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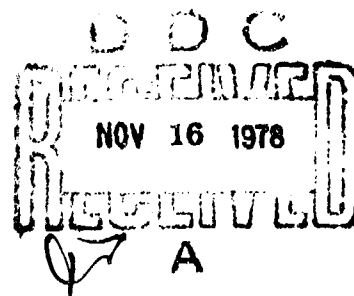
USNTPS-FTM-No. 104

# NAVAL TEST PILOT SCHOOL FLIGHT TEST MANUAL

## FIXED WING PERFORMANCE

### THEORY AND FLIGHT TEST TECHNIQUES

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NAVAL AIR TEST CENTER  
PATUXENT RIVER, MARYLAND

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Mrs. Crumbocker:

Attached is the Manual, "Fixed Wing Performance Theory and Flight Test Techniques" which we discussed on the telephone. The distribution sheet is inserted as first page. Please send us AD Number on this revised copy dated July 1977.

Thank you for all your help.

  
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test techniques to obtain a specific result more efficiently. All flight testing described in this manual is conducted to define engine and airframe performance characteristics for use in determining mission suitability and specification compliance where applicable.

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**FIXED WING PERFORMANCE  
THEORY  
AND  
FLIGHT TEST TECHNIQUES**

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## PERFORMANCE FLIGHT TESTS AND DATA REDUCTION

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- II. PERFORMANCE FLIGHT TEST CONDITIONS AND PILOT TECHNIQUES
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REVISIONS

Insert latest changes, destroying superseded pages.

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## PREFACE

"Consideration of the human variables involved in flight leads me to the conclusion that the pilot must be as wise in decision as a good chess player and as correct in his tempo of transaction as a bullfighter."

L. I. KELLY  
Aeronautics

The U.S. Naval Test Pilot School Performance Testing Manual has been prepared as a guide for the student who is learning the principles of performance flight testing. The flight techniques discussed herein are many of those which are used in performance flight test projects. The principles illustrated provide a basis for modifying these techniques to fit a specific performance test problem.

The acquisition of data and data analysis are fundamental to performance flight testing. Frequently it is a very laborious process. At the U.S. Naval Test Pilot School, the quantity of data obtained and the data analysis methods used are limited to those necessary to teach the principles. Though the data analysis methods presented in this manual are valid and are perfectly suitable for some purposes, more refined and costly methods exist.

**SECTION I**  
**INTRODUCTION**

## PURPOSE OF THE PERFORMANCE TESTING MANUAL

This manual is a guide for pilots and engineers attending the U.S. Naval Test Pilot School (USNTPS). The text presents basic theory, test techniques, and data reduction methods for fixed-wing performance testing. The scope of each flight investigation presented is limited to that necessary to teach the principles of the test and its associated data reduction methods. Students are encouraged to be alert, original, and where possible, make appropriate modifications to general test techniques to obtain a specific result more efficiently. All flight testing described in this manual is conducted to define engine and airframe performance characteristics for use in determining mission suitability and specification compliance where applicable.

## FLIGHT TEST SYLLABUS

The performance flight test syllabus is conducted concurrently with the academic course on airplane performance. A theoretical analysis of fixed-wing engine and airframe performance is presented in the academic syllabus. In addition, flight briefings on test techniques and "Scope of Test" are conducted prior to each flight. Equipped with a knowledge of theory and test techniques, the student plans, conducts, and reports on the assigned test flight.

Each fixed-wing study group will be assigned an airplane for the specific purpose of determining its engine and airframe performance characteristics. The syllabus flights flown by the group to accomplish this evaluation are typical of those conducted at the Naval Air Test Center (NATC) during Navy Preliminary Evaluations (NPE), Board of Inspection and Survey (BIS) Trials, and Performance Technical Evaluations (PTE). Many of the tests require several similar flights to

satisfactorily complete the scope of testing desired. For these tests, the various flights will be divided among the group pilots.

Each student should prepare an informal written report for all performance exercises as soon as possible after each flight in order to record the pilot's freshest impressions and interpretations of the test results. The report must be brief when discussing satisfactory characteristics and as detailed as necessary to describe particularly good and bad characteristics. As a minimum, it should include quantitative data on properly annotated rough plots, qualitative comments, and any airplane peculiarities. Upon completion, the report will be presented either as an informal debrief or as a formal NAFC written report. In any case, each individual report will be incorporated into a project notebook which will provide a running record of flight results for the entire group.

When all performance flights have been completed, the data from each individual report will be used to construct composite graphs of the various performance parameters. These graphs enable the group to evaluate the mission suitability of the airplane and to determine specification compliance. At the end of the performance syllabus, the group compiles data and information gathered during the course into a formal oral report similar to a PTE. Early in the performance syllabus, the various tests must be put in perspective with a view toward their inclusion in this final report.

#### SCHOOL APPROACH TO PERFORMANCE TESTING

The Test Pilot School (TPS) provides a modern airplane for performance testing; and although the airplane is not a new one, the school approach is to assume that it has never been evaluated by the Navy. The test project assumes no

NPE was conducted and that BIS had designated TPS to conduct the secondary phase Aircraft and Engine Performance Trials and Technical Evaluation. The airplane is assumed to be designated for present day use. Stability and control, weapons delivery, and other testing is assumed to be assigned to other divisions of NATC. The TPS student is charged with the responsibility of determining and reporting on the engine and airframe characteristics of the project airplane.

"Mission suitability" is an important phase at NATC, and its importance is reflected in the whole theme of flight testing at TPS. The fact that an airplane meets the requirements of pertinent Military Specifications may be of secondary importance if any performance characteristic degrades the airplane's normal operation in its planned environment. The mission of each airplane will be discussed, then students must conclude whether or not the performance characteristics they evaluate are suitable for the intended mission. This conclusion must be supported by a logical discussion and analysis of quantitative and qualitative observations. Since students possess and can communicate recent fleet experience, their conclusions should be based on a "users" concept.

The performance of an airplane cannot be evaluated for comparison with contract guarantees or other airplanes without accurate quantitative data. Consequently, accurate numbers are indispensable in providing the student with a foundation on which to build a logical and analytical analysis. At TPS every effort is made to test under ideal weather conditions and with all sensitive instrumentation operative, but problems in either of these areas may cause slight errors in the data. If the errors are not the result of improper test techniques, the test results will show the desired data trends which are essential for a satisfactory report. If bad weather, instrumentation failure, or poor pilot test technique result in large



magnitude errors or improper trends, critique on the data may be given; and if warranted, the flight will be re flown. It is important to remember that the primary purpose of the performance testing syllabus at TPS is teaching proper flight test techniques and the basic ideas supporting the methods.

The fact that TPS does not require the data to be exactly correct before it is presented in a report does not excuse the student from conducting a critical examination of the data. It is important for the student to know that errors in the final data do exist and that he should explain the effect of the errors on the results. It has been the experience of the school faculty that some students are too critical of the accuracy of their data because of the scatter they observe in their graphs. Many factors enter into the degree of scatter observed in performance data. High among these factors are inconsistent pilot technique and instrumentation errors. Often a certain amount of scatter is to be expected depending on the particular test under consideration. Again, a degradation in accuracy of results can also be attributed to poor pilot technique, which may in turn be traced to a student who did not properly prepare for a flight or did not follow instructions. Immediate flight planning following a flight brief is imperative to ensure that the proper and necessary preparation is accomplished. The flight brief will encompass the data accuracy which is expected. If these limits are not achieved, an explanation in the individual flight report is warranted.

SECTION II  
PERFORMANCE FLIGHT TEST CONDITIONS  
AND PILOT TECHNIQUES

## PERFORMANCE FLIGHT TEST CONDITIONS AND PILOT TECHNIQUES

### PURPOSE

The purpose of this section is to discuss performance flight test techniques which will be used in conducting the tests outlined in this manual. A demonstration flight, discussed in Section III, will be given to each student by a TPS flight instructor.

### DISCUSSION AND THEORY

#### General

There are three basic test conditions under which a pilot will operate an airplane while conducting performance testing. Each test condition requires special flight techniques and utilizes different primary flight instruments for pilot reference. These conditions are stable equilibrium, unstable equilibrium, and nonequilibrium. Equilibrium test conditions are present during tests in which the airplane is stabilized at a constant airspeed and altitude. A stable equilibrium condition is a condition in which the airplane, if disturbed, will return to its initial condition. An unstable equilibrium test point is a point from which the airplane, if disturbed, will continue to diverge. A nonequilibrium test is a test during which there is a change in airspeed and/or altitude.

#### Stable Equilibrium Conditions

A stable equilibrium test point represents a condition at which pressure altitude, thrust (power), and flight path angle are constant and airplane acceleration along each axis is zero. Stable equilibrium data points are obtained in

both level and turning flight when operating at airspeeds greater than the airspeed for minimum drag (stable portion of the thrust or power required curve). The test technique for obtaining stable equilibrium data is to adjust altitude first, power second, and then wait until the airplane stabilizes at the equilibrium flight airspeed. It is important to point out that altitude must be maintained precisely at the desired test level and that thrust/power must not be changed once set. If this technique is followed, a time history of airspeed can be used to determine when the equilibrium data point has been obtained. For most tests, when the airplane has changed less than 2 kt in the preceding 1 minute period, an equilibrium data point can be assumed to have been achieved. Stable equilibrium test conditions are obtained most rapidly by approaching them with excess airspeed. This approach ensures convergence, whereas an accelerating approach may converge only after fuel exhaustion. The flight test technique used in obtaining stable equilibrium conditions is called the constant altitude method in most flight test literature.

The primary parameters for pilot reference when obtaining data points under stable equilibrium conditions are altitude, vertical speed, heading for straight flight, and bank angle for turning flight. There is no substitute for a good visual horizon. In airplanes equipped with automatic flight control systems (AFCS) that incorporate attitude, altitude and heading hold modes, stable equilibrium data points can be obtained by using these modes provided the sensitivity of the AFCS is adequate for the test. In straight flight, stable equilibrium conditions can be achieved by using altitude and heading hold modes. In turning flight, stable equilibrium conditions can be achieved by using altitude and attitude hold modes.

### Unstable Equilibrium Conditions

Unstable equilibrium data points are more difficult to obtain but can be obtained rapidly if proper technique is employed. For the unstable equilibrium data points, indicated airspeed is held constant. Altitude, engine RPM, and bank angle may be adjusted as required by the test being conducted. Since the indicated airspeed must remain constant, it becomes apparent that good test techniques will assist in obtaining good unstable equilibrium data points. Unstable equilibrium data points are associated with the unstable portion of the thrust or power required curve. To obtain data points under these conditions, the desired test airspeed is established first, then the throttle is adjusted to climb or descend to the desired test altitude. The vertical speed indicator is an important instrument in achieving equilibrium conditions. With throttle set, the vertical speed is stabilized while maintaining the desired test airspeed. A throttle correction is made and the stabilized new vertical speed is observed. The approximate RPM required for level flight can be determined by correlating the values. For example, while attempting to obtain a level flight data point at 135 KIAS, it is determined that 88% RPM produces an 800 fpm climb in the vicinity of the desired test altitude and an 80% RPM produces a 200 fpm descent. The test pilot can determine that 1% RPM change represents a 125 fpm change in vertical speed. By adjusting throttle to 81.6%, he should achieve equilibrium level flight conditions. Normal pilot technique usually puts one within 1% or 2% of the proper RPM, but this averaging technique then applies. A variation of this technique must be used in turning flight when the throttle is set at MIL and cannot be used as the adjustable variable. In this case, bank angle (or load factor) may be related to vertical speed in the same manner

that RPM was related to vertical speed in the straight flight conditions. The flight test technique used in obtaining unstable equilibrium conditions is called the constant airspeed method in most flight test literature.

The primary parameters for pilot reference when obtaining data points under unstable equilibrium conditions are airspeed, vertical speed, heading for straight flight, and bank angle for turning flight. In tests in which rate of climb can be corrected to thrust or power required, it will not be necessary to achieve equilibrium at zero vertical speed. A small altitude change over a small time period (usually 5 min) can be used to correct the test results to level conditions. In other tests, it will be necessary to achieve zero vertical speed. A small amount of practice usually results in satisfactory ability to obtain zero vertical speed at a desired test altitude in less time than it takes to determine an average rate-of-climb correction. It should also be recognized that the constant airspeed techniques can be used to good advantage by a proficient test pilot when obtaining test data under stable equilibrium conditions. Normally, automatic flight control systems offer little advantage over manual control in obtaining unstable equilibrium data points. If control stick steering is available, this mode can be useful if the unstable region of the thrust required curve is also an unstable angle of attack region of flight. An airplane exhibits positive angle of attack stability if it initially tends to return to the same angle of attack after a sudden disturbance. If the airplane appears to exhibit positive or neutral angle of attack stability, manual control will probably be preferable. Altitude hold mode used in conjunction with throttle adjustment will seldom give satisfactory results in unstable thrust required flight conditions.

### Nonequilibrium Test Points

Nonequilibrium test points are usually the most difficult to obtain. They preclude the pilot's having stable conditions or being able to trim to maintain constant conditions. The pilot does, however, have some schedule which he can follow and which he can use to assist him in correcting to achieve a satisfactory flight path or flight test condition. Some nonequilibrium tests such as acceleration runs are performed at a desired constant altitude. Others such as climbs and descents are performed according to a desired airspeed schedule.

The primary reference parameters for nonequilibrium tests will be dictated by the specific test being performed. Automatic flight control system equipment can be a great aid in obtaining nonequilibrium condition data. The degree to which it can be employed will depend upon the specific test and the ability of the modes to perform their design functions. Good heading hold and altitude hold modes can be extremely valuable in obtaining level acceleration test data. If the altitude mode is unable to maintain a relatively constant altitude ( $\pm 100$  ft), use of control stick steering may be superior to manual control since this mode will provide automatic longitudinal retrim. Climb and descent tests can be performed using Mach or Indicated Airspeed hold modes if they are available and if they have sufficiently high gain to maintain the desired schedule accuracy ( $\pm 5$  kt or  $\pm 0.01M$ ). If these modes are not available, use of control stick steering will at least provide smooth automatic retrim and relieve the pilot of that task.

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SECTION III  
PERFORMANCE DEMONSTRATION FLIGHT

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## PERFORMANCE DEMONSTRATION FLIGHT

### INTRODUCTION

The performance demonstration flight conducted at TPS was established to introduce the student to the fixed-wing performance testing syllabus. The flight will usually be conducted in a different airplane than the student will utilize during the performance phase of the syllabus. Prior to the demonstration flight, the student must have completed the handbook exam, cockpit checkout, and at least one familiarization flight. In addition to test techniques, the student will be instructed in the proper use of instrumentation and the importance of knowing the correct preflight procedures and preparation required prior to each test flight.

### PURPOSE OF DEMONSTRATION FLIGHT

The primary purpose of the fixed-wing performance demonstration flight is to acquaint the student with the test techniques and method of tests utilized at TPS. The knowledge gained during this flight will enable the student to practice the correct techniques during succeeding familiarization and practice flights. The purpose of the demonstration flight is to teach test techniques and not fixed-wing performance theory. The importance of the various tests and the significance of the test results relating to mission suitability will be emphasized. Methods of test and pilot techniques will be demonstrated for the following tests:

- a. Takeoff and climb performance.
- b. Level flight performance (thrust required).
- c. Constant altitude acceleration runs (specific excess power).
- d. Level turning performance and buffet boundary characteristics (thrust and lift boundaries).

- e. Stall airspeed determination.
- f. Pitot - static system calibration (tower fly-by).

#### FLIGHT FORMAT

- a. The student will be briefed on the proper operation and utilization of photopanel and phenomenon lights and the proper flight test techniques.
- b. The importance of altimeter settings (29.92), sufficient leader on photopanel film and frame counter correlation will be discussed.
- c. The flight will be conducted simulating a fully-instrumented airplane (student in front seat).

#### Takeoff Techniques (Nonequilibrium)

The demonstration pilot will emphasize the importance of initial line-up and trim settings in takeoff performance testing. The use of brakes can seriously affect the takeoff distance and acceleration performance of the airplane. The technique to obtain good nose wheel line-up is uniform, simultaneous braking while lining up prior to takeoff. This will ensure that the nose wheel is not cocked at brake release. If nose wheel steering is available it may be used. Photopanel data are normally taken during the takeoff run. The photopanel is started prior to brake release, usually after takeoff thrust has been obtained. Brakes are released at a specified photopanel counter number and the phenomenon light is used to mark the film at that time. Takeoff rotation will be at a specified airspeed, gear and flaps will be retracted at a specified airspeed, and the climb schedule will be intercepted at as low an altitude as is feasible for normal operating conditions. During acceleration to initial climb airspeed, rate of climb should not exceed 500 fpm.

### Climb Techniques (Nonequilibrium)

The instructor pilot will demonstrate the transition to the climb schedule, pilot techniques for maintaining schedule, and the proper use of test instrumentation. The climb schedule represents a nonequilibrium flight condition in which the pilot is following a predetermined airspeed vs. altitude schedule. The schedule for the particular airplane will be described during the flight briefing. The demonstration should cover techniques for transitioning from a constant altitude condition to a climb schedule and should include demonstration of an interrupted climb. The instructor pilot should emphasize the requirement for making smooth, early corrections during the climb phase. In demonstrating an interrupted climb, it is important to emphasize that the conditions at the time the climb is interrupted must be duplicated exactly when re-establishing the climb. At least 1,000 ft of the climb schedule should overlap.

### Level Flight Performance

Stable equilibrium data points (points at airspeeds greater than the airspeed for minimum drag) will be demonstrated in level flight at the altitude designated for the particular demonstration airplane. If the test airplane is equipped with a pointer counter altimeter, it is essential that even 1,000 ft altitudes not be used in the demonstration because of the inherent altimeter instrument friction characteristics about the 12 o'clock position. The proper technique is to set the appropriate power or thrust setting first and allow the airspeed to decelerate and stabilize. After setting the thrust and stabilizing the altitude, the pilot's stopwatch should be started to determine airspeed acceleration or deceleration over a 1 min interval. When accelerations or decelerations are less than 2 kt/min, equilibrium

conditions can be assumed to have been achieved and the photopanel and/or cockpit data can be recorded. It is essential that the student understand this time history concept in order to minimize test time requirements. This is called the constant altitude method. The primary reference instruments are the altimeter, vertical speed indicator, and RMI for heading.

Unstable equilibrium data points (points at airspeeds less than the airspeed for minimum drag) will be demonstrated in level flight at the same altitude as the stable equilibrium points. This technique is called the constant airspeed method. The instructor pilot will emphasize that the airspeed must be maintained constant for unstable equilibrium data points. Pilot techniques for determining thrust or power required for level flight under these conditions will be demonstrated.

Trim the airplane at the desired airspeed and set a thrust (RPM) which results in a stabilized climb of less than 1,000 fpm at the trim airspeed. Reduce the thrust (RPM) to obtain a stabilized descent at the same trim airspeed. After determining the rate of change of vertical speed with RPM, an estimate of RPM required for level flight can be made. A throttle correction to position the airplane at the desired test altitude will then be made as the airplane approaches within +100 ft of the specified test altitude. After the airplane has maintained level, stabilized flight for approximately 60 sec, data can be recorded. An altitude gain or loss of 25 ft in 60 sec is considered acceptable. This method is applicable in airplanes which have no significant change in gross weight while obtaining unstable equilibrium data test points. The primary instruments for this technique are the airspeed indicator, the vertical speed indicator, the altimeter, and the RMI for heading.

