
18660	0.87170	-21.97	0.48593	14.54	0.55745	0.0013250	617.6
18680	0.87156	-22.01	0.48552	14.53	0.55707	0.0013241	617.5
18700	0.87143	-22.05	0.48514	14.52	0.55672	0.0013233	617.5
18720	0.87129	-22.09	0.48473	14.50	0.55634	0.0013224	617.4
18740	0.87115	-22.13	0.48432	14.49	0.55595	0.0013214	617.4
18760	0.87101	-22.17	0.48392	14.48	0.55558	0.0013206	617.3
18780	0.87088	-22.21	0.48354	14.47	0.55523	0.0013197	617.3
18800	0.87074	-22.25	0.48313	14.46	0.55485	0.0013188	617.3
18820	0.87060	-22.29	0.48272	14.44	0.55447	0.0013179	617.2
18840	0.87046	-22.33	0.48231	14.43	0.55409	0.0013170	617.2
18860	0.87033	-22.36	0.48193	14.42	0.55373	0.0013162	617.1
18880	0.87019	-22.40	0.48153	14.41	0.55336	0.0013153	617.1
18900	0.87005	-22.45	0.48112	14.40	0.55298	0.0013144	617.0
18920	0.86991	-22.49	0.48071	14.38	0.55260	0.0013135	617.0
18940	0.86978	-22.52	0.48033	14.37	0.55224	0.0013126	616.9
18960	0.86964	-22.56	0.47993	14.36	0.55187	0.0013117	616.9
18980	0.86950	-22.60	0.47952	14.35	0.55149	0.0013108	616.8

H Geopotential Altitude	θ Temperature Ratio	T Standard Temperature	δ Pressure Ratio	P Standard Pressure	σ Density Ratio	ρ Standard Density	a Speed of Sound
ft		deg C		in Hg		slug/ft^3	kn
19000	0.86936	-22.64	0.47912	14.34	0.55112	0.0013100	616.8
19020	0.86923	-22.68	0.47874	14.32	0.55076	0.0013091	616.7
19040	0.86909	-22.72	0.47834	14.31	0.55039	0.0013082	616.7
19060	0.86895	-22.76	0.47793	14.30	0.55001	0.0013073	616.6
19080	0.86881	-22.80	0.47753	14.29	0.54964	0.0013064	616.6
19100	0.86868	-22.84	0.47715	14.28	0.54928	0.0013056	616.5
19120	0.86854	-22.88	0.47675	14.26	0.54891	0.0013047	616.5
19140	0.86840	-22.92	0.47634	14.25	0.54853	0.0013038	616.4
19160	0.86826	-22.96	0.47594	14.24	0.54815	0.0013029	616.4
19180	0.86813	-23.00	0.47557	14.23	0.54781	0.0013021	616.3
19200	0.86799	-23.04	0.47516	14.22	0.54743	0.0013012	616.3
19220	0.86785	-23.08	0.47476	14.20	0.54705	0.0013003	616.2
19240	0.86771	-23.12	0.47436	14.19	0.54668	0.0012994	616.2
19260	0.86758	-23.16	0.47398	14.18	0.54632	0.0012985	616.1
19280	0.86744	-23.20	0.47358	14.17	0.54595	0.0012977	616.1
19300	0.86730	-23.24	0.47318	14.16	0.54558	0.0012968	616.0
19320	0.86716	-23.28	0.47278	14.15	0.54521	0.0012959	616.0
19340	0.86703	-23.32	0.47241	14.13	0.54486	0.0012951	615.9
19360	0.86689	-23.36	0.47201	14.12	0.54449	0.0012942	615.9
19380	0.86675	-23.40	0.47161	14.11	0.54411	0.0012933	615.8
19400	0.86661	-23.44	0.47120	14.10	0.54373	0.0012924	615.8
19420	0.86648	-23.47	0.47083	14.09	0.54338	0.0012916	615.7
19440	0.86634	-23.51	0.47043	14.08	0.54301	0.0012907	615.7
19460	0.86620	-23.55	0.47003	14.06	0.54263	0.0012898	615.6
19480	0.86606	-23.59	0.46964	14.05	0.54227	0.0012889	615.6
19500	0.86593	-23.63	0.46926	14.04	0.54191	0.0012881	615.5
19520	0.86579	-23.67	0.46887	14.03	0.54155	0.0012872	615.5
19540	0.86565	-23.71	0.46847	14.02	0.54118	0.0012863	615.4
19560	0.86551	-23.75	0.46807	14.00	0.54080	0.0012854	615.4
19580	0.86538	-23.79	0.46770	13.99	0.54046	0.0012846	615.3
19600	0.86524	-23.83	0.46730	13.98	0.54008	0.0012837	615.3
19620	0.86510	-23.87	0.46691	13.97	0.53972	0.0012829	615.3
19640	0.86496	-23.91	0.46651	13.96	0.53934	0.0012820	615.2
19660	0.86483	-23.95	0.46614	13.95	0.53900	0.0012811	615.2
19680	0.86469	-23.99	0.46574	13.93	0.53862	0.0012802	615.1

H Geopotential Altitude ft	θ Temperature Ratio	T Standard Temperature deg C	δ Pressure Ratio	P Standard Pressure in Hg	σ Density Ratio	ρ Standard Density slug/ft^3	a Speed of Sound kn
19700	0.86455	-24.03	0.46535	13.92	0.53826	0.0012794	615.1
19720	0.86441	-24.07	0.46495	13.91	0.53788	0.0012785	615.0
19740	0.86428	-24.11	0.46458	13.90	0.53753	0.0012777	615.0
19760	0.86414	-24.15	0.46419	13.89	0.53717	0.0012768	614.9
19780	0.86400	-24.19	0.46379	13.88	0.53679	0.0012759	614.9
19800	0.86386	-24.23	0.46340	13.86	0.53643	0.0012750	614.8
19820	0.86373	-24.27	0.46303	13.85	0.53608	0.0012742	614.8
19840	0.86359	-24.31	0.46264	13.84	0.53572	0.0012734	614.7
19860	0.86345	-24.35	0.46224	13.83	0.53534	0.0012724	614.7
19880	0.86331	-24.39	0.46185	13.82	0.53498	0.0012716	614.6
19900	0.86318	-24.42	0.46148	13.81	0.53463	0.0012708	614.6
19920	0.86304	-24.47	0.46109	13.80	0.53426	0.0012699	614.5
19940	0.86290	-24.51	0.46070	13.78	0.53390	0.0012690	614.5
19960	0.86276	-24.55	0.46031	13.77	0.53353	0.0012681	614.4
19980	0.86263	-24.58	0.45994	13.76	0.53318	0.0012673	614.4
20000	0.86249	-24.62	0.45955	13.75	0.53282	0.0012665	614.3
20020	0.86235	-24.66	0.45916	13.74	0.53245	0.0012656	614.3
20040	0.86221	-24.70	0.45877	13.73	0.53209	0.0012647	614.2
20060	0.86208	-24.74	0.45840	13.72	0.53174	0.0012639	614.2
20080	0.86194	-24.78	0.45801	13.70	0.53137	0.0012630	614.1
20100	0.86180	-24.82	0.45762	13.69	0.53100	0.0012621	614.1
20120	0.86166	-24.86	0.45723	13.68	0.53064	0.0012613	614.0
20140	0.86153	-24.90	0.45687	13.67	0.53030	0.0012605	614.0
20160	0.86139	-24.94	0.45648	13.66	0.52993	0.0012596	613.9
20180	0.86125	-24.98	0.45609	13.65	0.52957	0.0012587	613.9
20200	0.86111	-25.02	0.45570	13.63	0.52920	0.0012579	613.8
20220	0.86098	-25.06	0.45534	13.62	0.52886	0.0012570	613.8
20240	0.86084	-25.10	0.45495	13.61	0.52850	0.0012562	613.7
20260	0.86070	-25.14	0.45456	13.60	0.52813	0.0012553	613.7
20280	0.86056	-25.18	0.45417	13.59	0.52776	0.0012544	613.6
20300	0.86043	-25.22	0.45381	13.58	0.52742	0.0012536	613.6
20320	0.86029	-25.26	0.45342	13.57	0.52705	0.0012527	613.5
20340	0.86015	-25.30	0.45303	13.55	0.52669	0.0012519	613.5
20360	0.86001	-25.34	0.45265	13.54	0.52633	0.0012510	613.4
20380	0.85988	-25.38	0.45229	13.53	0.52599	0.0012502	613.4

H Geopotential Altitude	θ Temperature Ratio	T Standard Temperature	δ Pressure Ratio	P Standard Pressure	σ Density Ratio	ρ Standard Density	a Speed of Sound
ft		deg C		in Hg		slug/ft^3	kn
20400	0.85974	-25.42	0.45190	13.52	0.52562	0.0012493	613.3
20420	0.85960	-25.46	0.45151	13.51	0.52526	0.0012485	613.3
20440	0.85946	-25.50	0.45113	13.50	0.52490	0.0012476	613.2
20460	0.85933	-25.53	0.45077	13.49	0.52456	0.0012468	613.2
20480	0.85919	-25.57	0.45038	13.48	0.52419	0.0012459	613.1
20500	0.85905	-25.61	0.45000	13.46	0.52383	0.0012451	613.1
20520	0.85891	-25.66	0.44961	13.45	0.52347	0.0012442	613.0
20540	0.85878	-25.69	0.44925	13.44	0.52313	0.0012434	613.0
20560	0.85864	-25.73	0.44887	13.43	0.52277	0.0012426	612.9
20580	0.85850	-25.77	0.44849	13.42	0.52241	0.0012417	612.9
20600	0.85836	-25.81	0.44810	13.41	0.52204	0.0012408	612.8
20620	0.85823	-25.85	0.44774	13.40	0.52170	0.0012400	612.8
20640	0.85809	-25.89	0.44736	13.39	0.52134	0.0012392	612.8
20660	0.85795	-25.93	0.44698	13.37	0.52099	0.0012383	612.7
20680	0.85781	-25.97	0.44659	13.36	0.52062	0.0012375	612.7
20700	0.85768	-26.01	0.44624	13.35	0.52029	0.0012367	612.6
20720	0.85754	-26.05	0.44586	13.34	0.51993	0.0012358	612.6
20740	0.85740	-26.09	0.44547	13.33	0.51956	0.0012349	612.5
20760	0.85726	-26.13	0.44509	13.32	0.51920	0.0012341	612.5
20780	0.85713	-26.17	0.44474	13.31	0.51887	0.0012333	612.4
20800	0.85699	-26.21	0.44435	13.29	0.51850	0.0012324	612.4
20820	0.85685	-26.25	0.44397	13.28	0.51814	0.0012316	612.3
20840	0.85671	-26.29	0.44359	13.27	0.51778	0.0012307	612.3
20860	0.85658	-26.33	0.44324	13.26	0.51745	0.0012299	612.2
20880	0.85644	-26.37	0.44286	13.25	0.51709	0.0012291	612.2
20900	0.85630	-26.41	0.44248	13.24	0.51673	0.0012282	612.1
20920	0.85616	-26.45	0.44210	13.23	0.51638	0.0012274	612.1
20940	0.85603	-26.48	0.44174	13.22	0.51603	0.0012266	612.0
20960	0.85589	-26.53	0.44137	13.21	0.51569	0.0012257	612.0
20980	0.85575	-26.57	0.44099	13.19	0.51533	0.0012249	611.9
21000	0.85561	-26.61	0.44061	13.18	0.51497	0.0012240	611.9
21020	0.85548	-26.64	0.44026	13.17	0.51464	0.0012232	611.8
21040	0.85534	-26.68	0.43988	13.16	0.51428	0.0012224	611.8
21060	0.85520	-26.72	0.43950	13.15	0.51391	0.0012215	611.7
21080	0.85506	-26.76	0.43912	13.14	0.51355	0.0012207	611.7

H Geopotential Altitude	θ Temperature Ratio	T Standard Temperature	δ Pressure Ratio	P Standard Pressure	σ Density Ratio	ρ Standard Density	a Speed of Sound
ft		deg C		in Hg		slug/ft^3	kn
21100	0.85493	-26.80	0.43877	13.13	0.51322	0.0012199	611.6
21120	0.85479	-26.84	0.43839	13.12	0.51286	0.0012190	611.6
21140	0.85465	-26.88	0.43801	13.11	0.51250	0.0012182	611.5
21160	0.85451	-26.92	0.43764	13.09	0.51215	0.0012173	611.5
21180	0.85438	-26.96	0.43729	13.08	0.51182	0.0012165	611.4
21200	0.85424	-27.00	0.43691	13.07	0.51146	0.0012157	611.4
21220	0.85410	-27.04	0.43654	13.06	0.51111	0.0012149	611.3
21240	0.85396	-27.08	0.43616	13.05	0.51075	0.0012140	611.3
21260	0.85383	-27.12	0.43581	13.04	0.51042	0.0012132	611.2
21280	0.85369	-27.16	0.43543	13.03	0.51006	0.0012124	611.2
21300	0.85355	-27.20	0.43506	13.02	0.50971	0.0012115	611.1
21320	0.85341	-27.24	0.43468	13.01	0.50934	0.0012107	611.1
21340	0.85328	-27.28	0.43434	13.00	0.50902	0.0012099	611.0
21360	0.85314	-27.32	0.43396	12.98	0.50866	0.0012090	611.0
21380	0.85300	-27.36	0.43359	12.97	0.50831	0.0012082	610.9
21400	0.85286	-27.40	0.43321	12.96	0.50795	0.0012073	610.9
21420	0.85272	-27.44	0.43284	12.95	0.50760	0.0012065	610.8
21440	0.85259	-27.48	0.43249	12.94	0.50727	0.0012057	610.8
21460	0.85245	-27.52	0.43212	12.93	0.50692	0.0012049	610.7
21480	0.85231	-27.56	0.43175	12.92	0.50656	0.0012040	610.7
21500	0.85217	-27.60	0.43138	12.91	0.50621	0.0012032	610.6
21520	0.85204	-27.63	0.43103	12.90	0.50588	0.0012024	610.6
21540	0.85190	-27.68	0.43066	12.89	0.50553	0.0012016	610.5
21560	0.85176	-27.72	0.43029	12.87	0.50518	0.0012008	610.5
21580	0.85162	-27.76	0.42991	12.86	0.50481	0.0011999	610.4
21600	0.85149	-27.79	0.42957	12.85	0.50449	0.0011991	610.4
21620	0.85135	-27.83	0.42920	12.84	0.50414	0.0011983	610.3
21640	0.85121	-27.87	0.42883	12.83	0.50379	0.0011975	610.3
21660	0.85107	-27.91	0.42846	12.82	0.50344	0.0011966	610.2
21680	0.85094	-27.95	0.42811	12.81	0.50310	0.0011958	610.2
21700	0.85080	-27.99	0.42774	12.80	0.50275	0.0011950	610.1
21720	0.85066	-28.03	0.42737	12.79	0.50240	0.0011942	610.1
21740	0.85052	-28.07	0.42700	12.78	0.50205	0.0011933	610.0
21760	0.85039	-28.11	0.42666	12.77	0.50172	0.0011925	610.0
21780	0.85025	-28.15	0.42629	12.75	0.50137	0.0011917	609.9

H Geopotential Altitude	θ Temperature Ratio	T Standard Temperature	δ Pressure Ratio	P Standard Pressure	σ Density Ratio	ρ Standard Density	a Speed of Sound
ft		deg C		in Hg		slug/ft^3	kn
21800	0.85011	-28.19	0.42592	12.74	0.50102	0.0011909	609.9
21820	0.84997	-28.23	0.42555	12.73	0.50066	0.0011900	609.8
21840	0.84984	-28.27	0.42521	12.72	0.50034	0.0011893	609.8
21860	0.84970	-28.31	0.42484	12.71	0.49999	0.0011884	609.7
21880	0.84956	-28.35	0.42448	12.70	0.49965	0.0011876	609.7
21900	0.84942	-28.39	0.42411	12.69	0.49929	0.0011868	609.6
21920	0.84929	-28.43	0.42377	12.68	0.49897	0.0011860	609.6
21940	0.84915	-28.47	0.42340	12.67	0.49862	0.0011852	609.6
21960	0.84901	-28.51	0.42303	12.66	0.49826	0.0011843	609.5
21980	0.84887	-28.55	0.42267	12.65	0.49792	0.0011835	609.5
22000	0.84874	-28.59	0.42233	12.64	0.49760	0.0011827	609.4
22020	0.84860	-28.63	0.42196	12.63	0.49724	0.0011819	609.4
22040	0.84846	-28.67	0.42160	12.61	0.49690	0.0011811	609.3
22060	0.84832	-28.71	0.42123	12.60	0.49655	0.0011802	609.3
22080	0.84819	-28.74	0.42089	12.59	0.49622	0.0011795	609.2
22100	0.84805	-28.78	0.42053	12.58	0.49588	0.0011787	609.2
22120	0.84791	-28.82	0.42016	12.57	0.49552	0.0011778	609.1
22140	0.84777	-28.87	0.41980	12.56	0.49518	0.0011770	609.1
22160	0.84764	-28.90	0.41946	12.55	0.49486	0.0011762	609.0
22180	0.84750	-28.94	0.41909	12.54	0.49450	0.0011754	609.0
22200	0.84736	-28.98	0.41873	12.53	0.49416	0.0011746	608.9
22220	0.84722	-29.02	0.41837	12.52	0.49382	0.0011738	608.9
22240	0.84709	-29.06	0.41803	12.51	0.49349	0.0011730	608.8
22260	0.84695	-29.10	0.41767	12.50	0.49315	0.0011722	608.8
22280	0.84681	-29.14	0.41730	12.49	0.49279	0.0011713	608.7
22300	0.84667	-29.18	0.41694	12.47	0.49245	0.0011705	608.7
22320	0.84654	-29.22	0.41661	12.46	0.49213	0.0011697	608.6
22340	0.84640	-29.26	0.41624	12.45	0.49178	0.0011689	608.6
22360	0.84626	-29.30	0.41588	12.44	0.49143	0.0011681	608.5
22380	0.84612	-29.34	0.41552	12.43	0.49109	0.0011673	608.5
22400	0.84599	-29.38	0.41518	12.42	0.49076	0.0011665	608.4
22420	0.84585	-29.42	0.41482	12.41	0.49042	0.0011657	608.4
22440	0.84571	-29.46	0.41446	12.40	0.49007	0.0011648	608.3
22460	0.84557	-29.50	0.41410	12.39	0.48973	0.0011640	608.3
22480	0.84544	-29.54	0.41377	12.38	0.48941	0.0011633	608.2