### Acceleration Run (Nonequilibrium)

The constant altitude acceleration run is another example of a nonequilibrium flight test condition. In this case, a constant altitude will be maintained, but the airplane will not be in equilibrium condition (except at  $V_{mrt}$ ) because it is accelerating along the longitudinal axis. Level flight acceleration runs may be performed by constantly trimming or by attempting to hold all forces with the control stick. The trimming method is recommended; however, exact trim conditions should not be attempted. A small amount of pull or push force should be maintained throughout the run. The reason for this is to avoid the control system break-out forces about the force trim points. This condition may result in a force reversal from push to pull during the run which generally produces an oscillatory motion during the test. If the airplane has large excess thrust capabilities or slow engine spool-up time, it will be necessary to commence the acceleration run from climbing flight conditions to conduct the entire acceleration run at stable engine operating conditions. The instructor pilot should emphasize the importance of making smooth corrections and discuss the use of speed brakes to facilitate the start.

### Level Turning Performance

The techniques for determining level turning performance (both stable and unstable equilibrium data points) will be demonstrated at the designated test altitude. The instructor pilot should emphasize that  $V_{mrt}$  is a point in turning performance (1.0g point). For the stable equilibrium points, set the thrust/power at MIL, establish the airplane in a constant bank, stabilize the altitude with normal acceleration and allow the airspeed to stabilize. Stabilization can be assumed when

the airspeed, normal acceleration, and altitude are stabilized for 5 sec within 1,000 ft of test altitude. The instructor should emphasize the importance of trimming the airplane longitudinally to maintain the desired bank angle and normal acceleration, and of using the visual horizon. For unstable equilibrium points, advance the thrust to MIL and adjust normal acceleration in a turn to maintain desired airspeed. Once the airspeed is stabilized by applying the appropriate normal acceleration, the bank angle required for level flight can be established. The instructor should demonstrate the relationship of bank angle to vertical speed, and of normal acceleration to airspeed. The stabilization criteria is the same as for stable equilibrium conditions. This technique is difficult to master and will require considerable practice for proficiency.

#### Buffet Boundary (Nonequilibrium)

The technique for determining the buffet boundary (lift limit) characteristics of an airplane will be demonstrated at the same test altitude as level turning performance. The technique is called the constant Mach number/airspeed wind-up turn and is another example of a nonequilibrium test condition. Establish the airplane at a specified Mach number/airspeed and smoothly increase normal acceleration. In order to minimize scatter in the data, a constant thrust setting is preferred. For load factors above the maximum level flight sustained g-available, use bank angle (airplane weight vector) to supplement thrust in order to maintain a constant Mach number/airspeed during the maneuver. Satisfactory data can be obtained within 1,000 ft of test altitude. The instructor should emphasize the importance of increasing the normal acceleration at 0.5g per sec or less to avoid dynamic errors in the data. The instructor should point out and discuss the applicable buffet levels.

### Stall Speed Determination (Nonequilibrium)

In this test, the pilot is following a deceleration schedule of 1/2 kt/sec or less. The configuration of the airplane or engine limitations will determine the throttle setting. The primary instruments in this particular maneuver will be the airspeed indicator and the clock. Stalls at higher rates of deceleration (5 to 10 kt/sec) will also be demonstrated to illustrate the effects of nonsteady flow conditions. The instructor will also discuss or demonstrate the effects of trim setting on stall speed data.

### Phototheodolite Tower Fly-by

The instructor will familiarize the student with the local course (currently in use), and the student may perform practice pitot static source calibration runs if time and fuel permit.

### Student Participation

The student should be permitted to try each basic technique, but the instructor pilot must maintain the pace of the flight in order to complete all demonstration requirements. It is important to emphasize that the purpose of this flight is to introduce the three basic flight test conditions and the techniques used by test pilots to collect performance data. Instructors will answer questions, but avoid specific details of theory which may not yet have been introduced in the academic syllabus. Further detailed briefs and practice flights are included in the syllabus.

SECTION IV  
PITOT STATIC SYSTEM TESTING



## SECTION IV

### NOTATIONS INTRODUCED IN THIS SECTION

$\lambda$	- lag error constant
$P_s$	- static pressure
$P_a$	- ambient pressure
$P_T$	- free stream total pressure
$\rho_a$	- ambient density
$\rho_{SL}$	- sea level density
$\gamma$	- ratio of specific heats
$q_c$	- impact pressure
$q_{c_i}$	- indicated impact pressure
$V_i$	- indicated airspeed
$V_c$	- calibrated airspeed
$\Delta P$	- difference in static source pressure and ambient pressure, $P_s - P_a$
$\frac{\Delta P}{q_{c_i}}$	- static source error coefficient
Subscript w	- signifies data corrected to standard airplane weight
$H_p$	- actual pressure altitude of the test airplane
$H_{p_o}$	- observed pressure altitude of the test airplane (cockpit or photopanel recording)
$H_{p \text{ theodolite}}$	- pressure altitude of theodolite
$h_{tape}$	- tape height increment of test airplane above MSL
$h_p$	- pressure height of the test airplane above MSL

$W$	- airplane gross weight
$W_s$	- standard airplane gross weight
$\Delta H_{p_{ic}}$	- altimeter instrument error correction
$\Delta V_{ic}$	- airspeed indicator instrument error correction
$\Delta V_{pos}$	- airspeed position error correction
$\Delta H_{p_{pos}}$	- altimeter position error correction

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## PITOT STATIC SYSTEM TESTING

### INTRODUCTION

The altimeter, airspeed, and machmeter indicators are three universal flight instruments which require total and/or static pressure inputs to function. It is generally assumed that a properly designed pitot tube can provide error free total pressure measurements. However, it is difficult to accurately measure ambient static pressure. The error which results from the difference between the actual ambient pressure and the static pressure measured at the aircraft static pressure source is called position error. In addition, any pitot/static pressure system is subject to errors associated with both the mechanical instrument and the tubing which connects the instruments to the static pressure source. This section will briefly discuss these errors and the various test techniques used for their determination. The pertinent requirements of military specification MIL-F-6115A will be presented as well as a brief discussion of the recovery factor for an outside air temperature (OAT) probe.

### PURPOSE

The purpose of this test is to evaluate the altimeter and airspeed position errors of an airplane using the altimeter depression method. In addition, the OAT probe recovery factor will be determined.

## DISCUSSION AND THEORY

### General

The errors associated with any pitot/static pressure system may be generally classified as mechanical, operational, and installation errors. Mechanical errors primarily include instrument errors. Operational errors include readability and incorrect barometric settings. Installation errors primarily consist of lag errors and position errors.

### 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 instrument errors prior to an inflight determination of position errors. 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 often advisable. The problem arises from the fact that two calibrated instruments, placed side by side with their reading corrected by use of calibration charts, do not always provide the same resultant calibrated altitude. Tests such as the "Tower Fly-By" or the "Trailing Bomb" for altimeter calibration require an altimeter to provide a base line (reference) pressure altitude. These tests normally require that the reference altimeter be placed next to the service system altimeter prior to and after each flight. Each altimeter reading should be recorded, and if after calibration corrections have been applied there still exists a discrepancy between the two readings, this discrepancy should be incorporated in the data reduction. This correction is not made at TPS.

### Lag Error

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

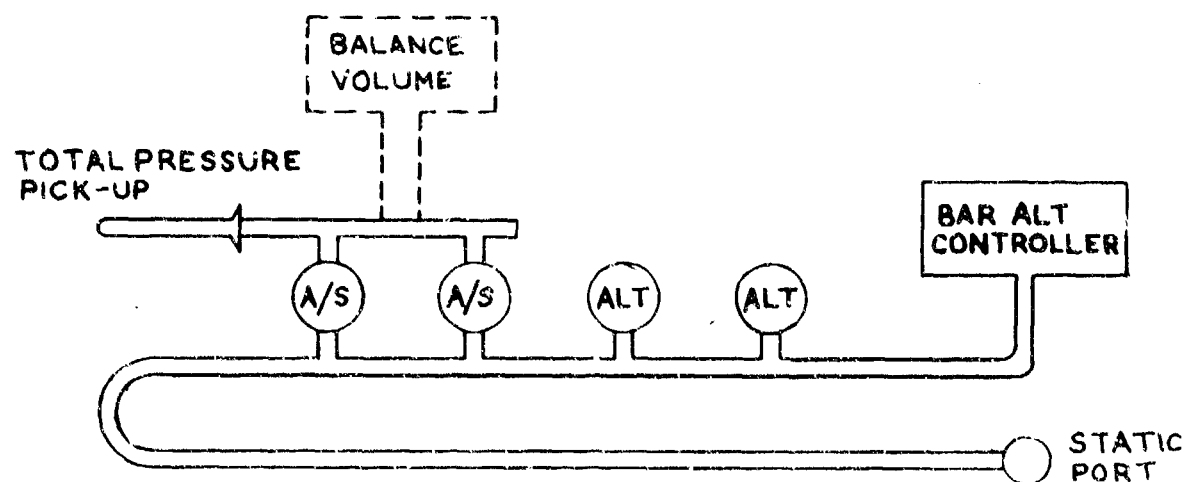
- a. Pressure drop in the tubing caused by viscous friction.
- b. Inertia of the air mass in the tubing.
- c. Instrument inertia and viscous and kinetic friction.
- d. The finite speed of pressure propagation; i.e., acoustic lag.

Over a small pressure range, the pressure lag is small and can be determined as a constant ( $\lambda$ ). 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 Test Center (NATC), 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 airspeed. This is particularly important for airplanes used as calibrated pacers.

### Lag Constant Test

The pitot and static pressure systems of a given airplane 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 of the tubing,

when added together for each pressure system, produce a volume mismatch between systems. Figure 1 illustrates a configuration in which 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 1, the two systems would require different periods of time to stabilize at the new pressure level and a momentary error (lag error) in indicated airspeed and indicated altitude would result. The time required for each system to stabilize can be related to the system time constant.



<u>SYSTEM</u>	<u>LENGTH OF 3/16"</u> <u>ID TUBE</u>	<u>TOTAL VOLUME OF</u> <u>INSTRUMENT CASES</u>
Static	18 ft	$370 \times 10^{-4}$ cu ft
Pitot	6 ft	$20 \times 10^{-4}$ cu ft

Figure 1  
Analysis of Pitot and Static Systems Construction

If an increment of pressure is instantaneously applied to a system, the lag constant ( $\lambda$ ) represents the period of time required for the pressure of the system to increase by an amount equal to 63.2 percent of the applied pressure increment. This is shown graphically in Figure 2(a). A test to determine the altimeter and airspeed indicator lag constants can be accomplished on the ground by applying a suction sufficient to develop a  $\Delta H_p$  equal to 500 ft or an indicated airspeed of 100 kt. Removal of the suction and timing the altitude drop to 184 ft, or the airspeed drop to 37 kt, results in the determination of  $\lambda_s$ , the static pressure lag constant, as shown in Figures 2(b) and 2(c). If a positive pressure is applied to the total pressure pickup (drain holes closed) to produce a 100 kt indication, the total pressure system lag constant ( $\lambda_T$ ) can be determined by measuring the time that is required for the airspeed indicator to drop to 37 kt when the pressure is removed. Generally, the  $\lambda_T$  will be much smaller than the  $\lambda_s$  because of the small volume of the airspeed instrument case.

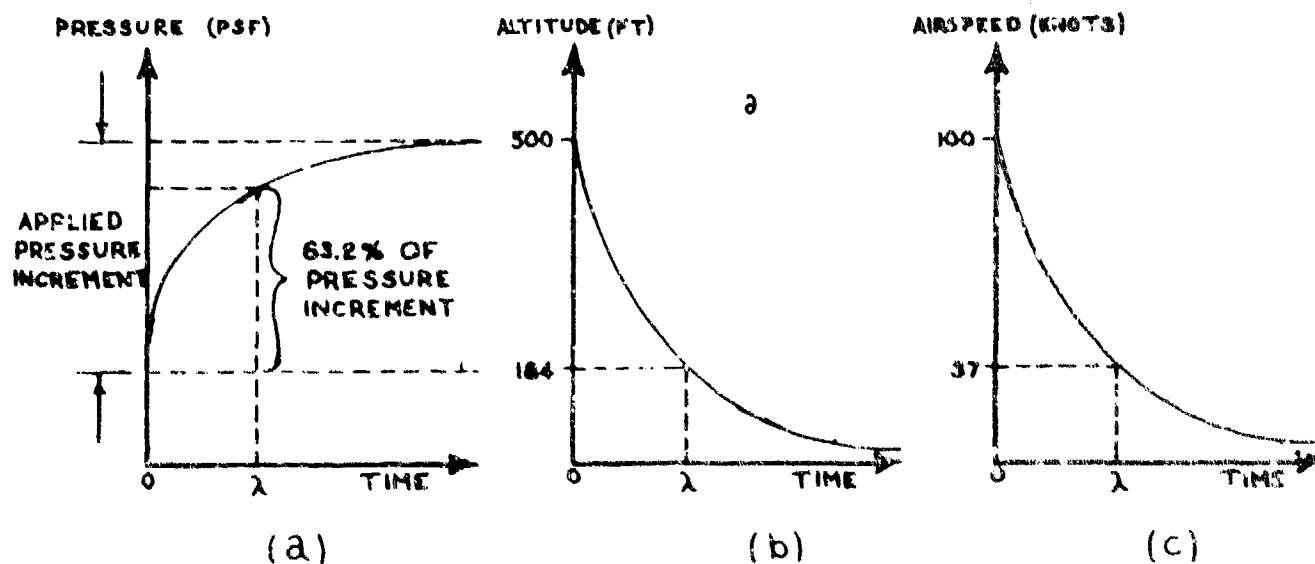


Figure 2  
Determination of Pitot/Static System Lag Constant ( $\lambda$ )



### System Balancing

The practical approach to lag error testing is to determine if a serious lag error does exist and to eliminate it where possible. To test for airspeed system balance, a small increment of pressure (0.5 psf) 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 (see Figure 1) will generally provide satisfactory airspeed indications but will not help the lag in the altitude indicator as this lag is primarily caused by 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.

### 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 an airplane these pressure errors are usually present only at the static pressure sensor and the total pressure measurement (pitot tube) is assumed to be error-free. The limits for this assumption are discussed in NACA Report 919.

Based on the assumption of an error-free total pressure measurement, airspeed and altimeter systems may be calibrated by either the altimeter depression method or by the pace method. Since the altimeter depression methods are generally more economical in terms of flight test time and do not require a calibrated pace

airplane, they are most often used. Differences between the pressure measured at the static source and the true ambient pressure result in an early determined static source position error. The effects on airspeed of errors in static pressure measurement may be best understood by examining the following equations for calibrated and indicated airspeed.

$$V_C = \left\{ \frac{2\gamma P_{SSL}}{(\gamma - 1) \rho_{SSL}} \left[ \left( \frac{P_T - P_A}{P_{SSL}} + 1 \right)^{\frac{\gamma-1}{\gamma}} - 1 \right] \right\}^{1/2}$$

$$V_i = \left\{ \frac{2\gamma P_{SSL}}{(\gamma - 1) \rho_{SSL}} \left[ \left( \frac{P_T - P_S}{P_{SSL}} + 1 \right)^{\frac{\gamma-1}{\gamma}} - 1 \right] \right\}^{1/2}$$

These equations show that a static pressure ( $P_S$ ) value that is lower than the true ambient pressure ( $P_A$ ) will result in an indicated airspeed ( $V_i$ ) that is greater than the calibrated airspeed ( $V_C$ ). If the static pressure is higher than ambient pressure, the reverse is true. In each case, the total pressure ( $P_T$ ) measurement is assumed to be error-free.

Other altimeter depression methods which may be used are the tower fly-by, the use of the trailing cone or trailing bomb, the stadiametric camera fly-by, and the use of the radar altimeter. The fly-by method is typical of the altimeter depression methods and is relatively simple to perform and analyze. It is the method now being used at TPS.

The theodolite fly-by test consists of a series of stabilized level flight runs at several airspeeds over a fixed course. For the Point-No-Point range, theodolite film readings of the actual altitude ( $h_{\text{tape}}$ ) of the test airplane for each run are obtained using a contraves phototheodolite. For the Webster Field range, actual airplane

altitude is determined using a special camera which produces polaroid pictures with a grid superimposed over the image. The altimeter position error is determined by comparing the actual pressure altitude of the test airplane (tapeline altitude corrected for temperature variations from standard) with the indicated pressure altitude of the test airplane. The airspeed position error (caused by the static source error) may be calculated directly from the altimeter position error with the assumption of no total pressure error. For airspeeds less than approximately 0.6M, the airspeed position error is mainly a function of angle of attack, and the static source position error coefficient ( $\frac{\Delta P}{q_{C_i}}$ ) determined in this test will be valid for all altitudes at standard gross weight. From the static source position error coefficient determined for the standard weight, the airspeed position error and the altimeter error may be calculated for other gross weights. For constant indicated airspeed, hence constant angle of attack, the altimeter position error increases with increasing altitude. This altimeter position error variation can be calculated and altimeter position error corrections for all altitudes and weights can be extrapolated from the data (static source position error coefficients) obtained at one altitude.

Extrapolation of airspeed and altimeter position error corrections based on sea level data may not yield acceptable results in all cases. Extrapolation requires certain simplifying assumptions, e.g., negligible total pressure source error and Reynold's number effects. For large extrapolations (sea level to 40,000 ft) these assumptions may fail, and the position error correction for a turbojet airplane determined from sea level data obtained must be verified by additional tests at high altitude using a calibrated pacer airplane.

### Military Specifications

Military Specification MIL-I-6115A, Amendment 3, of 31 December 1960 states the allowable limits of the altimeter and airspeed position error:

- a. Airspeed error - +4 kt at all airspeeds.
- b. Altitude error - 25 ft per 100 KIAS.

DOD Joint Chief of Staff memo of 2 July 1963 stipulates that for all military aircraft "THE MODE C ALTITUDE INFORMATION SHALL BE OBTAINED FROM AN ALTIMETER WITH A MAXIMUM ALTIMETER SYSTEM ERROR OF PLUS OR MINUS 250 FEET AT ALL SPEEDS AND OPERATING ALTITUDES WHILE IN THE MISSION OF FLIGHTS. ALTIMETER ACCURACIES IN OTHER PHASES OF FLIGHT, i.e., LANDING, TAKEOFF, INSTRUMENT APPROACHES, etc., SHALL MEET PRESENT MILITARY/FAA TSO SPECIFICATION STANDARDS."

### Probe Recovery Factor

The OAT is a necessary flight test parameter and is used in conjunction with airspeed and altitude to determine true airspeed. The OAT probe recovery factor can be determined while conducting airspeed and altimeter position error tests. The indication of the OAT gauge is a function of the ambient air temperature, the recovery factor of the OAT system's probe, and the airplane Mach number. The tower fly-by test is flown over a range of Mach numbers wide enough to determine the recovery factor (K) from the OAT readings observed during the test. Since the recovery factor is a measure of the efficiency of the probe design in preventing heat loss by radiation, it should become obvious that a recovery factor greater than 1.0 cannot be obtained. Typical recovery factors range between 0.85 to 0.98. Recovery factors for special flight test instrumentation should be at least 0.95.

### Data Recording

a. This test requires three observers in the ground station. Two observers are required to operate the camera equipment while a third observer calls "MARK" when the test airplane is over the Point-No-Point lighthouse or abeam-the-Webster-Tower. Technical Support Division (TSD) will supply the camera operators. TPS is required to supply the third observer.

b. Photopanel data are generally more accurate than cockpit observed data and will be used when available.

c. Sequence of Test Data: In configuration CR, commence runs at a mid-range airspeed and work up to maximum airspeed. Then work down from the initial airspeed to the slowest airspeed desired or allowed. For configuration PA, begin at the maximum limit airspeed and work down to minimum airspeed. Correlate runs with other airplanes in the pattern to optimize safety and utilization of the ranges.

### Preflight Procedures

a. Required Data:

1. Pilot - Kneeboard data card (Figure 3).

$V_o$ ,  $H_{p_o}$  at "MARK" called by the range observer.

$CAT_o$  (record for configuration CR points only).

Configuration.

Fuel remaining or fuel used.

Photopanel counter number.

Photopanel (if available).

Energize 10 sec prior to data point.

Turn OFF while climbing downwind.

2. Range Observer (prepare data card for each airplane).

Run No.

Theodolite end frame number for each run.

Time.

Ambient air temperature at beginning and end of tower runs  
(information only).

b. Know the following:

Airplane configuration details.

Airspeed ranges for each configuration.

Airplane restrictions.

Traffic pattern.

c. Prepare data card (see Figure 3).

d. Inspect photopanel for loading and operation.

e. Note location of pitot tube and static sources in order to reference the  
applicable Military Specification and to discuss the test results properly.

WEATHER <u>SAMPLE DATA CARD</u>				CARD NO.	
<u>AIR-SPEED CALIBRATION</u>				PTR DIS	
AIRPLANE TYPE	REG. NO.	TIME T.O.	DATE		
<u>UPF-7</u>	<u>117711</u>	TIME LAND	<u>7 Oct 78</u>		
PILOT	CONDITION	CLEAR SKY	WIND	T.O. GROSS WEIGHT	
<u>Derong</u>	<u>NORMAL Fighter</u>				
EXTERNAL CONFIGURATION					

No	Alt	V <sub>SOI</sub>	V <sub>2</sub>	H <sub>po</sub>	OATs	Fuel	CTA	Remarks
1	CR	175						
2		200						
3		225						
4		250						
5		275						
6		V <sub>MAX</sub>						
7		150						
8		125						
9	✓	100						
10	PA	125						
11		115						
12		105						
13		95						
14		90						
15		85						
16	↓	80						

FLIGHT DATA PRINC. DATE 1900 3

Figure 3  
Sample Data Cards

### Flight Procedures for Point-No-Point Range

a. Inform ground control that you will be operating in R4005, 1,500 feet to the surface, in the vicinity of POINT LOOKOUT. Closest point of approach to the field will be 7 miles.

b. After takeoff obtain clearance from Approach Control (354.8/281.8 mcs) for clearance into R4005. Approach Control will shift you to project frequency (385.2/330.4). Contact TANGO control (Point-No-Point Tower) on assigned frequency and enter pattern on a 1,000-1,500 ft downwind leg (see Figure 4). Fly the pattern depicted, descending to target altitude (500 ft) when established on the inbound track. Call SIDE NUMBER, RUN NUMBER, and TARGET AIRSPEED when inbound near POINT LOOKOUT. Call a 10 SEC STANDBY prior to passing over the lighthouse. The tower observer should call a "MARK" when your airplane is directly over the lighthouse and will record your run number for correlation of data.

c. Make the runs at approximately the target altitude in airspeed increments to cover the range of airspeeds to be investigated in the time allowed. To obtain an accurate measurement of height of the airplane above the lighthouse, it is essential that the flight path be along a straight track over the lighthouse. The target altitude for each run should be 500 ft AGL. For each run, allow sufficient straight-away to ensure that airspeed and altitude are stabilized 10 sec before passing over the lighthouse. Stabilized flight and good trim are essential to obtain satisfactory data in this test.

d. After passing the lighthouse, be alert for traffic. Make a climbing left turn and maintain 1,000-1,500 ft AGL downwind. Do not descend below 1,000 ft until clear of land.



e. For each run record the observed altitude ( $H_{p_o}$ ) and observed airspeed ( $V_o$ ) as the airplane passes over the lighthouse. When the photopanel is used, it should be energized about 10 sec before passing the lighthouse and switched OFF when over the lighthouse. This procedure results in a record of degree of stabilization in the final approach with the last frame of this film burst being the actual data used. Time and OAT may be recorded as soon as airspeed has stabilized on the inbound leg. Record film counter number, configuration, and fuel on the downwind leg.

f. Photopanel data is desirable for this test; however, flight reports will be submitted on kneepad data in case of photopanel malfunction.



### Flight Procedures for Webster Field Range

a. Inform ground control that you will be operating in R4005, 1,500 ft to the surface, in the vicinity of WEBSTER FIELD. Closest point of approach to Patuxent will be 7 miles.

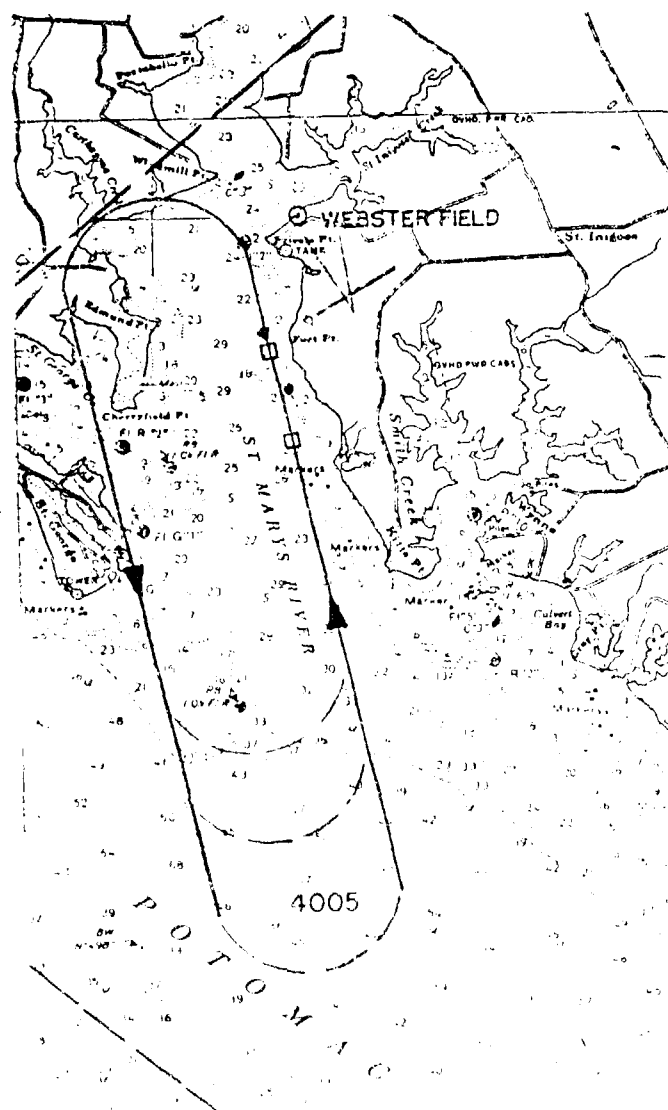
b. After takeoff, obtain clearance from Approach Control (354.8/281.8 mcs) for clearance into R4005. Approach Control will shift you to Webster Tower Frequency (358.0). Enter pattern on a 1,000-1,500 ft downwind leg. Fly the pattern so as to remain above 1,000 ft over land, altitude not less than 150 ft AGL when established on the inbound track (see Figure 5). Call SIDE NUMBER, RUN NUMBER, and TARGET AIRSPEED when inbound abeam St. Georges Island. Call a 10 SEC STANDBY prior to passing Webster Tower. The tower observer should call a "MARK" when your airplane is directly abeam the tower and will record your run number for correlation of data.

c. The target altitude for each run should be 200 ft AGL. Make the runs in airspeed increments to cover the range of airspeeds to be investigated in the time allowed. For each run, allow sufficient straight-away to ensure that airspeed and altitude are stabilized 10 sec before passing abeam the tower. Stabilized flight and good trim are essential to obtain satisfactory data in this test.

d. After passing the Tower be alert for traffic. Make a climbing left turn and maintain 1,000-1,500 ft AGL downwind. Do not descend below 1,000 ft until clear of land.

e. For each run, record the observed altitude ( $H_{p_o}$ ) and observed airspeed ( $V_o$ ) as "MARK" is called. When the photopanel is used, it should be energized about 10 sec before passing abeam the tower and switched OFF when "MARK" is called. This procedure results in a record of degree of stabilization in the final approach

f. Photopanel data are desirable for this test; however, flight reports will be submitted on kneepad data in case of photopanel malfunction.



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### Tower Observer Procedures

- a. Ensure a sensitive altimeter is available.
- b. Arrange own transportation to the ground stations. Dress warmly in cold weather -- both towers are well ventilated.
- c. Monitor control frequency. Make transmissions as required for coordination, call "MARK" as each data point is taken, and aid in recording data. Record the pressure altitude of the theodolite ( $H_{P_{theo}}$ ) for each run, using the sensitive altimeter in the tower. Obtain ambient air temperature (shaded area).

### Postflight Procedures

- a. Run photopanel film out after landing.

## DATA REDUCTION AND PRESENTATION

### General

Airspeed and altimeter position error corrections for the standard gross weight will be determined for both configurations tested. The OAT probe recovery factor will also be determined. The first and last data points must be worked by hand.

### Point-No-Point Procedures

From the theodolite observer's data sheet record the ambient temperature in the tower. From the theodolite film (see Figure 6) determine:

- a. Frame counter number.
- b. The elevation angle of the cross hairs.

c. The error in tracking the airplane. This error in feet may be determined directly from the film by using the airplane image to scale the film. When the cross hairs are below the airplane, the error is positive.

From the theodolite observer's data sheet and the information recorded from the theodolite film calculate:

a. Theodolite cross hair tapeline height above the tower; use the cross hair elevation angle and table 1.

b. Airplane altimeter tapeline height above the tower; cross hair height plus tracking error (Figure 7).

c.  $T_a (^{\circ}\text{F})$ ; use OAT data obtained at a low Mach number to approximate this value.

$$d. H_p(\text{tower}) = H_{p0} + \Delta H_{pic}$$

e. Airplane pressure height above mean sea level

$$h_{pc\text{airplane}} = h_{ptower} + \frac{T_{std}}{T_{test}} \left[ \overbrace{h_{tapeline\text{airplane}} - h_{tapeline\text{tower}}}^{\Delta h_{tape\text{ above tower}}} \right]$$

Where  $T_{std}$  is the standard day temperature at the tower ( $T_{std}$  and  $T_{test}$  are in deg Rankine).

From kneeboard or photopanel data record the following information:

- Altitude:  $H_{p0}$
- Airspeed:  $V_o$
- Fuel used or Fuel remaining
- Outside air temperature:  $OAT_o$

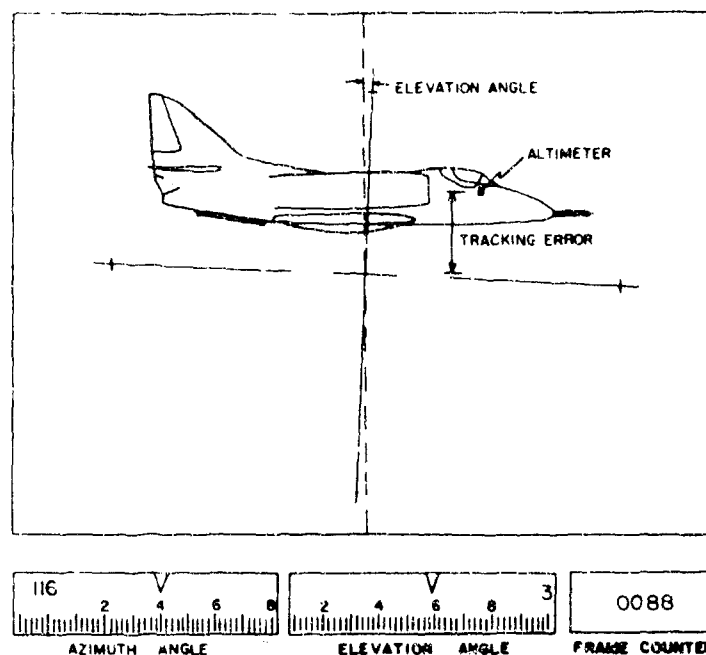


Figure 6  
Point-No-Point Theodolite Film Data

Table 1

Theodolite Elevation Angle vs. Tapeline Altitude Above

$\theta$ ELEV. ANGLE (DEG)	TAPE- LINE ALT (FT)	$\theta$ ELEV. ANGLE (DEG)	TAPE- LINE ALT (FT)	$\theta$ ELEV. ANGLE (DEG)	TAPE- LINE ALT (FT)	$\theta$ ELEV. ANGLE (DEG)	TAPE- LINE ALT (FT)	$\theta$ ELEV. ANGLE (DEG)	TAPE- LINE ALT (FT)
0	0	1.0	179	2.0	359	3.0	538	4.0	718
.1	18	1.1	197	2.1	376	3.1	556	4.1	736
.2	36	1.2	215	2.2	394	3.2	574	4.2	754
.3	54	1.3	233	2.3	412	3.3	592	4.3	772
.4	72	1.4	251	2.4	430	3.4	610	4.4	790
.5	89	1.5	269	2.5	448	3.5	628	4.5	808
.6	107	1.6	287	2.6	466	3.6	646	4.6	826
.7	125	1.7	305	2.7	484	3.7	664	4.7	844
.8	143	1.8	323	2.8	502	3.8	682	4.8	862
.9	161	1.9	341	2.9	520	3.9	700	4.9	880
1.0	179	2.0	359	3.0	538	4.0	718	5.0	898

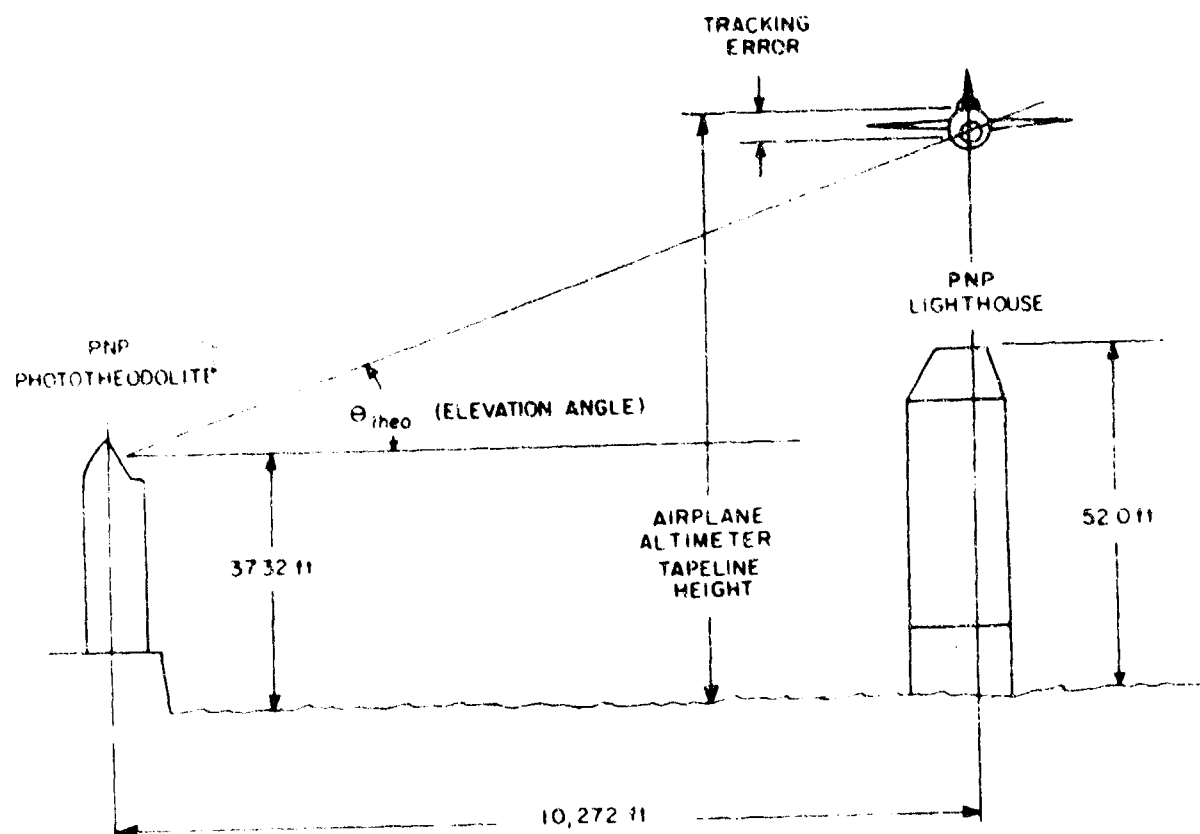


Figure 7  
Point-No-Point Theodolite Survey Data

#### Webster Field Procedures

- a. From the tower observer's data sheet, record the ambient temperature ( $^{\circ}\text{F}$ ) outside the tower.
- b. From the tower polaroid film determine the scale length (inches) of the test airplane and the scale height (inches) of the test airplane above the reference line (Figure 8).
- c. Determine the actual length (ft) of the test airplane from Appendix I. (Airplane Details)



d. Calculate the tapeline height (ft) of the test airplane above the reference line by the expression.

$$h_{\text{tape above reference}} = \left[ \frac{\text{airplane length (ft)}}{\text{scale length (inches)}} \right] \text{scale height}_{\text{above reference (inches)}}$$

e. Calculate the pressure altitude in the tower by

$$h_{\text{Ptower}} = h_{\text{Po}} + \Delta h_{\text{pic}}$$

f. Calculate the calibrated pressure altitude of the test airplane by the expression

$$h_{\text{Pc airplane}} = h_{\text{Ptower}} + \frac{T_{\text{std}}}{T_{\text{test}}} \left[ \overbrace{h_{\text{tapeline airplane}} - h_{\text{tapeline tower}}}^{\Delta h_{\text{tape above tower}}} \right]$$

where

$T_{\text{std}}$  is the standard day temperature at the tower.

$T_{\text{std}}$  and  $T_{\text{test}}$  are in deg Rankine.

#### Position Error Determination

a.  $h_{\text{Pi}} = h_{\text{Po}} + \Delta h_{\text{pic}}$

b.  $V_i = V_o + \Delta V_{\text{ic}}$

c.  $P_i = f(h_{\text{Pi}})$  (Use tables of P vs.  $H_p$  in Appendix IX.)

d.  $P_A = f(h_{\text{Pc}})$  (Use tables of P vs.  $H_p$  in Appendix IX.)

e.  $\Delta P = P_S - P_A$

f.  $q_{ci} = f(V_i)$  (Use charts of  $q_c$  vs.  $V_c$  in Flight Test Reference Book.)

g. Calculate  $\frac{\Delta P}{q_{ci}}$

h.  $W_t = W_{FULL} - [\text{Fuel used}]$  or  $[W_{EMPTY} + \text{Fuel remaining}]$

i.  $\frac{W_s}{W_t}$  (Use  $W_s$  from Airplane Details in Appendix I.)

j.  $V_{i_w} = V_i \left( \frac{W_s}{W_t} \right)^{1/2}$

Plot the variation of  $P/q_{ci}$  with  $V_{i_w}$  and fair a smooth curve through the data.

Label this curve for the standard weight.

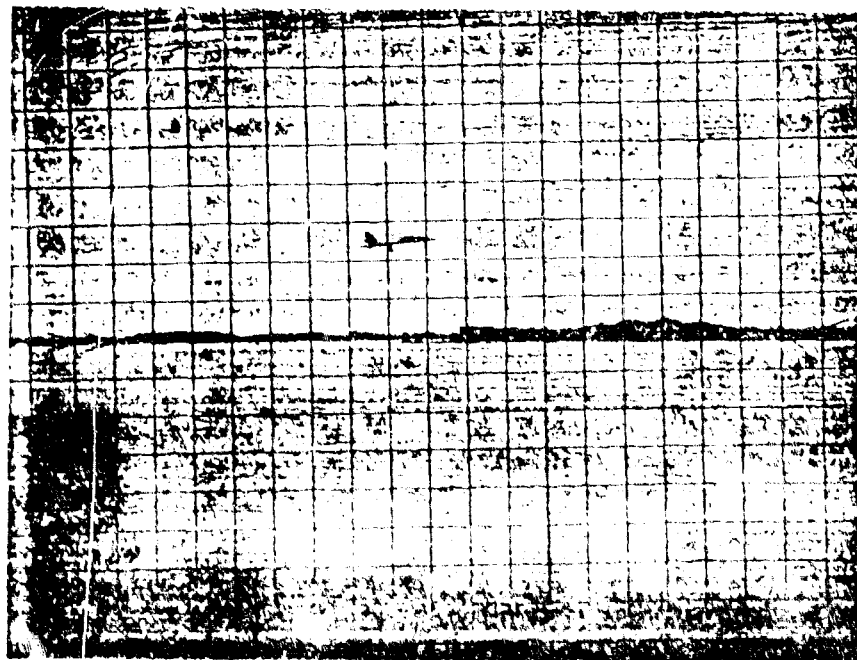


Figure 8  
Polaroid Film Data from Webster Field

The military specification requires that position error be determined at sea level (standard conditions) at a standard gross weight. In order to determine specification compliance, use the faired curve of  $\Delta P/q_{C_i}$  vs.  $V_{i_w}$  to determine  $\Delta P$  for various  $V_{i_w}$   $\left[ \Delta P = P/q_{C_i} \text{ where } q_{C_i} = f(V_{i_w}) \text{ since } V_{i_{\text{test}}} = V_{i_w} \right]$ . Determine and plot the variation of  $\Delta \text{pos}$  with  $V_{i_w}$  where  $\Delta \text{pos} = V_C$  (based on  $q_{C_i} + \Delta P$ ) less  $V_i$  (in this case equal to  $V_{i_w}$ ). Also determine and plot for sea level only the variation of  $\Delta H_{\text{pos}} = H_{\text{pc}} \text{ less } H_{\text{pi}}$  (based on  $P_a + \Delta P$ ).