H Geopotential Altitude	θ Temperature Ratio	T Standard Temperature	δ Pressure Ratio	P Standard Pressure	σ Density Ratio	ρ Standard Density	a Speed of Sound
ft		deg C		in Hg		slug/ft^3	kn
22500	0.84530	-29.58	0.41341	12.37	0.48907	0.0011625	608.2
22520	0.84516	-29.62	0.41305	12.36	0.48872	0.0011616	608.1
22540	0.84502	-29.66	0.41269	12.35	0.48838	0.0011608	608.1
22560	0.84489	-29.69	0.41236	12.34	0.48806	0.0011601	608.0
22580	0.84475	-29.74	0.41200	12.33	0.48772	0.0011593	608.0
22600	0.84461	-29.78	0.41164	12.32	0.48737	0.0011584	607.9
22620	0.84447	-29.82	0.41128	12.31	0.48703	0.0011576	607.9
22640	0.84434	-29.85	0.41095	12.30	0.48671	0.0011569	607.8
22660	0.84420	-29.89	0.41059	12.28	0.48637	0.0011561	607.8
22680	0.84406	-29.93	0.41023	12.27	0.48602	0.0011552	607.7
22700	0.84392	-29.97	0.40987	12.26	0.48567	0.0011544	607.7
22720	0.84379	-30.01	0.40954	12.25	0.48536	0.0011537	607.6
22740	0.84365	-30.05	0.40918	12.24	0.48501	0.0011528	607.6
22760	0.84351	-30.09	0.40883	12.23	0.48468	0.0011520	607.5
22780	0.84337	-30.13	0.40847	12.22	0.48433	0.0011512	607.5
22800	0.84324	-30.17	0.40814	12.21	0.48401	0.0011504	607.4
22820	0.84310	-30.21	0.40778	12.20	0.48367	0.0011496	607.4
22840	0.84296	-30.25	0.40743	12.19	0.48333	0.0011488	607.3
22860	0.84282	-30.29	0.40707	12.18	0.48299	0.0011480	607.3
22880	0.84269	-30.33	0.40674	12.17	0.48267	0.0011473	607.2
22900	0.84255	-30.37	0.40639	12.16	0.48233	0.0011465	607.2
22920	0.84241	-30.41	0.40603	12.15	0.48199	0.0011456	607.1
22940	0.84227	-30.45	0.40568	12.14	0.48165	0.0011448	607.1
22960	0.84214	-30.49	0.40535	12.13	0.48133	0.0011441	607.0
22980	0.84200	-30.53	0.40500	12.12	0.48100	0.0011433	607.0
23000	0.84186	-30.57	0.40464	12.11	0.48065	0.0011425	606.9
23020	0.84172	-30.61	0.40429	12.10	0.48031	0.0011416	606.9
23040	0.84159	-30.65	0.40396	12.09	0.48000	0.0011409	606.8
23060	0.84145	-30.69	0.40361	12.08	0.47966	0.0011401	606.8
23080	0.84131	-30.73	0.40325	12.07	0.47931	0.0011393	606.7
23100	0.84117	-30.77	0.40290	12.05	0.47898	0.0011385	606.7
23120	0.84104	-30.80	0.40257	12.04	0.47866	0.0011377	606.6
23140	0.84090	-30.84	0.40222	12.03	0.47832	0.0011369	606.6
23160	0.84076	-30.89	0.40187	12.02	0.47798	0.0011361	606.5
23180	0.84062	-30.93	0.40152	12.01	0.47765	0.0011353	606.5

H Geopotential Altitude	θ Temperature Ratio	T Standard Temperature	δ Pressure Ratio	P Standard Pressure	σ Density Ratio	ρ Standard Density	a Speed of Sound
ft		deg C		in Hg		slug/ft^3	kn
23200	0.84049	-30.96	0.40119	12.00	0.47733	0.0011346	606.4
23220	0.84035	-31.00	0.40084	11.99	0.47699	0.0011338	606.4
23240	0.84021	-31.04	0.40049	11.98	0.47665	0.0011329	606.3
23260	0.84007	-31.08	0.40014	11.97	0.47632	0.0011322	606.3
23280	0.83994	-31.12	0.39982	11.96	0.47601	0.0011314	606.2
23300	0.83980	-31.16	0.39947	11.95	0.47567	0.0011306	606.2
23320	0.83966	-31.20	0.39912	11.94	0.47534	0.0011298	606.1
23340	0.83952	-31.24	0.39877	11.93	0.47500	0.0011290	606.1
23360	0.83939	-31.28	0.39844	11.92	0.47468	0.0011283	606.0
23380	0.83925	-31.32	0.39809	11.91	0.47434	0.0011275	606.0
23400	0.83911	-31.36	0.39774	11.90	0.47400	0.0011267	605.9
23420	0.83897	-31.40	0.39739	11.89	0.47366	0.0011258	605.9
23440	0.83884	-31.44	0.39707	11.88	0.47336	0.0011251	605.8
23460	0.83870	-31.48	0.39672	11.87	0.47302	0.0011243	605.8
23480	0.83856	-31.52	0.39637	11.86	0.47268	0.0011235	605.7
23500	0.83842	-31.56	0.39603	11.85	0.47235	0.0011227	605.7
23520	0.83829	-31.60	0.39570	11.84	0.47203	0.0011220	605.6
23540	0.83815	-31.64	0.39536	11.83	0.47171	0.0011212	605.6
23560	0.83801	-31.68	0.39501	11.82	0.47137	0.0011204	605.5
23580	0.83787	-31.72	0.39466	11.81	0.47103	0.0011196	605.5
23600	0.83774	-31.76	0.39434	11.80	0.47072	0.0011189	605.4
23620	0.83760	-31.80	0.39400	11.79	0.47039	0.0011181	605.4
23640	0.83746	-31.84	0.39365	11.78	0.47005	0.0011173	605.3
23660	0.83732	-31.88	0.39330	11.77	0.46971	0.0011165	605.3
23680	0.83719	-31.91	0.39298	11.76	0.46940	0.0011157	605.2
23700	0.83705	-31.95	0.39264	11.75	0.46908	0.0011150	605.2
23720	0.83691	-31.99	0.39229	11.74	0.46874	0.0011141	605.1
23740	0.83677	-32.03	0.39195	11.73	0.46841	0.0011134	605.1
23760	0.83664	-32.07	0.39163	11.72	0.46810	0.0011126	605.0
23780	0.83650	-32.11	0.39128	11.71	0.46776	0.0011118	605.0
23800	0.83636	-32.15	0.39094	11.70	0.46743	0.0011110	604.9
23820	0.83622	-32.19	0.39060	11.69	0.46710	0.0011102	604.9
23840	0.83609	-32.23	0.39028	11.68	0.46679	0.0011095	604.8
23860	0.83595	-32.27	0.38993	11.67	0.46645	0.0011087	604.8
23880	0.83581	-32.31	0.38959	11.66	0.46612	0.0011079	604.7

H Geopotential Altitude	θ Temperature Ratio	T Standard Temperature	δ Pressure Ratio	P Standard Pressure	σ Density Ratio	ρ Standard Density	a Speed of Sound
ft		deg C		in Hg		slug/ft^3	kn
23900	0.83567	-32.35	0.38925	11.65	0.46579	0.0011071	604.7
23920	0.83554	-32.39	0.38893	11.64	0.46548	0.0011064	604.6
23940	0.83540	-32.43	0.38859	11.63	0.46515	0.0011056	604.6
23960	0.83526	-32.47	0.38824	11.62	0.46481	0.0011048	604.5
23980	0.83512	-32.51	0.38790	11.61	0.46448	0.0011040	604.5
24000	0.83499	-32.55	0.38759	11.60	0.46419	0.0011033	604.4
24020	0.83485	-32.59	0.38724	11.59	0.46384	0.0011025	604.4
24040	0.83471	-32.63	0.38690	11.58	0.46351	0.0011017	604.3
24060	0.83457	-32.67	0.38656	11.57	0.46318	0.0011009	604.3
24080	0.83444	-32.71	0.38625	11.56	0.46289	0.0011002	604.2
24100	0.83430	-32.75	0.38591	11.55	0.46256	0.0010995	604.2
24120	0.83416	-32.79	0.38556	11.54	0.46221	0.0010986	604.1
24140	0.83402	-32.83	0.38522	11.53	0.46188	0.0010978	604.1
24160	0.83389	-32.86	0.38491	11.52	0.46158	0.0010971	604.1
24180	0.83375	-32.90	0.38457	11.51	0.46125	0.0010963	604.0
24200	0.83361	-32.95	0.38423	11.50	0.46092	0.0010956	603.9
24220	0.83347	-32.99	0.38389	11.49	0.46059	0.0010948	603.9
24240	0.83334	-33.02	0.38358	11.48	0.46029	0.0010941	603.9
24260	0.83320	-33.06	0.38324	11.47	0.45996	0.0010933	603.8
24280	0.83306	-33.10	0.38290	11.46	0.45963	0.0010925	603.7
24300	0.83292	-33.14	0.38256	11.45	0.45930	0.0010917	603.7
24320	0.83279	-33.18	0.38225	11.44	0.45900	0.0010910	603.7
24340	0.83265	-33.22	0.38191	11.43	0.45867	0.0010902	603.6
24360	0.83251	-33.26	0.38157	11.42	0.45834	0.0010894	603.6
24380	0.83237	-33.30	0.38124	11.41	0.45802	0.0010887	603.5
24400	0.83224	-33.34	0.38092	11.40	0.45770	0.0010879	603.5
24420	0.83210	-33.38	0.38059	11.39	0.45738	0.0010871	603.4
24440	0.83196	-33.42	0.38025	11.38	0.45705	0.0010864	603.4
24460	0.83182	-33.46	0.37991	11.37	0.45672	0.0010856	603.3
24480	0.83169	-33.50	0.37960	11.36	0.45642	0.0010849	603.3
24500	0.83155	-33.54	0.37927	11.35	0.45610	0.0010841	603.2
24520	0.83141	-33.58	0.37893	11.34	0.45577	0.0010833	603.2
24540	0.83127	-33.62	0.37860	11.33	0.45545	0.0010826	603.1
24560	0.83114	-33.66	0.37828	11.32	0.45513	0.0010818	603.1
24580	0.83100	-33.70	0.37795	11.31	0.45481	0.0010810	603.0