#### Determination of OAT System Recovery Factor

- a. Calculate  $q_C$  for each data point flown

$$q_C = q_{C_i} + \Delta P$$

- b.  $V_C = f(q_C)$  (Use charts of  $q_C$  vs  $V_C$  in Flight Test Reference Book.)

- c. Determine Mach number from  $V_C$ ,  $H_p$ , and  $M$ . Use  $V_C$  from step b and  $H_{\text{pc}}$  determined earlier.

- d. Calculate  $M^2$

- e. Correct  $\text{OAT}_o$  for instrument error

$$\text{OAT}_i = \text{OAT}_o + \text{OAT}_{ic}$$

- f. Plot  $\text{OAT}_i$  against  $M^2$  and draw a straight line through the data extending the line so that it intersects the line of zero  $M^2$ .

- g. This plot is a graphical solution of the OAT equation:

$$\text{OAT} = T_a (1 + 0.2 \text{ KM}^2) - 273, \quad ^\circ\text{C}$$

- h. Measure the slope,  $m$ , of the straight line.

- i. Calculate  $T_a$  from  $t_a$  determined in step e

$$T_a = t_a + 273, ^\circ K$$

- j. Determine recovery factor, K

$$K = \frac{m}{0.2T_a}$$

### Presentation of Results

The following curves should be presented. Sample presentations are included in the Guidelines to Flight Reports, and a sample data reduction sheet is included as Figure 9.

- a.  $\Delta H_{p_{pos}}$  versus  $V_i$
- b.  $\Delta V_{p_{pos}}$  versus  $V_i$
- c.  $\frac{\Delta P}{q_{C_i}}$  versus  $V_i$
- d.  $M^2$  versus  $OAT_i$



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SECTION V  
STALL AIRSPEED DETERMINATION

## REFERENCES

### Section V

1. USNTPS - FTM - No. 102, Section II.
2. Flight Test Technical Memorandum 72-1, "Determination of Performance Stall  
Airspeed at Altitude and Sea Level."



## STALL AIRSPEED DETERMINATION BY THE PACING METHOD

### PURPOSE

The purpose of this test is to determine the stalling airspeed of the test airplane in the takeoff and landing configuration. The variation of indicated stalling speed with airplane gross weight is required for presentation in the flight handbook. The calibrated stall airspeed will be used to determine compliance with the contract guarantee.

### DISCUSSION AND THEORY

In the classic aerodynamic one g stall, the coefficient of lift increases as the airspeed decreases until an angle of attack is attained at which the lift abruptly diminishes and the airplane ceases to fly. The stall airspeed for this case is well defined. In many airplanes, the classic stall does not occur. In these cases, the airplane will exhibit some characteristic at low airspeed which defines a minimum useable flying speed. This characteristic may be loss of effective control about one or more axis or a high sink rate. Although the aerodynamic stall has not occurred, the minimum useable flying speed is, in effect, the contract stalling speed of the airplane. Frequently, this airspeed is not well defined and is subject to interpretation by the particular pilot concerned. Therefore, it is most important that the stall speed for a particular airplane be defined consistently.

A pilot is primarily concerned with the indicated airspeed at stall, and this value is easily determined from the indication of the service airspeed system. In performance testing, accurate calibrated stall airspeeds are required to determine liftoff and obstacle clearance airspeeds for takeoff performance and are used in

determining carrier launch criteria. In addition, they are used as an incentive or penalty for airplane contractors (performance guarantees). Because of the large angles of attack attendant to the stall, the service airspeed system position error usually is large and not accurately known. One means of determining the stall airspeed precisely is by equipping the test airplane with a special pitot-static system with a known position error. Another method is to measure the stall airspeed of the test airplane from a pacer airplane which has a lower stall speed. The pacer airplane is equipped with a special pitot-static system for which the position error is accurately known.

In an airplane loaded at one center of gravity (cg) location, the stall will occur at the same coefficient of lift regardless of the airplane gross weight. Unless large thrust effects are present, the stalling airspeed will vary with the square root of the airplane gross weight. This relationship can be used to extrapolate stall speed data for one gross weight to other gross weights. If the cg location moves as the airplane weight changes, the coefficient of lift at stall also will change and the square root relationship will be approximate only. Another factor which influences stall speed is altitude. As altitude increases, the maximum lift coefficient decreases because of Reynold's number and Mach number effects. This results in an increase in stall speed as altitude increases; however, this effect is usually negligible for altitudes below 10,000 ft.

Nonsteady aerodynamic flow effects must be considered in the determination of stall airspeed. These effects are caused by the change in vorticity over the actual lifting surface. Any change in vorticity must take place over some finite period of time. The problem has been treated by a number of researchers, and analysis shows that airfoils changing flow patterns in distances less than 25 chord

lengths experience appreciable nonsteady flow effects. A satisfactory first approximation for calculating the ratio of actual to steady state lift values following a step change in airspeed is shown in equations (1) and (2). Equation (1) is for accelerations and equation (2) is for decelerations.

$$\frac{L}{L_0} = \frac{R + 1}{R + 2} \quad (1)$$

$$\frac{L}{L_0} = \frac{R + 2}{R + 1} \quad (2)$$

$L$  is lift after travel of  $R$  semi chord lengths, and  $L_0$  is steady state lift after travel of an infinite number of semi chord lengths. For the decelerating case, this chord travel can be expressed in terms of airspeed as shown in equation (3).

$$V_S = \sqrt{\frac{R + 1}{R + 2}} (V_{0_s}) \quad (3)$$

For uniform accelerations, the relation of actual to steady state lift is a more complex relationship, but still retains a primary relationship to chord lengths traveled. The acceleration case has received considerable attention with regard to catapult design. The deceleration case has been generally neglected in technical literature. An empirical approach for determining  $R$  that appears to work well is proposed in equation (4).

$$R = \frac{V_S}{\frac{C_D}{2} V} \quad (4)$$

Here,  $V_s$  is the measured (test) stall airspeed in KIAS;  $V$  is deceleration rate in kt/sec, and  $c$  is the chord in feet.

An example which demonstrates both the use of these equations and the importance of a small deceleration rate follows.

$$V_s = 114 \text{ KIAS}$$

$$V = -0.5 \text{ kt/sec}$$

$$\frac{c}{\lambda} = 3 \text{ ft}$$

therefore,

$$R = \frac{114}{3(0.5)} = 76$$

$$V_{Os} = \frac{76 + 2}{76 + 1} (114) = 114.4 \text{ KIAS}$$

However, at a deceleration rate of 5 kt/sec, the test stall airspeed is 108 KIAS. In this case

$$R = \frac{108}{3(5)} = 7.2$$

$$V_{Os} = \frac{9.2}{8.2} (108) = 114.4 \text{ KIAS}$$

The example shows the importance of a deceleration rate that is no greater than 1 kt/sec, and that in the airspeed range of these test results, a 5 kt/sec deceleration yields a test stall airspeed that is 6.4 kt too low. Such results please contractors, but are of little test value.

## TEST PROCEDURES AND TECHNIQUES

### Pertinent Particulars

- a. Approach stall speed at a deceleration rate of 1 kt/sec or less (1/2 kt/sec preferred).
- b. Alternate the configurations being tested.
- c. All data will be kneeboard data (no photopanel).
- d. Self-paced flights may be conducted in appropriately instrumented airplanes capable of obtaining calibrated stall airspeeds. Attempt to achieve the maximum possible difference between the gross weight at the heaviest data point and the gross weight at the lightest data point.

### Preflight Procedures

- a. Prepare pilot's and pacer observer's data cards. In flight, at each stall, record the following data:

<u>Test Airplane</u>	<u>Pacer</u>
Run No.	Run No.
Airspeed	Airspeed as read from special airspeed system
Fuel used or remaining	Pacer configuration
Configuration	Remarks as to validity of data point (i.e., any relative motion between airplanes)
Pressure altitude	

- b. Brief with pacer pilot as to rendezvous point and altitude, radio frequency, formation position, and hand signals in case of communication failure.
- c. Review stall approach technique and stall recovery procedures.

### Flight Procedures

Before the pacer joins the test airplane, the test pilot should perform at least one stall in each configuration to observe stall characteristics. If the airplane tends to pitch up or roll at the stall, instruct the pacer to join on the "safe" side. When cleared by the test airplane pilot, the pacer should take an echelon formation position, but not so wide that quick detection of relative motion is lost.

When the pacer is in position, the thrust of the test airplane should be smoothly set for the desired configuration and a steady deceleration of 1/2 kt/sec or less commenced. Control the deceleration rate by adjusting rate of climb or descent. A few seconds before stall call out "STAND-BY" and at the stall "MARK." Recover from the stall and record data.

As the test airplane decelerates, the pacer must maintain a constant relative position. For the landing configurations, the pacer will usually require landing gear and flaps DOWN and speed brakes EXTENDED in order to maintain position. If the pacer tends to overrun the test airplane, the best results are obtained by allowing the test airplane to descend below the pacer while maintaining a constant position in the vertical plane. If there should be relative motion at the stall, make an estimate of the relative speed and record this information on the observer's data card. Keep the test airplane in sight during the stall recovery, but do not dive to follow him. A quicker rendezvous is accomplished when the pacer maintains his altitude. When the pacer is again in position, repeat the test in the other configuration. Continue to conduct paced stalls in alternate configurations until the required data is obtained or minimum fuel is reached.

## DATA REDUCTION

### General

Report the variation of calibrated and indicated airspeed at stall with gross weight for the test airplane. Determine compliance with the stall speed guarantee. Record the following data on the data reduction sheet:

$V_{0t}$	$V_{0p}$	$h_{p0}$	Configuration	Remarks
----------	----------	----------	---------------	---------

### Data Reduction and Presentation

- Correct pacer observed airspeed for instrument and position error to determine calibrated airspeed at the stall.
- Correct test airplane observed airspeed to instrument error to determine indicated airspeed at the stall.
- Calculate test airplane gross weights for each stall.
- Plot the variation of indicated airspeed with gross weight for configurations Takeoff and Landing. Plot the variation of calibrated airspeed with gross weight for configuration Landing and indicate the contract guarantee.

SECTION VI

DETERMINATION OF EXCESS POWER CHARACTERISTICS



## SECTION VI

### NOTATIONS INTRODUCED IN THIS SECTION

$\Delta H_p$	Difference in pressure altitude at the upper and lower limit of an altitude band centered at the test altitude
$R/C_i$	Average indicated rate of climb, $\Delta H_{pi}/\Delta t$
$W_A$	Average test gross weight, <u>sum of test weights at each point</u> number of points
$E_h$	Energy height, or specific energy
$dE_h/dt$	Rate of change of energy height
$h$	Geometric altitude, ft
$H_p$	Pressure altitude, ft
$s$	Subscript signifying standard day conditions and standard airplane gross weight
$\Delta t$	Increment of time required to climb through test altitude band
$P_s$	Specific excess power, equals $dE_h/dt$

## REFERENCES

### SECTION VI

1. AGARD Vol. I, pp. 4:37-4:40 and 7:27-7:31
2. Dommasch, Airplane Aerodynamics, pp. 289-290
3. Rutowski, DAC, "Energy Approach to the General Aircraft Performance Problem," 1953
4. Boyd, J. R., MAJ, USAF, "Energy - Maneuverability," 1966

## DETERMINATION OF EXCESS POWER - LEVEL ACCELERATION RUNS

### AND SAWTOOTH CLIMBS

#### PURPOSE

The purpose of conducting acceleration runs and sawtooth climbs is to determine the variation of excess power with airspeed. For acceleration runs, the data will be used to evaluate level acceleration characteristics and maximum level flight airspeed, to determine one or more climb schedules, and can be used to estimate level flight turning performance. For sawtooth climbs, the data will be used to evaluate climb characteristics.

#### DISCUSSION AND THEORY

##### General

In analyzing acceleration or climb performance, it is advisable to consider the airplane from the standpoint of total energy (TE) which is the sum of potential (PE) and kinetic (KE) energies. The total energy of an airplane at a given weight (W), airspeed (V), and geometric altitude  $^*(h)$  is,

$$TE = PE + KE = Wh + \frac{W}{2g} V^2 \quad (1)$$

dividing by W and defining energy height as:

$$E_h = \frac{TE}{W} = h + \frac{V^2}{2g} \quad (2)$$

\*Geometric altitude assumed equal to geopotential altitude

Energy height, or specific energy, can also be interpreted as energy altitude; the altitude which could be attained if all the kinetic energy were perfectly converted to potential energy. Differentiating, we obtain:

$$\frac{dE_h}{dt} = \frac{dh}{dt} + \frac{V}{g} \frac{dV}{dt} \quad (3)$$

Now consider the free-body diagram of an airplane in climbing flight where the thrust ( $F$ ) is assumed to be along the flight path:

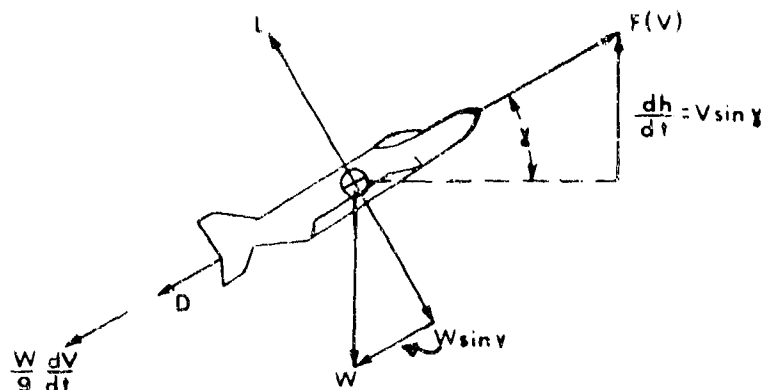


Figure 1  
Free-Body Diagram

$$\Sigma F = \frac{d(mV)}{dt} = ma = \frac{W}{g} \frac{dV}{dt} \quad (\text{if } W \text{ assumed constant}) \quad (4)$$

and summing forces along the flight path:

$$\Sigma F = F - D - W \sin \gamma = \frac{W}{g} \frac{dV}{dt} \quad (5)$$

or, rearranging:

$$\sin \gamma + \frac{1}{g} \frac{dV}{dt} = \frac{F - D}{W} \quad (6)$$

noting that:

$$\sin \gamma = \frac{\left(\frac{dh}{dt}\right)}{V} \quad (7)$$

substituting (7) into (6) and multiplying by V:

$$\frac{dh}{dt} + \frac{V}{g} \frac{dV}{dt} = \frac{V(F-D)}{W} \quad (8)$$

Now notice that we have an expression for the airplane's excess power  $V(F-D)$ .

Comparing with (3), we define specific excess power as:

$$P_S \equiv \frac{V(F-D)}{W} = \frac{dE_h}{dt} = \frac{dh}{dt} + \frac{V}{g} \frac{dV}{dt} \quad \left( \frac{\text{ft} - \text{lb}}{\text{lb} - \text{sec}} \right) \quad (9)$$

#### Level Acceleration Runs

In acceleration tests, the first term of equation (9) can be eliminated by maintaining constant altitude ( $\frac{dh}{dt} = 0$ ), and specific excess power can be measured in terms of airspeed and linear acceleration ( $\frac{dV}{dt}$ ). The variation of specific excess power with airspeed is determined for one altitude over the entire level flight speed range of the airplane. This is done by conducting level acceleration runs from near minimum airspeed ( $V_{\min}$ ) to the maximum level flight airspeed ( $V_{\text{crit}}$ ) or maximum airspeed ( $V_{\max}$ ).

Practically, a constant altitude acceleration run cannot be flown. Imprecise pilot technique may cause oscillations about the desired altitude and changes in altimeter position error may result in a slight climb or descent during the acceleration; however, because potential and kinetic energies are both computed, the resultant total rate of change of specific energy is unaffected.

In other words, the airplane performance is considered in terms of total energy rather than potential or kinetic energy separately, and slight variations in altitude (potential energy) are considered perfectly convertible to airspeed (kinetic energy) or vice versa. The validity of this assumption depends upon smooth piloting technique and a small divergence from the assigned altitude. Abrupt longitudinal control inputs result in unaccountable energy losses, and large deviations from test altitude result in unaccountable changes in engine performance from that at the test altitude. The accuracy desired dictates the allowable excursions in normal acceleration and altitude during any run. In general, the allowable excursions increase with excess power available and decrease with increasing altitude (less excess power). Typical values to maintain  $P_g$  accuracy within approximately 5% are:  $\pm 300$  ft altitude excursion and  $\pm 0.10g$  normal acceleration for an airplane with over  $60 \frac{\text{ft-lb}}{\text{lb-sec}}$  specific excess power.

Remember the two assumptions: no lift due to thrust and negligible gross weight change. The first of these will affect both level and climbing flight about equally. Since fuel consumption will be very small between data points recorded by the photopanel about every 2 sec, the assumption of negligible gross weight change should be valid for all current aircraft types (with the possible exception of supersonic, afterburning flight).

### Sawtooth Climbs

In sawtooth climbs tests, the second term of equation (9) is eliminated by maintaining a constant true airspeed ( $\frac{dV}{dt} = 0$ ), and excess power is measured in terms of vertical speed ( $\frac{dh}{dt}$ ). The variation of specific excess power with airspeed can be determined for one altitude by conducting a series of short, constant true airspeed climbs at various airspeeds over the speed range of the airplane.

Since climb performance at given calibrated airspeeds is usually the data desired from sawtooth climbs, true airspeeds may not be held constant. For this case, true airspeed is increasing throughout the test climb which will result in actual  $P_s$  being greater than the indicated rate of climb.

A graph of the variation of rate of climb with airspeed for the test conditions is prepared from these data. A horizontal tangent to this curve defines the airspeed for maximum rate of climb at that altitude and gross weight. A tangent through the origin ( $V = 0$ ,  $R/C = 0$ ) determines the airspeed for maximum climb angle.

The observed climb performance is affected by changes in airplane gross weight and by variations in power or thrust available. A change in airplane weight has two effects on climb performance. First, the rate of climb and climb angle are inversely proportional to weight. In addition, a weight change causes variations in induced drag, which alters the power and/or thrust required curve, thus affecting the airspeeds at which maximum rate of climb and maximum climb angle occur. The change in these airspeeds is negligible for small variations around a standard (mission-related) gross weight. Variations in ambient temperature produce changes

in power (or thrust) available that are essentially constant with airspeed for propeller driven (jet) airplanes. Therefore, variations in power available will affect only the value of the rate of climb for propeller driven airplanes and not the airspeed at which it occurs. Similarly, the effect of variations in ambient temperature on the airspeed for maximum climb angle for turbojet airplanes is negligible.