H Geopotential Altitude ft	θ Temperature Ratio	T Standard Temperature deg C	δ Pressure Ratio	P Standard Pressure in Hg	σ Density Ratio	ρ Standard Density slug/ft^3	a Speed of Sound kn
24600	0.83086	-33.74	0.37762	11.30	0.45449	0.0010803	603.0
24620	0.83072	-33.78	0.37728	11.29	0.45416	0.0010795	602.9
24640	0.83059	-33.82	0.37697	11.28	0.45386	0.0010788	602.9
24660	0.83045	-33.86	0.37664	11.27	0.45354	0.0010780	602.8
24680	0.83031	-33.90	0.37630	11.26	0.45320	0.0010772	602.8
24700	0.83017	-33.94	0.37597	11.25	0.45288	0.0010765	602.7
24720	0.83004	-33.97	0.37566	11.24	0.45258	0.0010757	602.7
24740	0.82990	-34.01	0.37533	11.23	0.45226	0.0010750	602.6
24760	0.82976	-34.05	0.37499	11.22	0.45193	0.0010742	602.6
24780	0.82962	-34.09	0.37466	11.21	0.45160	0.0010734	602.5
24800	0.82949	-34.13	0.37435	11.20	0.45130	0.0010727	602.5
24820	0.82935	-34.17	0.37402	11.19	0.45098	0.0010719	602.4
24840	0.82921	-34.21	0.37369	11.18	0.45066	0.0010712	602.4
24860	0.82907	-34.25	0.37336	11.17	0.45034	0.0010704	602.3
24880	0.82894	-34.29	0.37305	11.16	0.45003	0.0010697	602.3
24900	0.82880	-34.33	0.37272	11.15	0.44971	0.0010689	602.2
24920	0.82866	-34.37	0.37239	11.14	0.44939	0.0010682	602.2
24940	0.82852	-34.41	0.37206	11.13	0.44907	0.0010674	602.1
24960	0.82839	-34.45	0.37175	11.12	0.44876	0.0010667	602.1
24980	0.82825	-34.49	0.37142	11.11	0.44844	0.0010659	602.0
25000	0.82811	-34.53	0.37109	11.10	0.44812	0.0010651	602.0
25020	0.82797	-34.57	0.37076	11.09	0.44779	0.0010644	601.9
25040	0.82784	-34.61	0.37046	11.08	0.44750	0.0010637	601.9
25060	0.82770	-34.65	0.37013	11.07	0.44718	0.0010629	601.8
25080	0.82756	-34.69	0.36980	11.06	0.44686	0.0010621	601.8
25100	0.82742	-34.73	0.36947	11.05	0.44653	0.0010614	601.7
25120	0.82729	-34.77	0.36917	11.05	0.44624	0.0010607	601.7
25140	0.82715	-34.81	0.36884	11.04	0.44592	0.0010599	601.6
25160	0.82701	-34.85	0.36851	11.03	0.44559	0.0010591	601.6
25180	0.82687	-34.89	0.36818	11.02	0.44527	0.0010584	601.5
25200	0.82674	-34.92	0.36788	11.01	0.44498	0.0010577	601.5
25220	0.82660	-34.97	0.36755	11.00	0.44465	0.0010569	601.4
25240	0.82646	-35.01	0.36722	10.99	0.44433	0.0010561	601.4
25260	0.82632	-35.05	0.36690	10.98	0.44402	0.0010554	601.3
25280	0.82619	-35.08	0.36659	10.97	0.44371	0.0010547	601.3

H Geopotential Altitude	θ Temperature Ratio	T Standard Temperature	δ Pressure Ratio	P Standard Pressure	σ Density Ratio	ρ Standard Density	a Speed of Sound
ft		deg C		in Hg		slug/ft^3	kn
25300	0.82605	-35.12	0.36627	10.96	0.44340	0.0010539	601.2
25320	0.82591	-35.16	0.36594	10.95	0.44307	0.0010531	601.2
25340	0.82577	-35.20	0.36561	10.94	0.44275	0.0010524	601.1
25360	0.82564	-35.24	0.36531	10.93	0.44246	0.0010517	601.1
25380	0.82550	-35.28	0.36499	10.92	0.44214	0.0010509	601.0
25400	0.82536	-35.32	0.36466	10.91	0.44182	0.0010502	601.0
25420	0.82522	-35.36	0.36434	10.90	0.44151	0.0010494	600.9
25440	0.82509	-35.40	0.36403	10.89	0.44120	0.0010487	600.9
25460	0.82495	-35.44	0.36371	10.88	0.44089	0.0010480	600.8
25480	0.82481	-35.48	0.36339	10.87	0.44057	0.0010472	600.8
25500	0.82467	-35.52	0.36306	10.86	0.44025	0.0010464	600.7
25520	0.82454	-35.56	0.36276	10.85	0.43995	0.0010457	600.7
25540	0.82440	-35.60	0.36244	10.84	0.43964	0.0010450	600.6
25560	0.82426	-35.64	0.36211	10.83	0.43932	0.0010442	600.6
25580	0.82412	-35.68	0.36179	10.82	0.43900	0.0010435	600.5
25600	0.82399	-35.72	0.36149	10.82	0.43871	0.0010428	600.5
25620	0.82385	-35.76	0.36117	10.81	0.43839	0.0010420	600.4
25640	0.82371	-35.80	0.36085	10.80	0.43808	0.0010413	600.4
25660	0.82357	-35.84	0.36052	10.79	0.43775	0.0010405	600.3
25680	0.82343	-35.88	0.36020	10.78	0.43744	0.0010398	600.2
25700	0.82330	-35.92	0.35990	10.77	0.43714	0.0010390	600.2
25720	0.82316	-35.96	0.35958	10.76	0.43683	0.0010383	600.2
25740	0.82302	-36.00	0.35926	10.75	0.43651	0.0010375	600.1
25760	0.82288	-36.04	0.35894	10.74	0.43620	0.0010368	600.0
25780	0.82275	-36.07	0.35864	10.73	0.43590	0.0010361	600.0
25800	0.82261	-36.11	0.35832	10.72	0.43559	0.0010354	600.0
25820	0.82247	-36.16	0.35800	10.71	0.43527	0.0010346	599.9
25840	0.82233	-36.20	0.35768	10.70	0.43496	0.0010339	599.8
25860	0.82220	-36.23	0.35738	10.69	0.43466	0.0010331	599.8
25880	0.82206	-36.27	0.35706	10.68	0.43435	0.0010324	599.8
25900	0.82192	-36.31	0.35674	10.67	0.43403	0.0010316	599.7
25920	0.82178	-36.35	0.35642	10.66	0.43372	0.0010309	599.6
25940	0.82165	-36.39	0.35613	10.66	0.43343	0.0010302	599.6
25960	0.82151	-36.43	0.35581	10.65	0.43312	0.0010295	599.5
25980	0.82137	-36.47	0.35549	10.64	0.43280	0.0010287	599.5

H Geopotential Altitude	θ Temperature Ratio	T Standard Temperature	δ Pressure Ratio	P Standard Pressure	σ Density Ratio	ρ Standard Density	a Speed of Sound
ft		deg C		in Hg		slug/ft^3	kn
26000	0.82123	-36.51	0.35517	10.63	0.43249	0.0010280	599.4
26020	0.82110	-36.55	0.35488	10.62	0.43220	0.0010273	599.4
26040	0.82096	-36.59	0.35456	10.61	0.43188	0.0010265	599.3
26060	0.82082	-36.63	0.35424	10.60	0.43157	0.0010258	599.3
26080	0.82068	-36.67	0.35392	10.59	0.43125	0.0010250	599.2
26100	0.82055	-36.71	0.35363	10.58	0.43097	0.0010244	599.2
26120	0.82041	-36.75	0.35331	10.57	0.43065	0.0010236	599.1
26140	0.82027	-36.79	0.35300	10.56	0.43035	0.0010229	599.1
26160	0.82013	-36.83	0.35268	10.55	0.43003	0.0010221	599.0
26180	0.82000	-36.87	0.35239	10.54	0.42974	0.0010214	599.0
26200	0.81986	-36.91	0.35207	10.53	0.42943	0.0010207	598.9
26220	0.81972	-36.95	0.35175	10.52	0.42911	0.0010200	598.9
26240	0.81958	-36.99	0.35144	10.52	0.42880	0.0010192	598.8
26260	0.81945	-37.03	0.35114	10.51	0.42851	0.0010185	598.8
26280	0.81931	-37.07	0.35083	10.50	0.42820	0.0010178	598.7
26300	0.81917	-37.11	0.35051	10.49	0.42788	0.0010170	598.7
26320	0.81903	-37.15	0.35020	10.48	0.42758	0.0010163	598.6
26340	0.81890	-37.18	0.34991	10.47	0.42729	0.0010156	598.6
26360	0.81876	-37.22	0.34959	10.46	0.42697	0.0010149	598.5
26380	0.81862	-37.26	0.34928	10.45	0.42667	0.0010142	598.5
26400	0.81848	-37.30	0.34897	10.44	0.42636	0.0010134	598.4
26420	0.81835	-37.34	0.34867	10.43	0.42606	0.0010127	598.4
26440	0.81821	-37.38	0.34836	10.42	0.42576	0.0010120	598.3
26460	0.81807	-37.42	0.34805	10.41	0.42545	0.0010113	598.3
26480	0.81793	-37.46	0.34773	10.40	0.42513	0.0010105	598.2
26500	0.81780	-37.50	0.34744	10.40	0.42485	0.0010098	598.2
26520	0.81766	-37.54	0.34713	10.39	0.42454	0.0010091	598.1
26540	0.81752	-37.58	0.34682	10.38	0.42423	0.0010084	598.1
26560	0.81738	-37.62	0.34651	10.37	0.42393	0.0010076	598.0
26580	0.81725	-37.66	0.34622	10.36	0.42364	0.0010069	598.0
26600	0.81711	-37.70	0.34591	10.35	0.42333	0.0010062	597.9
26620	0.81697	-37.74	0.34559	10.34	0.42301	0.0010055	597.9
26640	0.81683	-37.78	0.34528	10.33	0.42271	0.0010047	597.8
26660	0.81670	-37.82	0.34500	10.32	0.42243	0.0010041	597.8
26680	0.81656	-37.86	0.34468	10.31	0.42211	0.0010033	597.7