When only the airspeeds for (and not the actual values of) maximum rate of climb and maximum climb angle are of interest, the effect of decreasing weight during the test may be minimized by obtaining the data in a particular order. If the constant airspeed climbs are flown in ascending order of airspeed, than in descending order at the same airspeeds, the average airplane gross weight for each pair of data points will be approximately equal. This technique may be further complemented by correcting each data point to an average test weight. A curve faired through all of the data will be representative of the climb performance at the average gross weight and the test day conditions (see Figure 7 for an example of such a graph).

Other factors which may affect the test results are the influence of vertical air currents and changing wind velocity with altitude (wind shear). Vertical air currents usually can be avoided by careful selection of test conditions. The effect of wind shear can be minimized by performing climbs on a heading which is perpendicular to the reported winds.



In general, high climb rates and excessive testing time render the sawtooth climb method impractical for turbojet airplanes except in the high lift and single engine configurations. The most significant advantage is that the data are collected in the climb attitude and the reduction in induced drag associated with climbs is accounted for.

## TEST PROCEDURES AND TECHNIQUES-ACCELERATION RUNS

### Preflight Procedures

a. Required data.

Time (or time increment).

$V_o$  (continuous record during run).

$H_{p_o}$  (continuous record during run).

OAT (start and end of run) or  $T_a$ .

Fuel used or fuel remaining (continuous record during run).

Photopanel is primary data source. Record correlating data on kneepad card.

b. Prepare data card (Figure 2).

WEATHER				CARD NO.				
ACCELERATION RUNS				LP-1				
				PTR DIS				
AIRPLANE TYPE		SN. NO.		TIME T.O.		DATE		
T-38A		158197		TIME LAND		25 Nov 78		
PILOT				T.O.C.G.		GEAR DOWN		
PIPE						UP 26.2 s		
CONDITION						T.O. GROSS WEIGHT		
NORMAL TRAINER - 1 PILOT						12,150 lb		
EXTERNAL CONFIGURATION								
NOSEBOOM, OAT probes								
start				END				
RUN	H <sub>pa</sub>	V <sub>0</sub>	OAT <sub>0</sub>	FUEL	V <sub>0</sub>	OAT <sub>0</sub>	FUEL	P/P
1								
2								
3								
4								
5								
6								

FLIGHT DATA PRNC DATE: 5900 S

Figure 2  
Sample Pilot's Data Card

### Flight Procedures

- a. Conduct a practice acceleration run at the test altitude commencing as near  $V_{\min}$  as practicable. Use the acceleration run technique described in Section II and demonstrated during the performance demonstration flight.
- b. Take data on next run.
- c. Repeat procedure at other assigned altitudes.

### DATA REDUCTION - ACCELERATION RUNS

#### General

Data reduction to standard conditions is normally performed by Computer Services Division (CSD). Data inputs are presented in Section XIII. This data reduction scheme accounts for the effects of nonstandard gross weight, ambient temperature (thrust), and induced drag changes.

For manual data reduction, follow the order presented below in setting up the data reduction form. Reduce the best acceleration run at each altitude assigned. Runs made near standard gross weight are preferred because the corrections will be smaller. Record elapsed time  $t$ ,  $V_O$ , and  $H_{p_O}$  each 10 sec (at least 15 data points required). Record  $OAT_O$  for three data points.

#### Manual Determination of Specific Excess Power, $P_s$

Test day rate of change of specific energy corrected to a standard airplane weight will be determined for each data point. Rate of change of specific energy is analogous to specific excess power.

- a. Correct  $V_o$  for instrument and position error to obtain  $V_c$ . Assume lag error negligible.
- b. Correct  $H_{p_o}$  for instrument and position error to obtain  $H_p$ .
- c. Determine Mach number corresponding to above values of  $V_c$  and  $H_p$ .
- d. If necessary, calculate  $T_a$  ( $^{\circ}\text{K}$ ) from  $\text{OAT}_o$  at three data points.
  1. Correct  $\text{OAT}_o$  for instrument error.
  2. Calculate  $M$  corresponding to time of observation of  $\text{OAT}$ .
  3. Determine  $t_a$  by entering Appendix IV with  $\text{OAT}$  and  $M$ .
  4. Calculate  $T_a$  ( $^{\circ}\text{K}$ ) =  $t_a$  ( $^{\circ}\text{C}$ ) + 273.
- e. Calculate TAS for each data point. Use the average value of  $T_a$  determined in step d.

$$V_T = 11 \sqrt{59RT_a} = 65.7 \sqrt{T_a} \text{ (}^{\circ}\text{K)} \quad (\text{fns})$$

- f. Construct a working plot of the variation of  $V_T$  and  $H_p$  with elapsed time. Use scales of 100 fps/in. for airspeed, 200 ft/in. for altitude, and 20 sec/in. for time. Fair a curve through the data points. After fairing the data, you will observe that at an instant when the data shown is greater than an average rate of climb, the linear acceleration is less than average (see Figure 3 for an example of such a plot).

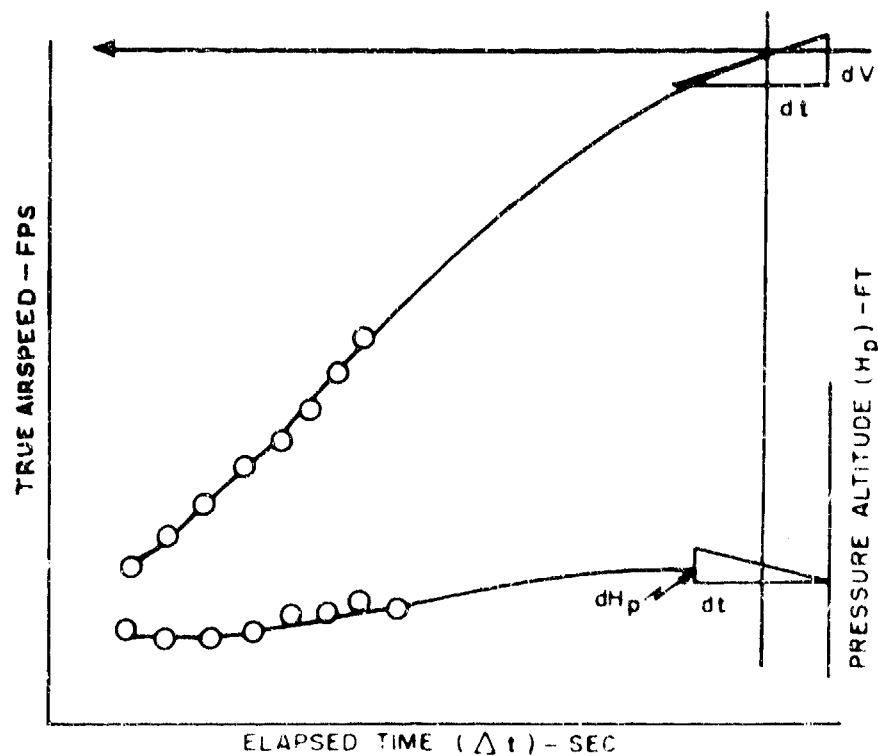


Figure 3  
Acceleration Run Working Plot

g. Graphically measure  $dV/dt$  and  $dH_p/dt$  for at least eight points on the curves of step f. Record also the corresponding value of  $V_T$ . Caution - a protractor should not be used for measuring these derivatives.

h. Obtain standard day ambient temperature ( $T_{a_s}$ ) from ICAO atmosphere table. Calculate rate of change of geometric altitude,  $dh/dt$ . This correction is necessary because of nonstandard density.

$$\frac{dh}{dt} = \frac{dH_p}{dt} \times \frac{1.0}{(T_{a_s})} \quad (\text{fps})$$

i. Determine airplane weight for the run. Use fuel data recorded at the end of the run.

j. Calculate ratio of airplane weight for test to standard airplane weight,  $W_t/W_s$ . Use the values of airplane standard weight listed in TPS Airplane Details.

#### Correction of $P_s$ to Standard Conditions

In determining rate of change of specific energy for standard conditions, it is necessary to account for the differences in airplane weight from standard weight and ambient temperature at the test altitude from standard day ambient temperature. Differences in weight affect the momentum and the induced drag. These differences result in a different acceleration characteristic being observed for the test conditions than would have occurred for standard conditions. While the corrections for gross thrust and ram drag cannot be considered negligible (particularly near the service ceiling of the airplane), these corrections are tedious and will not be made manually. The induced drag correction is small in relation to the rest of the terms and will also be neglected for the manual data reduction. Both these corrections are included in the computer data reduction.

k. Calculate rate of change of specific energy for standard conditions. The gross weight correction is implicit in the equation.

$$P_s = \left( \frac{W_t}{W_s} \right) \left( \frac{P_t}{P_s} \right) \left( \frac{P_t}{P_s} \right) + \left( \frac{W_t}{W_s} \right) \left( \frac{P_t}{P_s} \right) \left( \frac{P_t}{P_s} \right)$$

1. Compute Mach number for each value of  $V_T$  obtained in step g.

$$M = \frac{V_T \text{ (fps)}}{65.7 \times T_a \text{ (}^\circ\text{K)}}$$

### Graphical Presentation of Results

- a. Plot the variation of specific excess power for standard weight and ambient temperature with Mach number or calibrated airspeed (Figure 4).

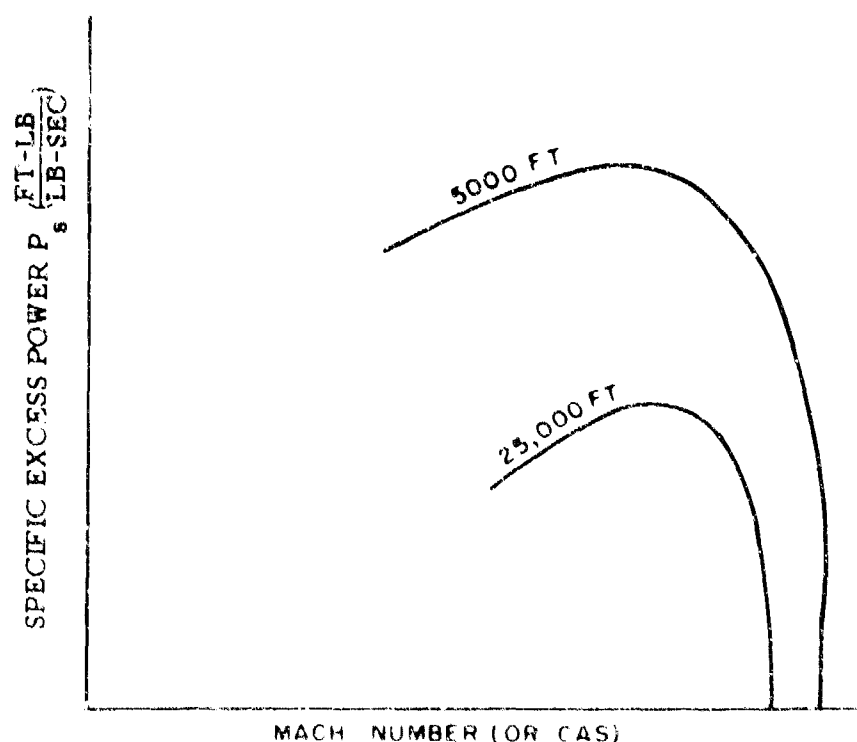


Figure 4  
Graph of Excess Power Characteristics

- b. When submitting the flight report, include the working plot of the variation of airspeed and altitude with elapsed time (step f).

## TEST PROCEDURES AND TECHNIQUES - SAWTOOTH CLIMBS

### Pertinent Particulars

a. Make several constant airspeed climbs (or descents) in ascending order of observed airspeeds from the minimum test airspeed to the maximum. Then make the same number of climbs in descending order at approximately the same airspeeds.

b. Use military or combat rated thrust in jets. Maintain constant power in propeller driven airplanes by adjusting MAP or torque as necessary during the climb. For the altitudes at which full throttle is reached attempting to maintain constant power, use the MAP or torque at the test altitude as an average value. Note the altitude at which full throttle is reached.

c. Photopanel data are desired; however, prepare the report on the basis of kneeboard data if photopanel data are not available.

### Preflight Procedures

a. Required Data.

$V_o$

Elapsed time for climb ( $\Delta t$ ).

Altitude band ( $\Delta H_p$ ).

Fuel remaining.

OAT at test altitude (or  $T_a$ ).

RPM and MAP or torque at test altitude.

b. Prepare data card (Figure 5).



WEATHER				LOAD V.			
SAWTOOTH CHAIRS				LP-1			
DISPLACED (LBS)				PIU BLS			
T-28B		151451		TIME LHD		TPS-76-70-1	
PIPE				DATE		15 MAR 78	
NORMAL TRAINER (2 pilots)				UP 25.5		8395	
ALTITUDE BAND: 2000 - 4000 FT							
RUN	V <sub>0</sub>	dt	WF	CAT	RPM	MAP	
1	85						
2	110						
3	120						
4	130						
5	140						
6	150						
7	160						
8	170						
9	180						
10	190						
11	200						
12	210						
13	220						

SPEED BRAKES

Figure 5  
Sample Pilot's Data Card

- c. Review assigned scope of test and make notes on data card to be used as a reminder during the test, such as: configuration details, desired climb airspeeds, and altimeter barometric setting of 29.92 in Hg.
- d. Obtain test altitude forecast winds from aerology.
- e. Carry a 60-sec stopwatch.
- f. Review photopanel operating procedures and switchology. For this test, use a camera speed of 1/2 frame per second.
- g. Before leaving the flight line, check the operation of the photopanel.

#### Flight Procedures

The sawtooth climbs method consists of a series of constant observed airspeed climbs through an altitude band. All climbs are performed on the same heading perpendicular to the forecast wind at the test altitude.

- a. At 300 to 500 ft below the altitude band on the required climb heading, trim the airplane in level flight at the desired climb airspeed.
- b. Set the power or thrust and establish a stabilized climb. Check configuration details and trim the airplane. Maintain constant power by adjusting MAP or torque frequently in the climb.
- c. Energize the photopanel about 10 sec before entering the altitude band.
- d. With the stopwatch, measure the time required to transit through the altitude band or 2 min, whichever is less.
- e. When passing the test altitude, mentally note the OAT, MAP (or torque), and RPM readings.
- f. About 10 sec after passing out of the altitude band, de-energize the photopanel, level off, and record data (Figure 6).

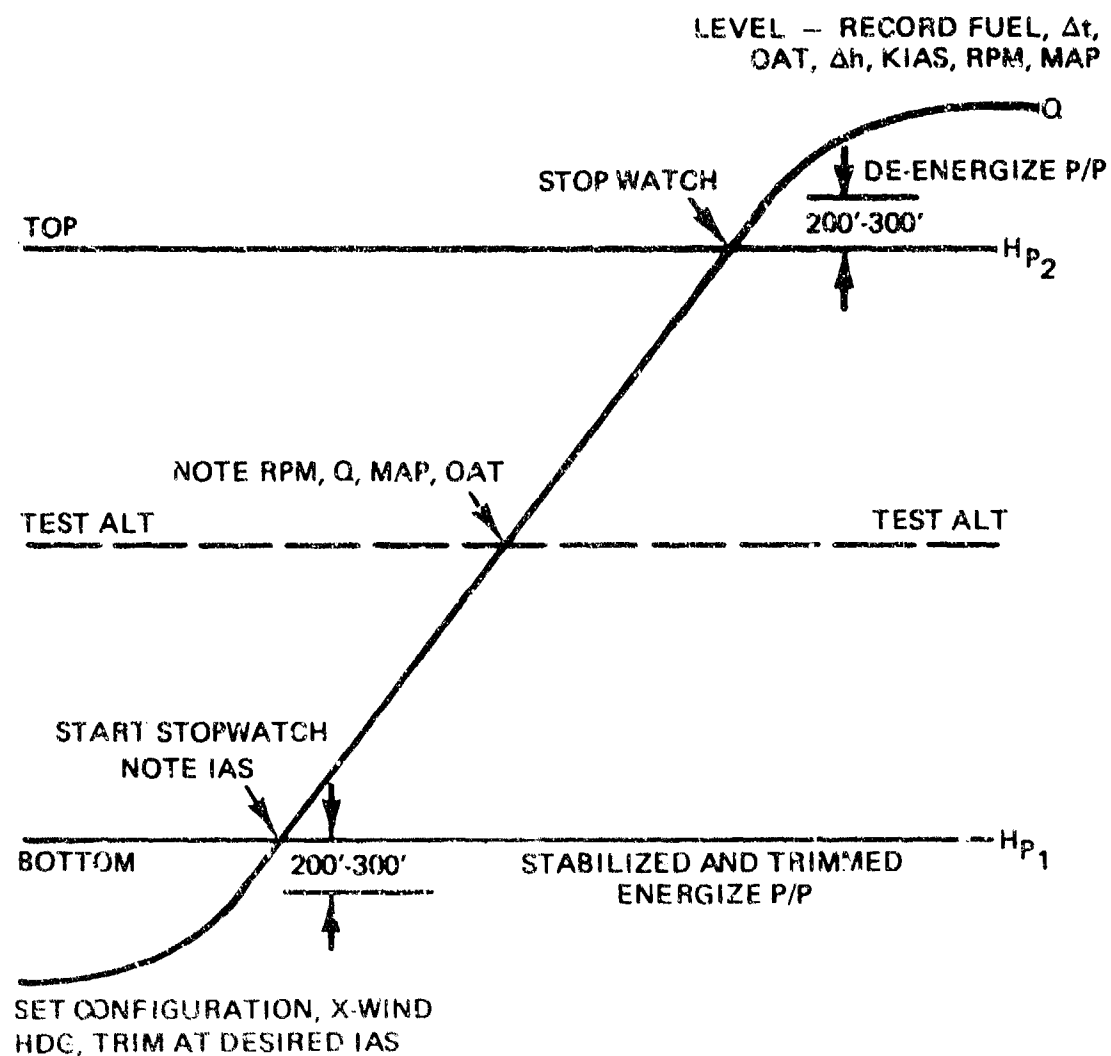


Figure 6  
Sawtooth Climb Flight Procedures

g. Reverse course, descend, and repeat the procedure at the next desired airspeed.

h. Since the rate of climb will be zero, the maximum airspeed run must be made at the test altitude. Ensure that the airplane has stabilized in airspeed before recording data.

i. It is not imperative that the climb airspeeds be precisely the planned airspeed for the climb. For good results, the airspeed at the upper limit of the band must be identical with the airspeed at the lower limit. In the climb, attempt to maintain airspeed within  $\pm 1$  kt of the initial value.

j. If the wind at the test altitude is not strong, it is preferred to make the climbs in a racetrack pattern. In stronger wind conditions, use an "S" pattern with all turns upwind.

## DATA REDUCTION - SAWTOOTH CLIMBS

### General

Sawtooth climb data will be reduced on the school's Hewlett-Packard computers, which make corrections to standard airplane gross weight and to standard atmosphere conditions (thrust correction). Review the appropriate program folder to determine data requirements and input format.

Transcribe the following data on a data reduction sheet.

$V_o$ ,  $H_{p_{o1}}$ ,  $H_{p_{o2}}$ ,  $\Delta t$ , RPM or torque, MAP,  $w_f$ , and OAT at mid-altitude.

### Manual Determination of Airspeeds for Highest Rate of Climb and Maximum Climb

#### Angle

a. Correct observed airspeed and altitude for instrument error. Assume negligible difference between the altimeter position error at the lower altitude and the error at the upper altitude.

$$V_i = V_o + V_{ic}$$

$$H_{pi} = H_{po} + H_{pic}$$

b. Calculate.

$$\Delta H_{pi} = H_{pi1} - H_{pi2}$$

c. Calculate indicated rate of climb.

$$R/C_i = \frac{\Delta H_{pi}}{\Delta t} \quad \text{ft/min}$$

d. Calculate airplane gross weight for each data point.

e. Calculate average test gross weight

$$W_A = \frac{W \text{ for each test point (sum)}}{\text{number of runs}}$$

f. Calculate weight correction factor for each data point

$$\frac{W_t}{W_A}$$

g. Ignoring the induced drag correction, corrected rate of climb for each data point may be calculated

$$R/C_c = \left( \frac{W_t}{W_A} \right) (R/C_i)$$

This approximation is not always valid for rates of descent.

h. Correct at least three observed OAT values for instrument error.

i. Determine Mach number at which each OAT value in step h was obtained.