H Geopotential Altitude	θ Temperature Ratio	T Standard Temperature	δ Pressure Ratio	P Standard Pressure	σ Density Ratio	ρ Standard Density	a Speed of Sound
ft		deg C		in Hg		slug/ft^3	kn
26700	0.81642	-37.90	0.34437	10.30	0.42180	0.0010026	597.7
26720	0.81628	-37.94	0.34406	10.29	0.42150	0.0010019	597.6
26740	0.81615	-37.98	0.34378	10.29	0.42122	0.0010012	597.6
26760	0.81601	-38.02	0.34347	10.28	0.42091	0.0010005	597.5
26780	0.81587	-38.06	0.34316	10.27	0.42061	0.0009997	597.5
26800	0.81573	-38.10	0.34285	10.26	0.42030	0.0009990	597.4
26820	0.81560	-38.13	0.34256	10.25	0.42001	0.0009983	597.4
26840	0.81546	-38.18	0.34225	10.24	0.41970	0.0009976	597.3
26860	0.81532	-38.22	0.34194	10.23	0.41939	0.0009968	597.3
26880	0.81518	-38.26	0.34163	10.22	0.41909	0.0009961	597.2
26900	0.81505	-38.29	0.34135	10.21	0.41881	0.0009955	597.2
26920	0.81491	-38.33	0.34104	10.20	0.41850	0.0009947	597.1
26940	0.81477	-38.37	0.34073	10.19	0.41819	0.0009940	597.1
26960	0.81463	-38.41	0.34042	10.19	0.41788	0.0009933	597.0
26980	0.81450	-38.45	0.34014	10.18	0.41761	0.0009926	597.0
27000	0.81436	-38.49	0.33983	10.17	0.41730	0.0009919	596.9
27020	0.81422	-38.53	0.33952	10.16	0.41699	0.0009911	596.9
27040	0.81408	-38.57	0.33922	10.15	0.41669	0.0009904	596.8
27060	0.81395	-38.61	0.33893	10.14	0.41640	0.0009897	596.8
27080	0.81381	-38.65	0.33863	10.13	0.41610	0.0009890	596.7
27100	0.81367	-38.69	0.33832	10.12	0.41580	0.0009883	596.7
27120	0.81353	-38.73	0.33801	10.11	0.41549	0.0009876	596.6
27140	0.81340	-38.77	0.33773	10.10	0.41521	0.0009869	596.6
27160	0.81326	-38.81	0.33743	10.10	0.41491	0.0009862	596.5
27180	0.81312	-38.85	0.33712	10.09	0.41460	0.0009855	596.5
27200	0.81298	-38.89	0.33682	10.08	0.41430	0.0009847	596.4
27220	0.81285	-38.93	0.33653	10.07	0.41401	0.0009841	596.4
27240	0.81271	-38.97	0.33623	10.06	0.41371	0.0009833	596.3
27260	0.81257	-39.01	0.33592	10.05	0.41340	0.0009826	596.3
27280	0.81243	-39.05	0.33562	10.04	0.41311	0.0009819	596.2
27300	0.81230	-39.09	0.33534	10.03	0.41283	0.0009813	596.2
27320	0.81216	-39.13	0.33503	10.02	0.41252	0.0009805	596.1
27340	0.81202	-39.17	0.33473	10.02	0.41222	0.0009798	596.1
27360	0.81188	-39.21	0.33443	10.01	0.41192	0.0009791	596.0
27380	0.81175	-39.24	0.33415	10.00	0.41164	0.0009784	596.0

H Geopotential Altitude	θ Temperature Ratio	T Standard Temperature	δ Pressure Ratio	P Standard Pressure	σ Density Ratio	ρ Standard Density	a Speed of Sound
ft		deg C		in Hg		slug/ft^3	kn
27400	0.81161	-39.28	0.33384	9.99	0.41133	0.0009777	595.9
27420	0.81147	-39.32	0.33354	9.98	0.41103	0.0009770	595.9
27440	0.81133	-39.37	0.33324	9.97	0.41073	0.0009763	595.8
27460	0.81120	-39.40	0.33296	9.96	0.41045	0.0009756	595.8
27480	0.81106	-39.44	0.33266	9.95	0.41015	0.0009749	595.7
27500	0.81092	-39.48	0.33235	9.94	0.40984	0.0009741	595.7
27520	0.81078	-39.52	0.33205	9.93	0.40954	0.0009734	595.6
27540	0.81065	-39.56	0.33177	9.93	0.40926	0.0009728	595.6
27560	0.81051	-39.60	0.33147	9.92	0.40896	0.0009721	595.5
27580	0.81037	-39.64	0.33117	9.91	0.40867	0.0009714	595.5
27600	0.81023	-39.68	0.33087	9.90	0.40837	0.0009707	595.4
27620	0.81010	-39.72	0.33059	9.89	0.40809	0.0009700	595.4
27640	0.80996	-39.76	0.33029	9.88	0.40779	0.0009693	595.3
27660	0.80982	-39.80	0.32999	9.87	0.40749	0.0009686	595.3
27680	0.80968	-39.84	0.32969	9.86	0.40719	0.0009678	595.2
27700	0.80955	-39.88	0.32941	9.86	0.40691	0.0009672	595.2
27720	0.80941	-39.92	0.32911	9.85	0.40660	0.0009664	595.1
27740	0.80927	-39.96	0.32882	9.84	0.40632	0.0009658	595.1
27760	0.80913	-40.00	0.32852	9.83	0.40602	0.0009651	595.0
27780	0.80900	-40.04	0.32824	9.82	0.40574	0.0009644	595.0
27800	0.80886	-40.08	0.32794	9.81	0.40543	0.0009637	594.9
27820	0.80872	-40.12	0.32764	9.80	0.40513	0.0009630	594.9
27840	0.80858	-40.16	0.32734	9.79	0.40483	0.0009622	594.8
27860	0.80845	-40.20	0.32707	9.79	0.40456	0.0009616	594.8
27880	0.80831	-40.24	0.32677	9.78	0.40426	0.0009609	594.7
27900	0.80817	-40.28	0.32647	9.77	0.40396	0.0009602	594.7
27920	0.80803	-40.32	0.32618	9.76	0.40367	0.0009595	594.6
27940	0.80790	-40.35	0.32590	9.75	0.40339	0.0009588	594.6
27960	0.80776	-40.39	0.32560	9.74	0.40309	0.0009581	594.5
27980	0.80762	-40.43	0.32531	9.73	0.40280	0.0009574	594.5
28000	0.80748	-40.47	0.32501	9.72	0.40250	0.0009567	594.4
28020	0.80735	-40.51	0.32474	9.72	0.40223	0.0009561	594.4
28040	0.80721	-40.55	0.32444	9.71	0.40193	0.0009553	594.3
28060	0.80707	-40.59	0.32414	9.70	0.40163	0.0009546	594.3
28080	0.80693	-40.63	0.32385	9.69	0.40134	0.0009539	594.2