Enter Appendix II with  $V_i$  and mid-altitude.

j. Determine ambient air temperature from Appendix IV.

k. Construct a graph of variation of corrected rate of climb with indicated airspeed. Fair one curve through the data. This curve shows the variation of corrected rate of climb with airspeed for the average gross weight of the test and test conditions (see Figure 7).

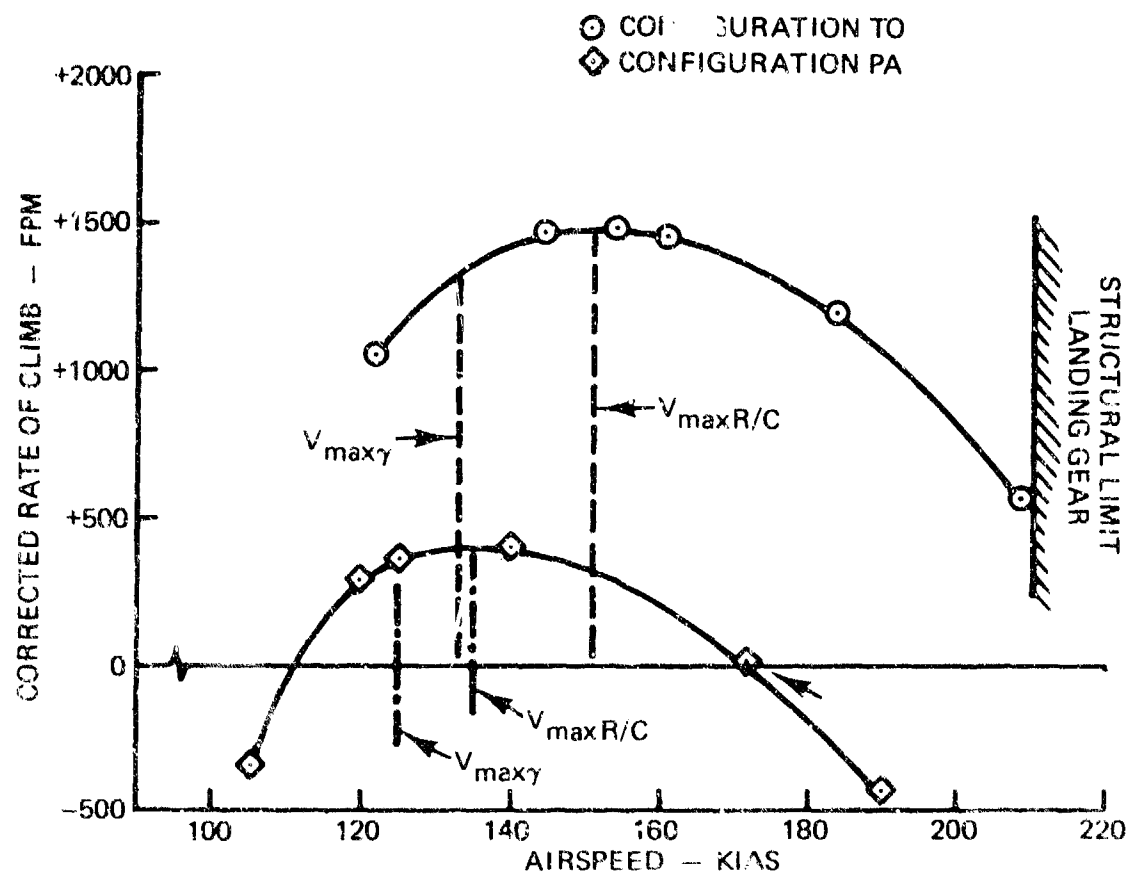


Figure 7  
 Sample of Graphical Presentation of Results

SECTION VII  
LEVEL FLIGHT PERFORMANCE



## REFERENCES

### Section VII

1. AGARD, VOL I, Chapter 4; pp 4:1-4:23; Chapter 6, Sec 6.2.
2. Petersen, Aircraft and Engine Performance, Chapters 4, 5, and 6.

### NOTATIONS INTRODUCED IN SECTION VII-A

$F_g/\delta$	Engine referred gross thrust (a function of the engine exhaust nozzle pressure ratio)
$w_f$	Fuel flow, pph
$F_n/\delta$	Referred net thrust, lb
$w_f/\delta\sqrt{\theta}$	Referred fuel flow, pph
$N/\sqrt{\theta}$	Referred engine speed, percent of rated RPM
$W/\delta$	Referred gross weight, lb
$W_F$	Fuel weight

### NOTATIONS INTRODUCED IN SECTION VII-C

$SHP_c$	Chart shaft horsepower determined from the manufacturer's engine operating chart for the conditions of the test.
$SHP_t$	Test shaft horsepower actually delivered by the engine for the test conditions. Determined from correction of $SHP_c$ or from engine torquemeter installation.
$\Delta SHP_{R/C}$	SHP correction for rate of climb or descent.
$BHP_{rt}$	Brake horsepower required for stabilized, level flight (i.e., airplane drag) for the test conditions (gross weight, pressure altitude, and ambient air temperature).
$SHP_{ew}$	Equivalent shaft horsepower required referred to a standard airplane gross weight.
$V_{md}$	Airspeed for minimum drag.
$w_f$	Referred fuel flow.
$\delta\sqrt{\theta}$	
$SHP$	Referred shaft horsepower.
$\delta\sqrt{\theta}$	
$RNI$	Reynold's No. Index
$\frac{\delta}{\rho\sqrt{\theta}}$	

## SECTION VII-A

### JET AIRPLANE LEVEL FLIGHT PERFORMANCE

#### PURPOSE OF TEST

The purpose of this test is to evaluate the level flight performance of a jet airplane by:

- a. Determining the thrust required and/or fuel flow required.
- b. Determining the referred gross thrust or fuel flow available.
- c. These data will be determined for one value of referred gross weight ( $W/\delta$ ) and will subsequently be correlated with similar data for other  $W/\delta$  values to determine the level flight performance over the operating range of the airplane.

#### DISCUSSION AND THEORY

The thrust required for level flight of an airplane may be described in terms of the functional expression

$$\frac{F}{\delta} = f \left( M, \frac{W}{\delta} \right) \quad (1)$$

This expression was derived from a theoretical analysis of level flight performance (drag equation). It suggests a flight test procedure for experimentally determining the thrust required characteristics; maintain a constant value of  $W/\delta$  in flight and measure the referred thrust required at several values of Mach number. If we repeat this procedure for several values of  $W/\delta$  and construct a suitable graph of these results, the thrust required for the airplane may be determined for any combination of airplane gross weight, pressure altitude, and Mach number. A flight test procedure of this form is particularly suitable for jet airplanes because the

changes of airplane drag caused by Mach number effects are accounted for, and a wide range of airplane gross weight can be analyzed in only a few flights. But thrust is difficult to measure and correct to standard conditions (EPR gauge would be useful), and we are more concerned with range and endurance data. In order to evaluate range and endurance, we must define the combined performance of the airframe and the engine. By dimensional analysis of the factors affecting jet engine performance, we can show that:

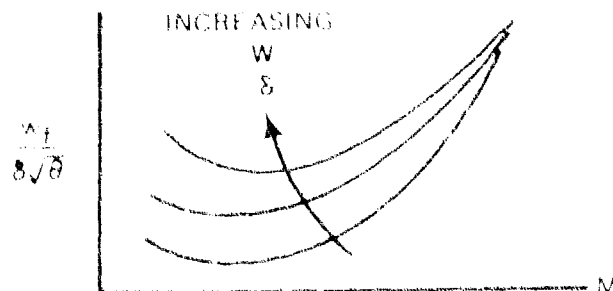
$$\frac{W}{\delta \sqrt{\theta}} = f(M, \frac{N}{\sqrt{\theta}}) \quad (2)$$

$$\frac{F}{\delta} = \frac{D}{\delta} = f(M, \frac{N}{\sqrt{\theta}}) \quad (3)$$

These relationships describe the installed jet engine. Combined with the airframe relationship (Equation (1)) the generalized equation for the total airplane in level flight is

$$\frac{W}{\delta \sqrt{\theta}} = f(M, \frac{W}{\delta}) \quad (4)$$

Now we can fly our airplane at constant ratios of  $\frac{W}{\delta}$ , measure (referred) fuel flow when stabilized in level flight at several Mach numbers, and define curves describing the performance of the airplane over a wide range of gross weights and altitudes.



Two assumptions were made: no lift due to thrust (high  $\alpha$ ) and negligible Reynold's number effects.

In level flight, referred gross weight decreases as fuel is used. A constant value of  $W/\delta$  may be maintained by increasing pressure altitude at a predetermined schedule as airplane gross weight decreases. In obtaining a series of level flight thrust required data points each at the same value of  $W/\delta$ , the altitude must be increased for each successive data point with the airplane arriving at each altitude with sufficient excess fuel to stabilize in airspeed before the desired  $W/\delta$  occurs. The flight path may be visualized as a series of ascending steps.

## TEST PROCEDURES AND TECHNIQUES

### Pertinent Particulars

a. Use constant altitude technique for data points at airspeeds above that for minimum drag and constant airspeed (thrust adjustment) techniques for airspeeds near and below minimum drag where stabilization at constant thrust is difficult.

b. Obtain sufficient data to define airframe and engine characteristics for one value of  $W/\delta$ . Concentrate on defining the minimum thrust (fuel flow) and maximum range portions of the curve, if preliminary data are available.

c. It is recommended that the  $V_{\max}$  data point at assigned  $W/\delta$  be taken first and the remainder of the  $W/\delta$  data be obtained in order of decreasing airspeeds.

d. Determine the thrust or fuel flow available at the assigned reference pressure altitude using an acceleration run technique. Obtain data from  $1.7 V_{\text{mrt}}$  to the maximum level flight airspeed. For the lowest  $W/\delta$  values, the thrust available data should be obtained first in order to reduce the airplane's gross weight and thus increase the initial  $W/\delta$  test altitude.

## Preflight Procedures

- a. Obtain  $W/\delta$  assignment.
- b. Prepare pilot's data card. In flight, at each stabilized airspeed at the assigned  $W/\delta$  value, record the following data on the data card and the photopanel.

Airspeed (V)

Pressure altitude ( $H_p$ )

Fuel flow ( $w_f$ )

Engine RPM (N)

Engine Pressure Ratio

OAT or  $T_a$

Fuel remaining ( $W_f$ ) or fuel used

Photopanel counter number

- c. Construct a  $W/\delta$  card. The  $W/\delta$  card is a kneeboard size graph which shows a variation of pressure altitude (corrected for instrument error) with fuel. It is used to maintain a constant referred gross weight ( $\frac{W}{\delta}$ ). The card is constructed as follows: for at least four fuel weight values between the appropriate initial airplane weight at altitude and 1,000 lb fuel remaining:

1. Calculate airplane gross weight,  $W$ .
2. Calculate  $\delta$  required to maintain assigned  $W/\delta$

$$\delta = \frac{W}{W/\delta}$$

3. Obtain corresponding pressure altitude from interpolation of ICAO atmosphere tables.

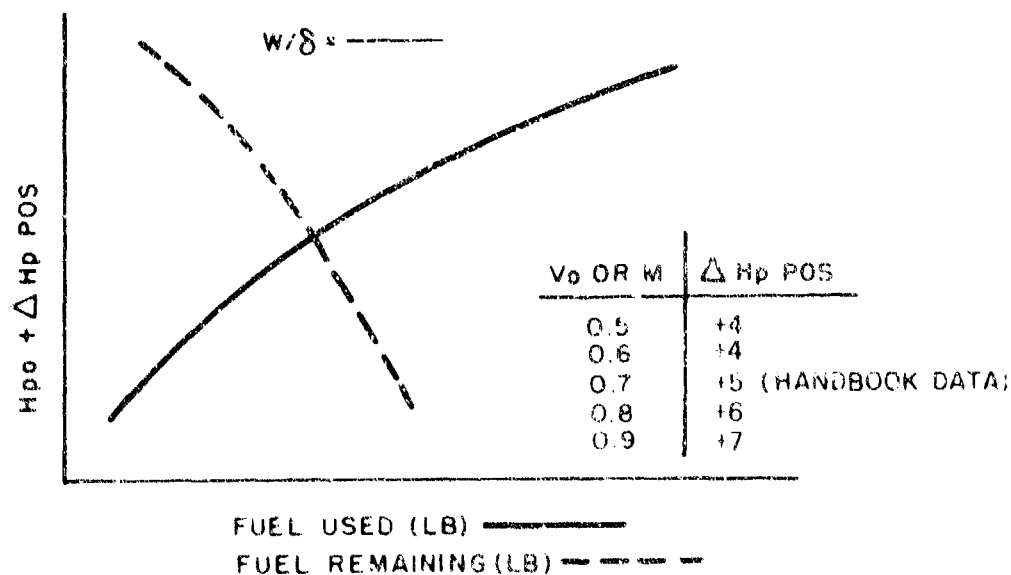
4. Compute

$$H_{p0} + \Delta H_{p\text{pos}} = H_{pC} - \Delta H_{p\text{ic}}$$

5. For several values of Mach number at the average pressure altitude of the test, determine the altimeter position error correction.

6. Construct a graph of the variation of  $H_{p_0} + \Delta H_{p_{pos}}$  with fuel used. Include altimeter position error information in a table on the graph.

d. Use of  $W/\delta$  card. Estimate the fuel used or remaining value for the next data point, enter the graph, and obtain the required pressure altitude. Subtract from this figure the approximate position error and stabilize the airplane at this computed observed pressure altitude. An example of such a graph is:



e. A simple and easy method of determining the observed pressure altitude required to maintain an assigned  $W/\delta$  value is to have radio contact with a flight test engineer who can compute the required altitude as the airplane gross weight decreases during the flight. The pilot informs him of the estimated fuel and the airspeed at the next data point. The engineer computes the airplane gross weight and the pressure altitude required to maintain the assigned  $W/\delta$  at that weight. He then subtracts the altimeter position error and instrument error and informs the pilot of the required observed pressure altitude.

## Flight Procedures

### a. Constant Altitude Technique (Stable Equilibrium Conditions)

1. This technique requires the pilot to estimate the fuel required to stabilize at each data point. The amount of fuel required will be dictated by: (a) engine/airplane characteristics, (b) weather conditions, and (c) pilot proficiency. For constant altitude points, 5 min or 300 lb fuel can be used as an initial allowance for stabilization until the pilot becomes more familiar with the requirements. At the first point, subtract the estimated fuel required from the present fuel reading; the resulting figure will be the fuel weight reading at the data point. From the W/δ card determine the observed pressure altitude required to maintain the assigned W/δ at that fuel reading. Record the computed observed pressure altitude and fuel reading for reference.

2. Climb to the computed observed pressure altitude, set the thrust, and allow the airspeed to stabilize. Military Rated Thrust is recommended for the first data point ( $V_{mrt}$  point).

3. Check configuration.

4. Precisely maintain the computed altitude until the reference fuel reading is obtained. If the airspeed is stabilized, record the data. If not stabilized, record data anyway, but set up a new point at the same thrust.

5. Estimate fuel weight at the next data point and repeat steps 2, 3, and 4.

### b. Constant Airspeed or Thrust Adjustment Techniques (Unstable Equilibrium Conditions)

1. At airspeeds less than the airspeed for minimum drag, where stabilization at a constant altitude and thrust setting is difficult, use the constant airspeed technique.



2. Determine the reference fuel reading and pressure altitude as in the constant altitude technique.

3. At the required altitude, maintain the desired airspeed and adjust the thrust to maintain level flight. The task is made more difficult by the necessity of attaining the required test altitude. A certain degree of pilot skill and patience is required to do this well. First set a thrust which results in a climb at the trim airspeed. Then reduce the thrust to obtain a descent. At the required altitude, adjust thrust to obtain level flight or a slight climb. Continue these "bracketing" corrections until stabilized at the required altitude and airspeed.

4. When the reference fuel reading is obtained, record data.

c. If using a fuel used system, record fuel remaining at the last  $W/\delta$  data point also. Later, calculate gal fuel remaining and add to final fuel used reading to check accuracy of fuel used counter.

d. Determination of thrust available. Before or after obtaining the required  $W/\delta$  data, fly to the assigned reference altitude. Stabilize at approximately  $0.7 V_{mrt}$  with MIL thrust set and the speedbrakes partially extended. When the engine thrust has stabilized, energize the photopanel, leave MIL thrust set, and retract the speedbrakes. As the airplane accelerates in level flight, engine pressure differential, airspeed, pressure altitude, and OAT or  $T_a$  will be recorded out to  $V_{mrt}$ . These data define a curve of the variation of MIL thrust with airspeed for the test day conditions. For engines without thrust measurements, referred fuel flow data will be used. These data will later be corrected to standard day conditions to determine standard day  $V_{mrt}$  (correlated with thrust required data).

e. The recommended airspeed for endurance is sometimes determined from minimum acceptable flying qualities rather than minimum fuel flow. During the flight, record the airspeed which you would consider the minimum desirable from the standpoint of satisfactory handling qualities.

## DATA REDUCTION

### General

Airframe (drag) and engine (fuel flow) characteristics will be determined in terms of the variation of referred gross thrust, referred engine speed, and referred fuel flow with Mach number for one value of  $W/\delta$ . Because this presentation of results is generalized, data obtained on different days but in the same airplane should be compatible. Normally, data reduction is performed on the Hewlett-Packard computer with the first and last data points worked manually. Record the following data on the data reduction sheet:

$V_o$ ,  $H_{p_o}$ , Engine Pressure Ratio,  $w_{f_o}$ ,  $W_F$ ,  $N_o$ , OAT or  $T_a$

### Data Reduction - Thrust required (Omit steps d, e, and f if thrust data not available)

- a. Correct observed airspeed and pressure altitude for instrument and position error.
- b. Determine Mach number from  $V_c - M - H_p$  chart.
- c. Determine  $\delta$ . Interpolate from ICAO Standard Atmosphere table.
- d. Calculate ambient air pressure.

$$P_a = 29.92 (\delta) \quad (\text{in. Hg})$$

- e. Calculate and/or correct engine exhaust total pressure ( $P_{T5}$ ) or ( $P_{T7}$ ) for instrument error and calculate  $P_T/P_a$ .

- f. Enter calibration curve for test engine with  $P_T/P_a$  to obtain corresponding  $F_g/\delta$ .

- g. Correct observed fuel flow for instrument error.
- h. Calculate total fuel flow in lb/hr. Assume density of JP-5 fuel is 6.8 lb/gal and JP-4 fuel is 6.5 lb/gal.
- i. Correct observed OAT for instrument error and determine ambient air temperature from Appendix IV.
- j. Calculate

$$\gamma_0 = \sqrt{\frac{t_a + 273}{288}}$$

- k. Calculate airplane gross weight.
- l. Calculate:

$$\text{referred fuel flow} = \left( \frac{w_f}{\delta} \right)_{\text{Test}}$$

$$\text{referred gross weight} = \left( \frac{W}{\delta} \right)_{\text{Test}}$$

- m. Construct the following graphs for the average value of  $W/\delta$  flown.