H Geopotential Altitude	θ Temperature Ratio	T Standard Temperature	δ Pressure Ratio	P Standard Pressure	σ Density Ratio	ρ Standard Density	a Speed of Sound
ft		deg C		in Hg		slug/ft^3	kn
28100	0.80680	-40.67	0.32357	9.68	0.40105	0.0009533	594.2
28120	0.80666	-40.71	0.32328	9.67	0.40076	0.0009526	594.1
28140	0.80652	-40.75	0.32298	9.66	0.40046	0.0009519	594.1
28160	0.80638	-40.79	0.32269	9.65	0.40017	0.0009512	594.0
28180	0.80625	-40.83	0.32242	9.65	0.39990	0.0009505	594.0
28200	0.80611	-40.87	0.32212	9.64	0.39960	0.0009498	593.9
28220	0.80597	-40.91	0.32183	9.63	0.39931	0.0009491	593.9
28240	0.80583	-40.95	0.32154	9.62	0.39902	0.0009484	593.8
28260	0.80570	-40.99	0.32126	9.61	0.39873	0.0009477	593.8
28280	0.80556	-41.03	0.32097	9.60	0.39844	0.0009471	593.7
28300	0.80542	-41.07	0.32068	9.59	0.39815	0.0009464	593.6
28320	0.80528	-41.11	0.32038	9.59	0.39785	0.0009456	593.6
28340	0.80515	-41.15	0.32011	9.58	0.39758	0.0009450	593.5
28360	0.80501	-41.19	0.31982	9.57	0.39729	0.0009443	593.5
28380	0.80487	-41.23	0.31953	9.56	0.39700	0.0009436	593.4
28400	0.80473	-41.27	0.31923	9.55	0.39669	0.0009429	593.4
28420	0.80460	-41.30	0.31896	9.54	0.39642	0.0009423	593.3
28440	0.80446	-41.34	0.31867	9.53	0.39613	0.0009416	593.3
28460	0.80432	-41.39	0.31838	9.53	0.39584	0.0009409	593.2
28480	0.80418	-41.43	0.31809	9.52	0.39555	0.0009402	593.2
28500	0.80405	-41.46	0.31782	9.51	0.39527	0.0009395	593.1
28520	0.80391	-41.50	0.31753	9.50	0.39498	0.0009388	593.1
28540	0.80377	-41.54	0.31724	9.49	0.39469	0.0009381	593.0
28560	0.80363	-41.58	0.31695	9.48	0.39440	0.0009374	593.0
28580	0.80350	-41.62	0.31668	9.48	0.39413	0.0009368	592.9
28600	0.80336	-41.66	0.31639	9.47	0.39383	0.0009361	592.9
28620	0.80322	-41.70	0.31610	9.46	0.39354	0.0009354	592.8
28640	0.80308	-41.74	0.31581	9.45	0.39325	0.0009347	592.8
28660	0.80295	-41.78	0.31554	9.44	0.39298	0.0009341	592.7
28680	0.80281	-41.82	0.31525	9.43	0.39268	0.0009334	592.7
28700	0.80267	-41.86	0.31496	9.42	0.39239	0.0009327	592.6
28720	0.80253	-41.90	0.31467	9.41	0.39210	0.0009320	592.6
28740	0.80240	-41.94	0.31441	9.41	0.39184	0.0009314	592.5
28760	0.80226	-41.98	0.31412	9.40	0.39154	0.0009307	592.5
28780	0.80212	-42.02	0.31383	9.39	0.39125	0.0009300	592.4

H Geopotential Altitude	θ Temperature Ratio	T Standard Temperature	δ Pressure Ratio	P Standard Pressure	σ Density Ratio	ρ Standard Density	a Speed of Sound
ft		deg C		in Hg		slug/ft^3	kn
28800	0.80198	-42.06	0.31354	9.38	0.39096	0.0009293	592.4
28820	0.80185	-42.10	0.31328	9.37	0.39070	0.0009287	592.3
28840	0.80171	-42.14	0.31299	9.36	0.39040	0.0009279	592.3
28860	0.80157	-42.18	0.31270	9.36	0.39011	0.0009273	592.2
28880	0.80143	-42.22	0.31241	9.35	0.38982	0.0009266	592.2
28900	0.80130	-42.26	0.31215	9.34	0.38955	0.0009259	592.1
28920	0.80116	-42.30	0.31186	9.33	0.38926	0.0009252	592.1
28940	0.80102	-42.34	0.31158	9.32	0.38898	0.0009246	592.0
28960	0.80088	-42.38	0.31129	9.31	0.38868	0.0009239	592.0
28980	0.80075	-42.41	0.31102	9.31	0.38841	0.0009232	591.9
29000	0.80061	-42.45	0.31074	9.30	0.38813	0.0009225	591.9
29020	0.80047	-42.49	0.31045	9.29	0.38783	0.0009218	591.8
29040	0.80033	-42.53	0.31017	9.28	0.38755	0.0009212	591.8
29060	0.80020	-42.57	0.30990	9.27	0.38728	0.0009205	591.7
29080	0.80006	-42.61	0.30962	9.26	0.38700	0.0009199	591.7
29100	0.79992	-42.65	0.30933	9.26	0.38670	0.0009191	591.6
29120	0.79978	-42.69	0.30905	9.25	0.38642	0.0009185	591.6
29140	0.79965	-42.73	0.30878	9.24	0.38614	0.0009178	591.5
29160	0.79951	-42.77	0.30850	9.23	0.38586	0.0009172	591.5
29180	0.79937	-42.81	0.30822	9.22	0.38558	0.0009165	591.4
29200	0.79923	-42.85	0.30793	9.21	0.38528	0.0009158	591.4
29220	0.79910	-42.89	0.30767	9.21	0.38502	0.0009152	591.3
29240	0.79896	-42.93	0.30739	9.20	0.38474	0.0009145	591.3
29260	0.79882	-42.97	0.30710	9.19	0.38444	0.0009138	591.2
29280	0.79868	-43.01	0.30682	9.18	0.38416	0.0009131	591.2
29300	0.79855	-43.05	0.30656	9.17	0.38390	0.0009125	591.1
29320	0.79841	-43.09	0.30628	9.16	0.38361	0.0009118	591.1
29340	0.79827	-43.13	0.30599	9.16	0.38332	0.0009111	591.0
29360	0.79813	-43.17	0.30571	9.15	0.38303	0.0009104	591.0
29380	0.79800	-43.21	0.30545	9.14	0.38277	0.0009098	590.9
29400	0.79786	-43.25	0.30517	9.13	0.38249	0.0009091	590.9
29420	0.79772	-43.29	0.30489	9.12	0.38220	0.0009085	590.8
29440	0.79758	-43.33	0.30461	9.11	0.38192	0.0009078	590.8
29460	0.79745	-43.36	0.30435	9.11	0.38165	0.0009071	590.7
29480	0.79731	-43.41	0.30406	9.10	0.38136	0.0009065	590.7

H Geopotential Altitude	θ Temperature Ratio	T Standard Temperature	δ Pressure Ratio	P Standard Pressure	σ Density Ratio	ρ Standard Density	a Speed of Sound
ft		deg C		in Hg		slug/ft^3	kn
29500	0.79717	-43.45	0.30378	9.09	0.38107	0.0009058	590.6
29520	0.79703	-43.49	0.30350	9.08	0.38079	0.0009051	590.5
29540	0.79690	-43.52	0.30324	9.07	0.38052	0.0009045	590.5
29560	0.79676	-43.56	0.30296	9.06	0.38024	0.0009038	590.4
29580	0.79662	-43.60	0.30268	9.06	0.37996	0.0009031	590.4
29600	0.79648	-43.64	0.30240	9.05	0.37967	0.0009024	590.3
29620	0.79635	-43.68	0.30215	9.04	0.37942	0.0009018	590.3
29640	0.79621	-43.72	0.30187	9.03	0.37913	0.0009012	590.2
29660	0.79607	-43.76	0.30159	9.02	0.37885	0.0009005	590.2
29680	0.79593	-43.80	0.30131	9.02	0.37856	0.0008998	590.1
29700	0.79580	-43.84	0.30105	9.01	0.37830	0.0008992	590.1
29720	0.79566	-43.88	0.30077	9.00	0.37801	0.0008985	590.0
29740	0.79552	-43.92	0.30049	8.99	0.37773	0.0008978	590.0
29760	0.79538	-43.96	0.30022	8.98	0.37745	0.0008972	589.9
29780	0.79525	-44.00	0.29996	8.97	0.37719	0.0008965	589.9
29800	0.79511	-44.04	0.29968	8.97	0.37690	0.0008959	589.8
29820	0.79497	-44.08	0.29940	8.96	0.37662	0.0008952	589.8
29840	0.79483	-44.12	0.29913	8.95	0.37634	0.0008945	589.7
29860	0.79470	-44.16	0.29887	8.94	0.37608	0.0008939	589.7
29880	0.79456	-44.20	0.29859	8.93	0.37579	0.0008932	589.6
29900	0.79442	-44.24	0.29832	8.93	0.37552	0.0008926	589.6
29920	0.79428	-44.28	0.29804	8.92	0.37523	0.0008919	589.5
29940	0.79414	-44.32	0.29776	8.91	0.37495	0.0008912	589.5
29960	0.79401	-44.36	0.29751	8.90	0.37469	0.0008906	589.4
29980	0.79387	-44.40	0.29723	8.89	0.37441	0.0008899	589.4
30000	0.79373	-44.44	0.29696	8.89	0.37413	0.0008893	589.3
30020	0.79359	-44.48	0.29668	8.88	0.37385	0.0008886	589.3
30040	0.79346	-44.51	0.29643	8.87	0.37359	0.0008880	589.2
30060	0.79332	-44.55	0.29615	8.86	0.37330	0.0008873	589.2
30080	0.79318	-44.60	0.29588	8.85	0.37303	0.0008867	589.1
30100	0.79304	-44.64	0.29560	8.84	0.37274	0.0008860	589.1
30120	0.79291	-44.67	0.29535	8.84	0.37249	0.0008854	589.0
30140	0.79277	-44.71	0.29507	8.83	0.37220	0.0008847	589.0
30160	0.79263	-44.75	0.29480	8.82	0.37193	0.0008840	588.9
30180	0.79249	-44.79	0.29453	8.81	0.37165	0.0008834	588.9

APPENDIX VII

LEVEL FLIGHT WORKING CHARTS

APPENDIX VII

LEVEL FLIGHT WORKING CHARTS

VII.1 GENERAL

The technique of obtaining level flight data at prescribed conditions requires using a programmable, handheld calculator or a working plot which aids the flight test crew in maintaining a constant W / σ or W / δ and $N_R / \sqrt{\theta}$ ratio. The working plot assists the pilot in flying a pressure altitude and outside air temperature which corresponds to the desired conditions. For a known engine start gross weight (ESGW), a curve is constructed to show the variation of altitude with pounds of fuel used in terms of fuel counter which is displayed to the crew.

VII.2 CONSTANT W / σ METHOD

The crew uses the curve in Figure VII.1 by programming in advance a fuel counter number, entering vertically to the desired W / σ value, moving horizontally to the ambient temperature, then proceeding vertically to obtain the required pressure altitude.

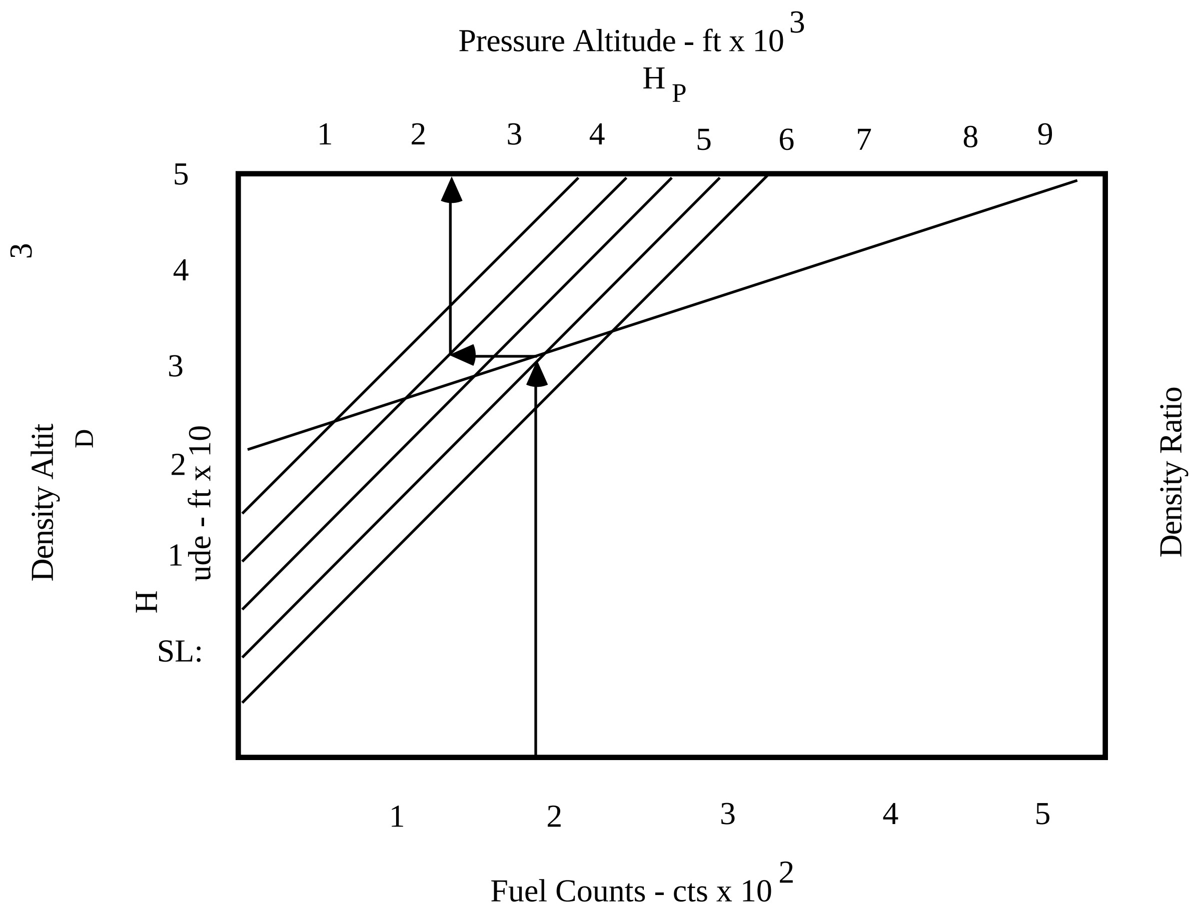

Figure VII.1

Using the W/σ Working Chart

The problem of constructing a constant W / σ chart is dependent on the starting gross weight and atmospheric conditions. In the following example, it is assumed that the desired take-off gross weight can be obtained by ballasting or reducing fuel loading. The example aircraft and test conditions are given below:

Aircraft weight:

Pilots + 30 minutes of fuel	10,000 lb
Pilots + full fuel	12,000 lb
Max gross weight (including ballast)	15,000 lb

Test conditions:

W / σ	14,000 lb
OAT at surface	+10 °C
H_P at surface	-200 ft

Example 1

1. Surface conditions of +10 °C and -200 ft H_P were obtained from the weather service and are used to compute surface density altitude, H_D, and surface density ratio, σ:

Surface H_D	-850 ft
Surface σ	1.02504

2. A desired W / σ of 14,000 lb was specified for testing and it is necessary to determine the ESGW to provide the W / σ at engine start:

$$ESGW = W / \sigma\ (\sigma)\ \text{(at surface)}$$
$$ESGW = 14{,}000\ (1.02504)$$
$$ESGW = 14{,}350 \text{ lb}$$

3. The ballast is determined :

Ballast = ESGW - Pilot and fuel weight
Ballast = 14,350 - 12,000
Ballast = 2,350 lb

4. Plot the ESGW on the W / σ chart (Figure VII.2).

5. Determine a second gross weight which corresponds to the expected weight after one hour of flight:

ESGW	14,350 lb
-Fuel used (one hour)	409 lb
GW (after one hour)	13,941 lb

6. Determine σ required for GW = 13,941 lb to obtain W / σ = 14,000 lb:

σ = 13,941 / 14,000 = 0.99579

H_D = 145 ft

7. Plot GW after one hour on the working chart. Connect the two points to obtain the desired constant W / σ = 14,000 lb.

If for the same ambient conditions, a W / σ of 15,500 lb is desired, a different approach to constructing the working plot is required since the max GW is 15,000 lb.

Example 2

1. Determine σ required for W / σ at max GW of 15,000 lb:

W / σ = 15,500 lb

σ = 15,000 / 15,500 = 0.96774

H_D = 1,115 ft

2. Plot the ESGW on the W / σ chart.

3. Determine a second gross weight which corresponds to the expected weight after one hour of flight:

ESGW	15,000 lb
-Fuel used (one hour)	500 lb
GW (after one hour)	14,500 lb

4. Determine σ required for GW = 14,500 lb to obtain W / σ = 15,500 lb:

$$\sigma = 14{,}500 / 15{,}500 = 0.93548$$

$$H_D = 2{,}260 \text{ ft}$$

5. Plot GW after one hour on the working chart. Connect the two points to obtain the desired constant W / σ = 15,500 lb.

If altitude restrictions are imposed on the test conditions, adjust the test altitude range accordingly. This often requires starting at an altitude of 1,000 ft AGL. Compute a corresponding starting σ.

Make allowances in the fuel loading required to reach the test area from the departure point.