Airframe (drag) Characteristics ( $F_g/\delta$  vs.  $M$ )

(omit for airplanes without thrust measurement)

Airplane Characteristics ( $w_f/\delta\sqrt{\theta}$  vs.  $M$ )

- n. On the graph of the variation of referred gross thrust required with Mach number also plot the variation of referred gross thrust available from the acceleration run) at the reference altitude. For airplanes in which inflight thrust was not measured, plot the variation of referred fuel flow available at the reference altitude on the graph of the variation of referred fuel flow required with Mach number. The intersection of these curves defines the maximum airspeed at the reference altitude and at the airplane's gross weight for the test day ambient

temperature. Correction of the thrust available (or fuel flow available) data to standard temperature as discussed in Section VII-B will give the standard day maximum level flight airspeed.

o. From the faired curve of  $w_f/\delta\sqrt{\theta}$  vs. M, read values of referred fuel flow for several values of Mach.

p. Determine the ambient air pressure ratio,  $\delta$ , at the reference altitude from the standard atmosphere tables, Appendix L.

q. Calculate specific range for each value of Mach (step o).

$$S.R. = \frac{661M}{\delta \left( \frac{w_f}{\delta\sqrt{\theta}} \right)} \quad \text{nmi/lb}$$

r. Construct a graph of S.R. vs. M for the value of  $W/\delta$  flown.

s. Using the equation

$$w_f = \delta\sqrt{\theta} (w_f/\delta\sqrt{\theta}) \quad \text{pph}$$

the standard day values of actual fuel flow may be plotted against  $V_o$ .

## SECTION VII-B

### DETERMINATION OF STANDARD DAY MAXIMUM LEVEL FLIGHT AIRSPEED

#### General

The maximum level flight airspeed of a thrust limited jet airplane is determined by the intersection of its thrust available and thrust required curves for one altitude. In Section VII-A, methods for determining standard day thrust required and test day thrust or fuel flow available were determined. The thrust or fuel flow of a jet engine at MIL thrust varies with altitude, airspeed, and ambient temperature. Standard day thrust or fuel flow for the test altitude can be computed by making an incremental correction for the temperature difference from standard based on estimated or previously determined engine characteristics. If the engine is not operating at rated RPM, a similar thrust or fuel flow correction can be made for the nonstandard RPM setting. Corrections to engine thrust and fuel flow data are usually imprecise. However, large corrections to inflight engine parameter measurements should be treated with caution. It is important to realize that more rigorous and tedious data reduction procedures exist.

#### Determination of Standard Day Thrust Available

a. For the several values of MIL thrust available discussed in Section VII-A, calculate standard day referred gross thrust available.

$$\left( \frac{F_g}{\delta} \right)_s = \left( \frac{F_g}{\delta} \right)_t + \left( \frac{d F_g / \delta}{dt} \right) (T_a - T_{a_s})$$

Obtain values of  $\frac{d(F_g/\delta)}{dt}$  from graphs similar to Figure 1 for the appropriate engine.

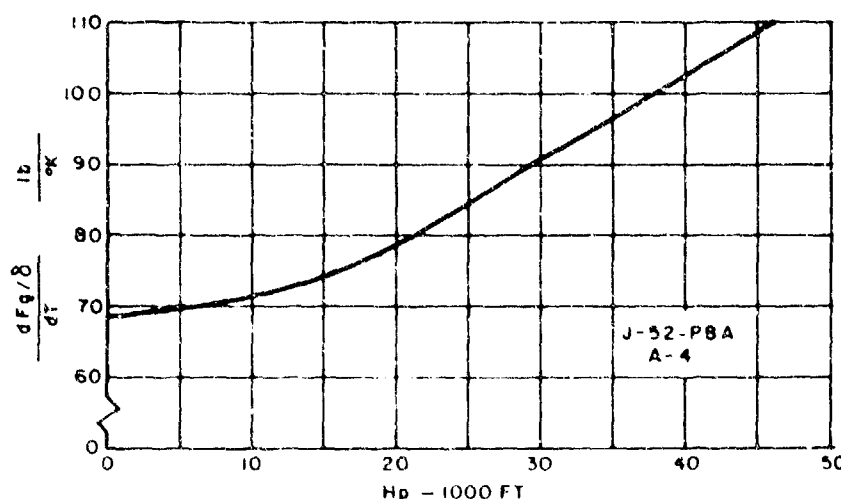


Figure 1  
Engine Performance

b. Plot the standard day referred gross thrust available for each altitude on the graph of referred gross thrust required. Read the standard day maximum Mach number and calculate the corresponding true airspeed.

#### Determination of Standard Day Fuel Flow Available

Tailpipe pressure ratio data from which to compute  $F_g/\delta$  sometimes is not available. Referred fuel flow, based on total temperature and pressure conditions ( $w_f/\delta_T\sqrt{\bar{\theta}_T}$ ), is a function of throttle setting, OAT, and Mach number. Since the throttle setting is constant (MIL), and since Mach effects are generally small, it is possible to plot  $w_f/\delta_T\sqrt{\bar{\theta}_T}$  vs. OAT for one throttle setting. With this graph of referred fuel flow available, the test day data may be corrected to standard day conditions. The following procedure is used:

- a. For MIL thrust data points discussed in Section VII-A calculate

$$\frac{w_f}{\delta_T \sqrt{\theta}} = \left( \frac{w_f}{\delta \sqrt{\theta}} \right)_{\text{Test}} \times \frac{1}{(1 + .2M^2)^4}$$

for each engine separately.

- b. For each engine plot  $\frac{w_f}{\delta_T \sqrt{\theta}}$  for the MIL thrust points vs. OAT. Fair one curve to average data from both engines.

- c. For the reference altitude at the standard temperature, determine the standard day OAT at several Mach numbers between  $.7 V_{\text{mrt}}$  and  $1.1 V_{\text{mrt}}$  for the test day.

- d. With the curve generated in step b, determine referred fuel flow available for each OAT calculated in step c.

- e. Determine standard day referred fuel flow available based on ambient temperature and pressure from

$$\left( \frac{w_f}{\delta \sqrt{\theta}} \right)_s = \left( \frac{w_f}{\delta_T \sqrt{\theta_T}} \right) (1 + .2A^2)^4 \quad (\text{double for 2 engines})$$

- f. Plot the referred fuel flow available (step e) vs. Mach number on the referred fuel flow required curve in Section VII-A. The intersection of the fuel flow required and available curves is the standard day  $V_{\text{mrt}}$ .

## SECTION VII-C

### POWER REQUIRED AND FUEL CONSUMPTION

#### PURPOSE

The purpose of this test is to determine the generalized power required characteristics for a propeller driven airplane and to determine the variation of referred fuel flow with referred shaft horsepower of a turboprop engine for one altitude. These characteristics are required to describe the level flight performance of the airplane.

#### DISCUSSION AND THEORY

##### General

In describing airplane level flight performance, the parameter of greatest interest is specific range, the ratio of true airspeed to fuel flow. Crudely, specific range may be determined by flying the airplane at a constant airspeed and altitude and measuring the corresponding fuel flow. The level flight performance for a large range of airspeeds, altitudes, and gross weights is more efficiently evaluated by investigating separately the airframe characteristics (power required for level flight) and the engine characteristics (fuel flow variation with power and altitude).

Dimensional analysis of the level flight performance of a low speed airplane shows that for a given loading and configuration, the variation of equivalent shaft horsepower required with equivalent airspeed is a single curve which is applicable to all altitudes of airplane operation. This generalization is valid if a constant parasite drag coefficient, propeller efficiency, and effective aspect ratio can be assumed. Although this generalization is valid for the cruise airspeed range of



most propeller driven airplanes, the assumptions fail at very high angles of attack (where effective aspect ratio and propeller efficiency decrease markedly) and at high airspeeds where transonic effects may be observed. This generalization is a convenient method of fairing test data because for a given airplane gross weight we may write

$$(\text{BHP}_{ew} V_{ew}) = K_1 + K_2 (V_{ew}^4)$$

where the subscript, w, signifies values for a standard gross weight. This is the equation of a straight line when plotted in terms of the variables enclosed in parentheses. Figure 1 is an example of such a plot.

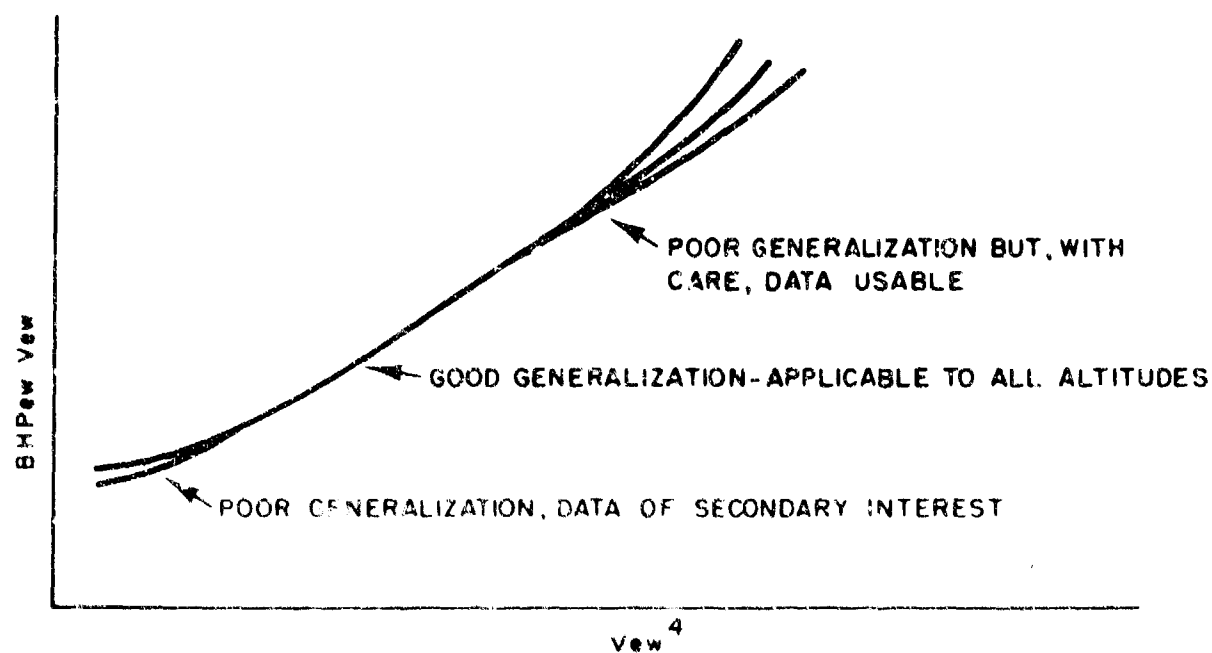


Figure 1  
Generalize Power Required

In the power required test, it is necessary that the power and fuel flow data be representative of stabilized flight conditions. The data must be obtained in smooth air which is free of vertical air currents. At airspeeds greater than the airspeed

for minimum power required, stabilization in airspeed and altitude is easily obtained. At lower airspeeds, data may be obtained by accepting a small vertical velocity (less than 50 fpm) at constant airspeed and later calculating a correction for the incorrect power setting. The power required for level flight at a given airspeed varies with airplane gross weight and flight data must be corrected to a standard gross weight.

#### Turboprop Engine Data Generalization

The relationship between fuel flow and shaft horsepower output for a turboprop engine can be expressed by the following functional relationship:

$$\frac{w_f}{\delta \sqrt{\theta}} = f\left(\frac{\text{SHP}}{\delta \sqrt{\theta}}, M, \text{RNI}\right)$$

Flight tests with current turboprop engines have shown that the Mach (and RNI) effects on the above relationship are weak or not significant for the normal operating envelope of the engine, and thus the data can be generalized using the following simplified relationship:

$$\frac{w_f}{\delta \sqrt{\theta}} = f\left(\frac{\text{SHP}}{\delta \sqrt{\theta}}\right)$$

If this form fails to generalize the data, good generalization may be obtained by using total free static conditions, i.e.,

$$\frac{w_f}{\delta \sqrt{\theta}} = f\left(\frac{\text{SHP}}{\delta \sqrt{\theta}}, \frac{Y}{Y_0}, \frac{M^2}{M_0^2}, \frac{Y}{Y_0}\right)$$

$$\text{and } \sqrt{\theta} = \sqrt{\theta_0} \left( \frac{Y}{Y_0} \right)^{\frac{1}{\gamma-1}} \left( \frac{M^2}{M_0^2} \right)^{\frac{1}{2}}$$

## TEST PROCEDURES AND TECHNIQUES

### Pertinent Particulars

- a. For reciprocating engines, set the power needed to obtain the desired airspeed but keep BMEP high by maintaining a minimum RPM and maximum MAP within the handbook limits.
- b. Use the constant altitude technique at airspeeds faster than the airspeed for minimum drag ( $V_{md}$ ) and the constant airspeed technique at airspeeds slower than  $V_{md}$ .
- c. It is recommended that the  $V_{mrt}$  data point be taken first and the remainder of the data be obtained in order of decreasing airspeeds.
- d. The photopanel will not be used for this flight.

### Preflight Procedures

- a. Review engine limits, single engine operation, and airstart procedures.
- b. Prepare data cards. In flight at each data point, record:
  - Run number
  - Pressure altitude
  - Airspeed
  - OAT/CAT
  - Torque/MAP
  - Fuel Flow
  - Fuel remaining
  - RPM/EGT
  - Configuration

## Flight Procedures

a. Perform a series of stabilized, level flight runs at the assigned altitude using various power settings. Although a particular sequence of data is not required, it is recommended that the  $V_{mrf}$  point be obtained first and the remaining data be obtained in order of decreasing airspeeds.

b. For the constant altitude method, set the maximum cruise power setting and allow the airplane to stabilize in airspeed while maintaining a constant altitude. Several minutes at constant power and altitude are necessary to obtain good stabilization of airspeed. When certain of stabilization, record data. Reduce power setting to that estimated to give the next desired airspeed and wait for airspeed to stabilize.

c. For the constant airspeed method, establish a trimmed, constant airspeed and adjust the power until level flight is achieved at the test altitude. A bracketing technique, similar to the one described in Section II and demonstrated on the performance demonstration flight, should be used. If level flight cannot be achieved after several power adjustments, acceptable data can be obtained with the airplane in a climb or descent of 50 fpm or less. When this condition is established, record data and commence a 5 min run at constant airspeed and power setting. After 5 min record data again to obtain the average rate of climb and average power for this interval. The data will later be corrected for this condition.

d. The airspeed at which the airplane exhibits minimum acceptable flying qualities for maximum flying should be recorded for later use in determining the recommended endurance airspeed.

## DATA REDUCTION

### General

Follow the order of the data reduction presented below in setting up the data reduction form. Normal data reduction is by Hewlett-Packard Computer program but the first and last data points must be done manually.

### Manual Data Reduction - Fuel Flow and Generalized Power Required for Turboprop

#### Airplane

- a. Correct observed pressure altitude for instrument and position error.

$$H_{p_c} = H_{p_o} + \Delta H_{p_{ic}} + \Delta H_{p_{pos}}$$

- b. Correct observed airspeed for instrument and position error.

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

- c. Calculate equivalent airspeed. Use Appendix II for compressibility correction.

$$V_e = V_c + \Delta V_c$$

- d. Correct observed OAT for instrument error.

- e. Determine M by use of  $M - V_c - H_p$  chart.

- f. Determine ambient air temperature ( $T_a$ ) using Appendix IV (Assume  $K = 0.95$ ).

- g. Determine  $1/\sqrt{\sigma}$  by entering Galcit chart with  $H_p$  and  $T_a$ .

- h. Determine  $\sigma$  from PTM standard atmosphere tables and  $H_{p_c}$ .

i. Compute  $\sqrt{\theta}$  from  $\sqrt{\theta} = \left( \frac{T_a + 273^\circ}{288} \right)^{1/2}$

j. Compute  $\delta/\sqrt{\theta}$

k. Correct observed torque and RPM for instrument error for one engine.

l. Compute test shaft horsepower

$$SHP_t = (K) (Q) (RPM)$$

where K is the engine torquemeter constant, Q is torque or pressure (psi or ft/lb) and RPM is propeller RPM.

m. Repeat steps k and l for the other engine.

n. Compute total test shaft horsepower (sum of both engines).

o. Compute airplane test gross weight ( $W_t$ ).

p. For constant airspeed data, compute rate of climb. Assume that the altimeter instrument and position error corrections remain constant over the test interval.

$$R/C = \frac{\Delta H}{\Delta t} \frac{P_o}{P_o} \text{ fpm } (+50 \text{ fpm maximum})$$

or

$$R/C = \frac{\Delta H}{\Delta t} \frac{P_o}{P_o} \left( \frac{T_{a \text{ test}}}{T_{a \text{ std}}} \right) \text{ fpm } (T_{a \text{ std}} \text{ at test } H_p)$$

q. Calculate shaft horsepower correction for rate of climb.

$$\Delta SHP_{R/C} = \frac{R/C (W_t)}{13,000} \eta_p$$

Assume  $\eta_p = 0.85$  (an average value)

r. Compute shaft horsepower required for stabilized, level flight for the test conditions.

$$SHP_{rt} = SHP_t + \Delta SHP_{R/C}$$

s. Compute equivalent shaft horsepower required.

$$SHP_e = \frac{SHP_{rt}}{1/\sigma}$$

t. Compute ratio of standard gross weight to test gross weight  $W_s/W_t$  and the correction factors  $(W_s/W_t)^{1/2}$  and  $(W_s/W_t)^{3/2}$ .

u. Correct equivalent shaft horsepower required and equivalent airspeed to the standard gross weight.

$$SHP_{ew} = SHP_e \left( \frac{W_s}{W_t} \right)^{3/2}$$

$$V_{ew} = V_e \left( \frac{W_s}{W_t} \right)^{1/2}$$

v. Compute  $V_{ew}^4$  and  $V_{ew} \times SHP_{ew}$ .

w. For each engine compute referred shaft horsepower =

$$\frac{SHP_t}{\delta \sqrt{\theta}}$$

x. For each engine, correct observed fuel flow for instrument error and compute referred fuel flow.

$$\frac{W_f}{\delta \sqrt{\theta}}$$

Manual Data Reduction - Fuel Flow and Generalized Power Required -  
Reciprocating Engine Airplane

- a. Perform steps a through g listed under turboprop data reduction.
- b. Correct observed RPM and MAP for instrument error.
- c. Determine chart shaft horsepower from engine chart. The chart method is not as accurate as the engine torquemeter which should be used if available.
- d. Obtain standard day ambient air temperature for test altitude from standard atmosphere tables.

$$\left( T_{a_s} \right)_k = T_{a_s} + 273 \quad ^\circ K$$

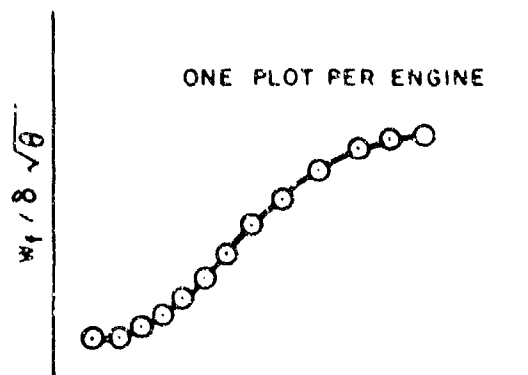
- e. Correct observed CAT for instrument error.
- f. Compute test day shaft horsepower

$$SHP_t = SHP_c \sqrt{\frac{T_{a_s}}{CAT_t}} \quad ^\circ K$$

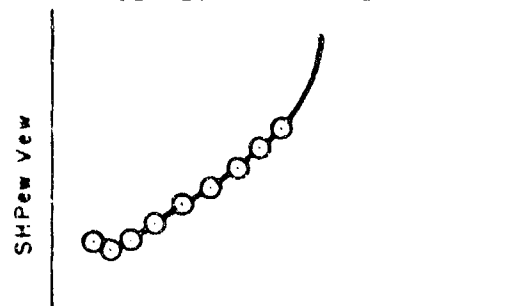
- g. Perform steps o through v.