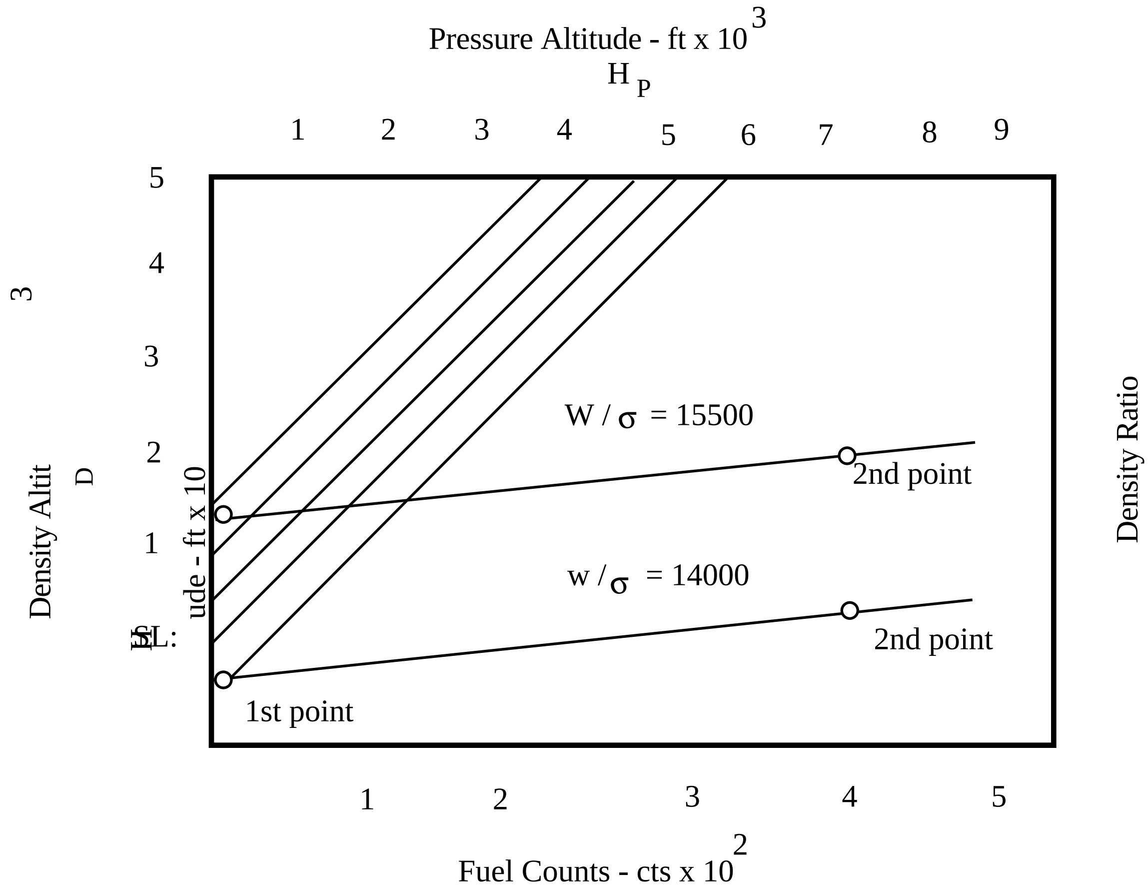

Figure VII.2
Constructing a W/σ Working Chart

VII.3 CONSTANT W / δ AND N_R / √θ METHOD

For the W / δ and N_R / $\sqrt{\theta}$ method, two charts are required to maintain a constant W / δ and N_R / $\sqrt{\theta}$ ratio. For this example, W / δ = 14,000 lb and N_R / $\sqrt{\theta}$ = 100 % is used.

1. Surface conditions of +10 °C and -200 ft H_P were obtained from the weather service and are used to compute surface pressure ratio, δ, and surface temperature ratio, θ:

Surface pressure ratio, δ	1.00725
Surface temperature ratio, θ	0.98265

2. A desired W / δ of 14,000 lb was specified for testing and it is necessary to determine the ESGW to provide the W / δ at engine start:

ESGW = (W / δ) (δ, at surface)
ESGW = 14,000 (1.00725)
ESGW = 14,102 lb

3. The ballast is determined:

Ballast = ESGW - Pilot and fuel weight
Ballast = 14,102- 12,000
Ballast = 2,102 lb

4. Plot the ESGW on the W / δ chart (Figure VII.3).

5. Determine a second gross weight which corresponds to the expected weight after one hour of flight:

ESGW	14,102 lb
-Fuel used (one hour)	409 lb
GW (after one hour)	13,693 lb

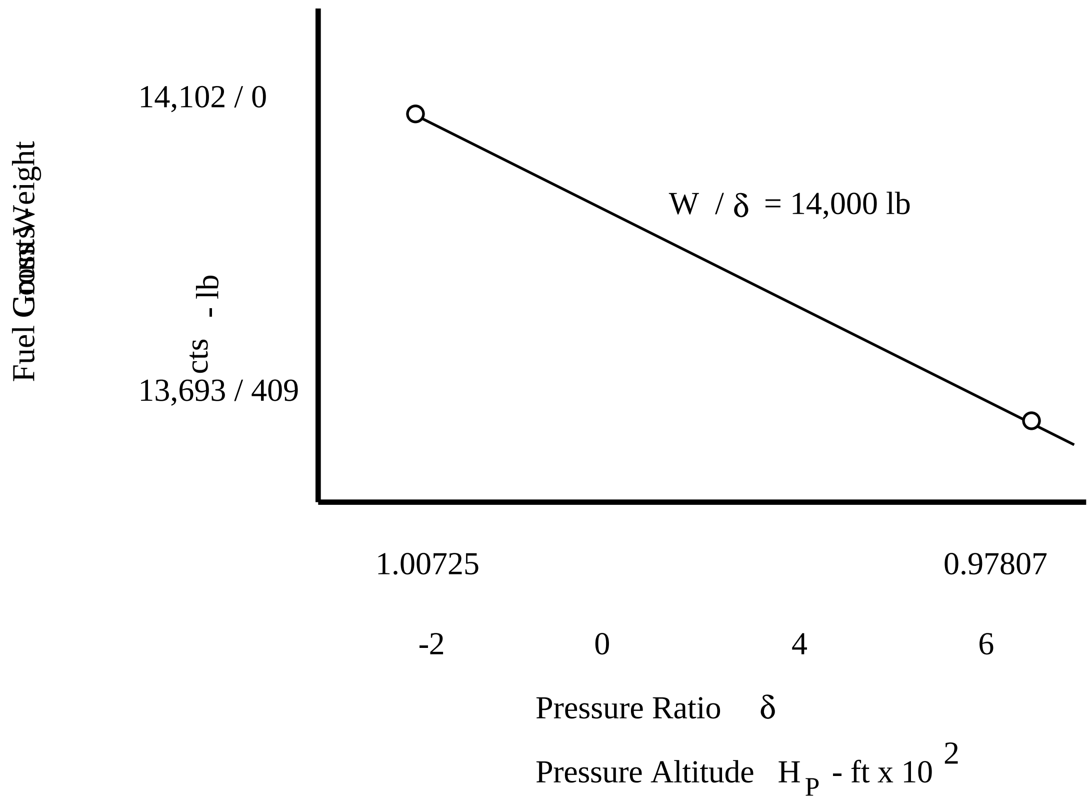

Figure VII.3
W/δ Working Chart

6. Determine δ and H_P required for GW = 13,693 lb to obtain W / δ = 14,000 lb:

$$\delta = 13{,}693 / 14{,}000 = 0.97807$$

$$H_P = 610 \text{ ft}$$

7. Plot GW after one hour on the working chart. Connect the two points to obtain the desired constant W / δ = 14,000lb (Figure VII.3).

8. Within the altitude range determined in 7 above, determine the outside air temperature from soundings or use the standard lapse rate. Determine the corresponding θ:

$$\theta = 0.97710 \quad (H_P = 610)$$

9. For $N_R / \sqrt{\theta} = 100$ % and $\theta = 0.98265$, determine N_R:

$$N_R = 100\,(\sqrt{\theta})$$

$$N_R = 100\,(0.99129)$$

$$N_R = 99.1\ \%$$

10. For $N_R / \sqrt{\theta} = 100$ % and $\theta = 0.97710$ determine N_R:

$$N_R = 100\,(\sqrt{\theta})$$

$$N_R = 100\,(0.98848)$$

$$N_R = 98.8\%$$

11. Plot and connect the two N_R points (Figure VII.4).

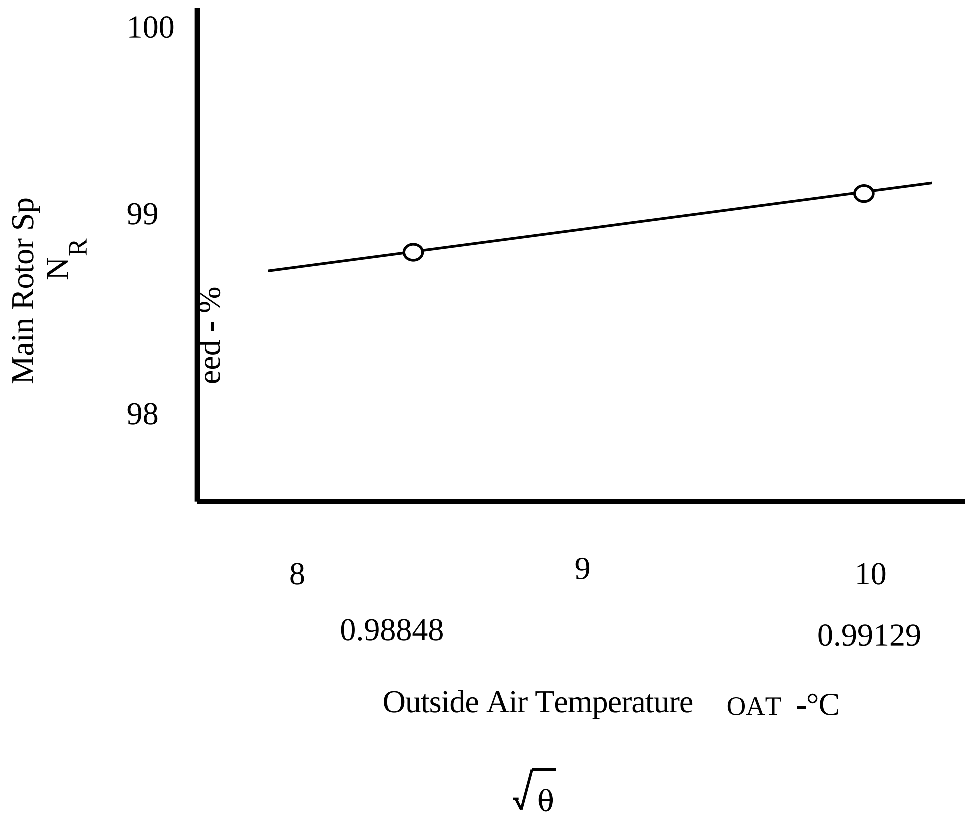

Figure VII.4
$N_R/\sqrt{\theta}$ **Working Chart**

If altitude restrictions are imposed on the test conditions, adjust the test altitude range accordingly. This often requires starting at an altitude of 1,000 ft AGL. Compute a corresponding starting δ , H_P, and N_R.

Make allowances in the fuel loading required to reach the test area from the departure point.