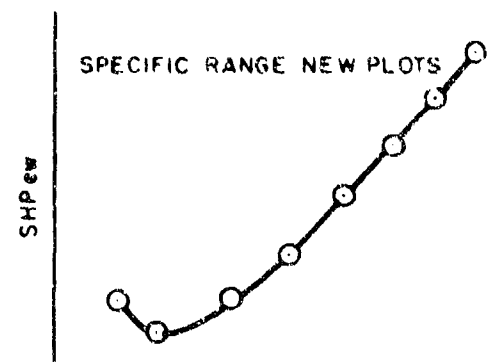
# Graphical Presentation of Results



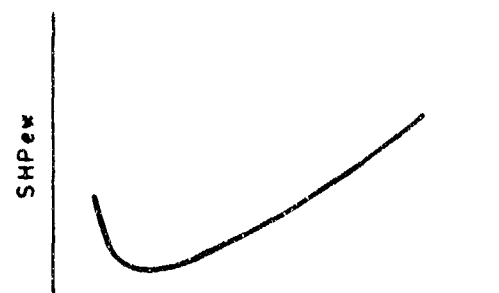
FUEL FLOW CHARACTERISTICS



GENERALIZED POWER REQUIRED



GENERALIZED POWER REQUIRED



COMPUTED POWER REQUIRED

SECTION VIII

MANEUVERING PERFORMANCE

## REFERENCES

### Section VIII

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#### Chapter V of Theory

2. Dommasch, Airplane Aerodynamics, Section 11:5
3. Hurt, Aerodynamics for Naval Aviators, pp. 176-182
4. Petersen, Aircraft and Engine Performance, Chapter 9
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## MANEUVERING PERFORMANCE

### INTRODUCTION

Aircraft maneuverability can best be defined as the ability to change direction and/or magnitude of the velocity vector. While this definition is accurately described and clearly understood by most pilots, the best method to optimize aircraft maneuverability is more difficult to define due to the complexity and dynamic nature of the combat maneuvering situation. Pilot selection of turning maneuvers, in terms of the change of direction achieved as a function of energy state, is strongly dependent on many factors, not the least of which is the energy state and position of the opponent aircraft or his weapons system. Assuming parameters such as pilot proficiency, roll performance, flying qualities, field of view, and fuel management techniques are equal, we have learned from experience that the offensive advantage is available to the airplane which initially has and can maintain the highest energy level. In addition, the airplane with the highest energy level has a better opportunity to engage and disengage at will. To achieve or deny this advantage depends primarily on two factors:

- a. The weapons system capability.
- b. The aircraft maneuvering performance.

Quantitatively, the weapons system capability can be depicted by release or firing envelopes, and once these initial conditions are known, the optimization problem becomes one of maneuvering into the effective weapons envelope. Such maneuvering is dependent upon the ability of the pilot to control turn, altitude, airspeed, and acceleration characteristics.

## PURPOSE

The purpose of this section is to discuss aircraft maneuverability in terms of sustained and instantaneous g available, turn rate, turn radius, and the interface of these performance parameters with specific excess power characteristics.

## DISCUSSION AND THEORY

A measurement of the relative tactical value of high performance airplanes is the maximum normal acceleration available at a given airspeed and altitude. It is important to distinguish between sustained g available which can only be achieved while maintaining a constant energy level, and instantaneous g available which can be achieved while gaining or losing energy at a given rate.

### Sustained g Available

Sustained g available is determined by level flight turning performance. When an airplane is in straight, level flight, it develops lift equal to its weight. When this airplane is banked into a level turn, the lift increases so that the vertical component of lift remains equal to the airplane weight. Because the lift has increased, the drag has increased also, and the engine thrust must be increased to maintain the airspeed. The maximum bank angle for which both airspeed and altitude can be maintained is a function of both the aerodynamic and thrust characteristics but it is usually limited by the thrust available. If this maximum bank angle (or normal acceleration) is measured at several airspeeds, the sustained turning performance is determined.

### Instantaneous g Available

Instantaneous g available is determined by the variation with Mach number of the normal acceleration for onset, tracking, and limit buffet. As an airplane is maneuvered more and more vigorously, a normal acceleration will be attained at which buffeting of the airframe is first detected (onset buffet). As the normal acceleration is increased further, the buffeting increases in intensity until it becomes so severe as to prevent satisfactory accomplishment of an offensive maneuver (tracking buffet). When the normal acceleration is increased still more, a maximum buffet acceptable for defensive operation of the airplane is observed (limit buffet). The variation with Mach number of the normal acceleration at which a specified level of buffet is observed is known as a buffet boundary, and for the specified condition, this buffet boundary defines the operational V-n envelope of the airplane (Figure 1).

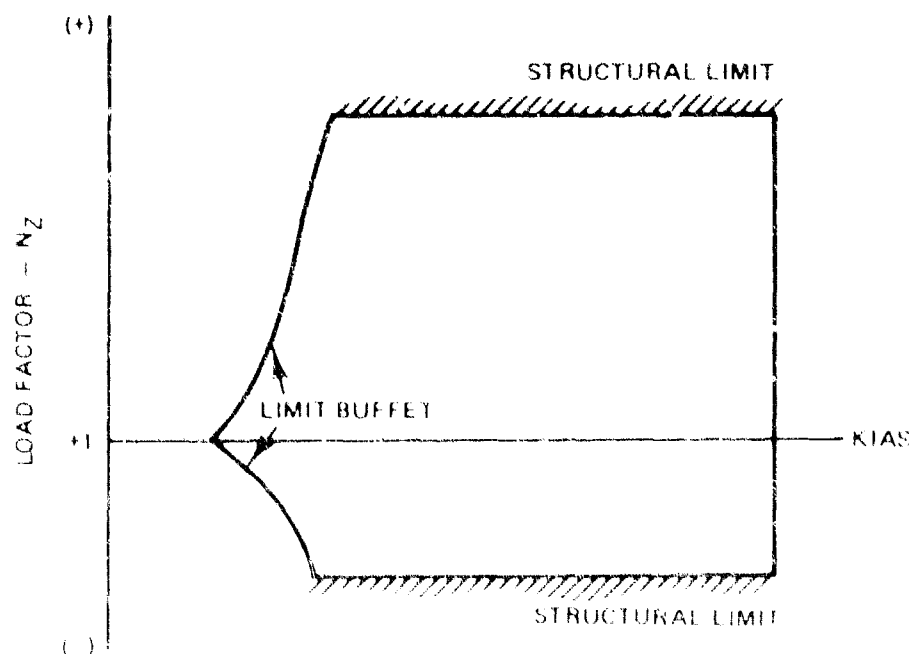


Figure 1  
Typical V-n Diagram

For most flight conditions, the limit buffet level may be defined by: the structural limit, aerodynamic limit ( $C_{L_{max}}$ ), longitudinal control limit, or maximum load factor for acceptable flying qualities of the airplane.

In all cases, buffet levels must be defined in relation to the mission of the airplane. Some airplanes (transports, straight-wing trainers, etc.) may have no offensive or defensive maneuvers related to their mission which bear upon instantaneous turning performance. These airplanes may exhibit little or no buffeting prior to reaching their aerodynamic limit (stall); or they may be prohibited from buffeting flight due to comfort or structural safety reasons. Onset and limit buffet may be coincident; and there may be no relatable or applicable tracking buffet level for an airplane.

Airframe buffet is associated with the disturbance of airflow somewhere on the airplane's structure. The intensity of the disturbance will vary with angle of attack and Mach number. For a given Mach number, a specified buffet level will occur at the same angle of attack (or lift coefficient). Figure 2 depicts buffet characteristics typical of a supersonic fighter airplane.

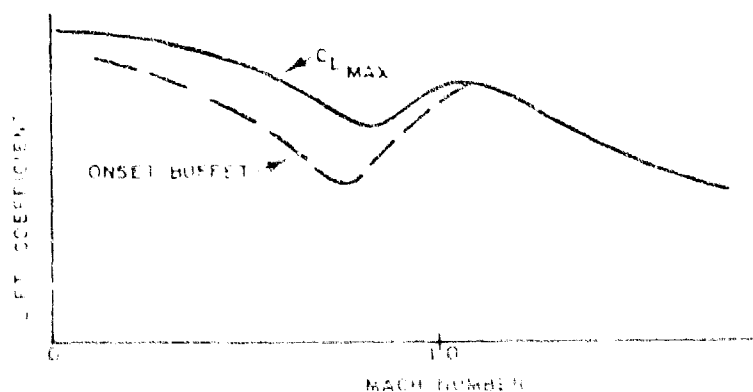


Figure 2  
Buffet Characteristics of a Fighter Airplane

### Rate and Radius of Turn

For a given normal acceleration,  $n_z$ , at a given altitude, the radius of turn is given by

$$R = \frac{V_T^2 \text{ (fps)}}{g \sqrt{N_z^2 - 1}} \quad \text{ft}$$

and the rate of turn is given by

$$\omega = \frac{57.3 V_T}{R} = \frac{57.3 \sqrt{N_z^2 - 1}}{V_T} \quad \text{deg/sec}$$

Turn rate is perhaps the most tactically significant of all the parameters used to measure turn performance. It measures rate of change of flight path direction as a function of sustained or instantaneous  $g$  available and airspeed. In the air-to-air maneuvering environment, the relative turn rate capability of two adversaries will directly influence the tactics used by each combatant. Of course, turn radius (which also is a function of TAS and  $g$  available) must be considered when determining the mission suitability of an airplane. An example of the use of turn radius is the gun tracking solution in which a certain amount of lead (smaller turn radius) is required.

### Tactical Considerations

Another important consideration in maneuvering performance is the use of the vertical gravity vector to effectively add to the airplane's lift vector. The effect of increasing bank angle on turn rate is shown in Figure 3. The requirements of the mission will dictate acceptable load factors, turn rates, and radii. Consideration must be given to required maneuvers, visual contact distances, performance of potential adversaries, pilot load factor tolerance, and the expected altitude at



which various maneuvers will be conducted. The influence of flying qualities, energy level maintenance, and weapons systems must also be considered when evaluating the turning performance of an airplane.

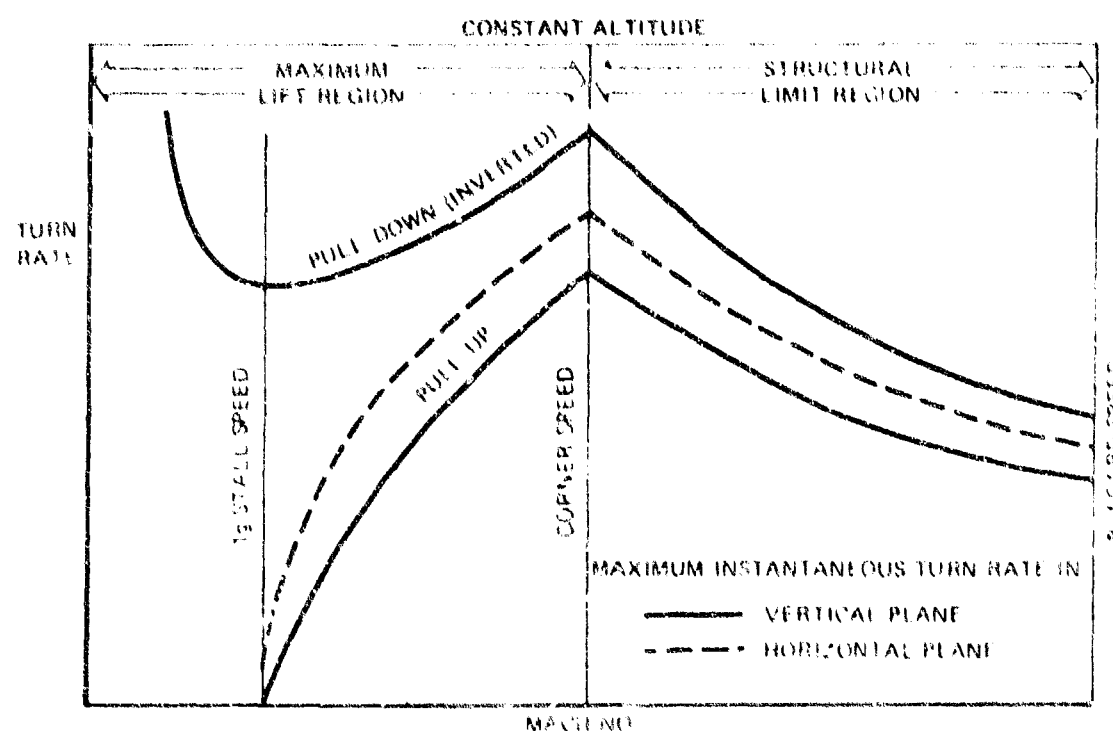


Figure 3  
Three-Dimensional Turn Comparison

### Wing Loading and Thrust-to-Weight Considerations

Assuming there is no appreciable lift due to thrust, the relationship of wing loading (W/S) on turn performance may be expressed

$$L = C_L \frac{1}{2} \rho V_T^2 S$$

where

$$L = W \cdot n_z \text{ (available instantaneous)}$$

Rearrangement of this expression results in

$$n_{z \text{ avail inst}} = \frac{C_L \rho V_T^2}{2 (W/S)}$$

At any given altitude and airspeed, instantaneous g available is solely dependent upon wing loading (W/S) and the airplane lift coefficient ( $C_L$ ).

The effect of thrust-to-weight ratio (T/W) on turn performance may be expressed in stabilized, unaccelerated flight as

$$T = D$$

$$T = (D/L) L$$

$$\text{where } L = W \cdot n_z \text{ (available instantaneous)}$$

Rearrangement of this expression results in

$$n_{z \text{ avail sust}} = (T/W) (C_L / C_{L_D})$$

The sustained  $n_z$  available is directly proportional to the thrust-to-weight ratio (T/W).

Sustained g available is also a function of aerodynamic (airframe) characteristics through its relationship with the lift-to-drag ratio ( $C_L/C_D$ ).

A graphical presentation of the variation in sustained and instantaneous turn rate with wing loading and thrust-to-weight ratio is shown in Figures 4 and 5, respectively. At a constant altitude and Mach, the sustained turning performance in Figure 4 is increased with higher thrust-to-weight ratios and improved aerodynamic characteristics (reduced wing loading). However, at the same constant altitude and Mach, the instantaneous turning performance in Figure 5 is increased with lower wing loading and is unaffected by the thrust-to-weight ratio.

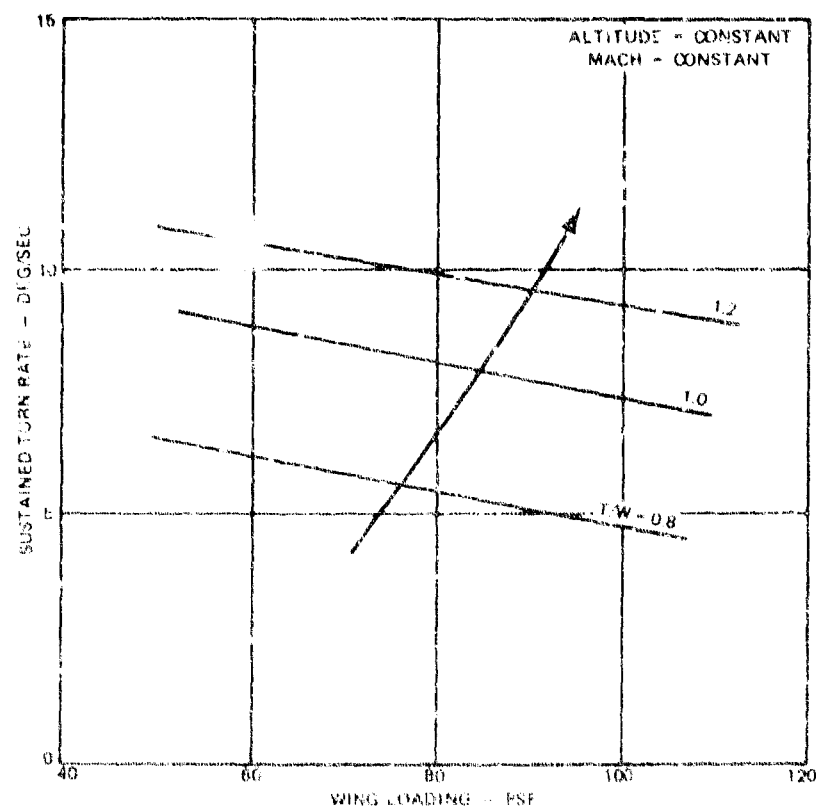
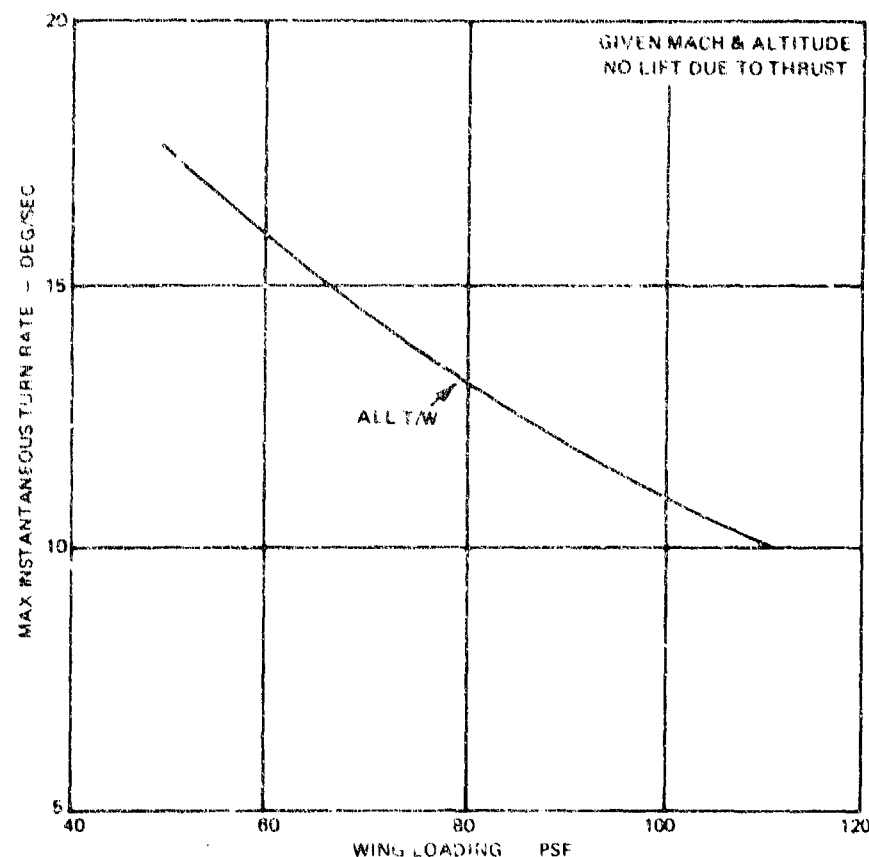


Figure 4  
Effect of (T/W) on Sustained Turning Performance



**Figure 5**  
Effect of (W/S) on Instantaneous Turning Performance

### Energy Maneuverability

Thus far, aircraft maneuverability has been explained in terms of directional change (turn) only, but magnitude changes of the velocity vector are equally important. From the specific energy equation

$$E_h = h + \frac{v^2}{2g}$$

it can be deduced that SUSTAINED maneuverability is closely related to specific energy levels in terms of altitude and airspeed. By observing the correlation of energy rate variation with turning performance, tactical maneuverability can be related to the amount of energy possessed and how well that energy is managed.

A comparison of specific excess power characteristics ( $P_s$ ) can be used to determine the operational flight envelope in which an airplane can gain or lose energy more quickly than an adversary. The amount of energy gained or lost is directly related to maintaining maneuvering position against an adversary aircraft. The envelope for optimum energy maneuverability may be represented on an altitude vs. Mach number (H-V) diagram containing contour lines of constant specific excess power (Figure 6). Energy gain per unit time is maximum at the points where the lines of constant energy altitude ( $E_h$ ) are tangent to the lines of constant  $P_s$ . Flight along the maximum energy path will allow the most efficient energy conversion of altitude and airspeed.

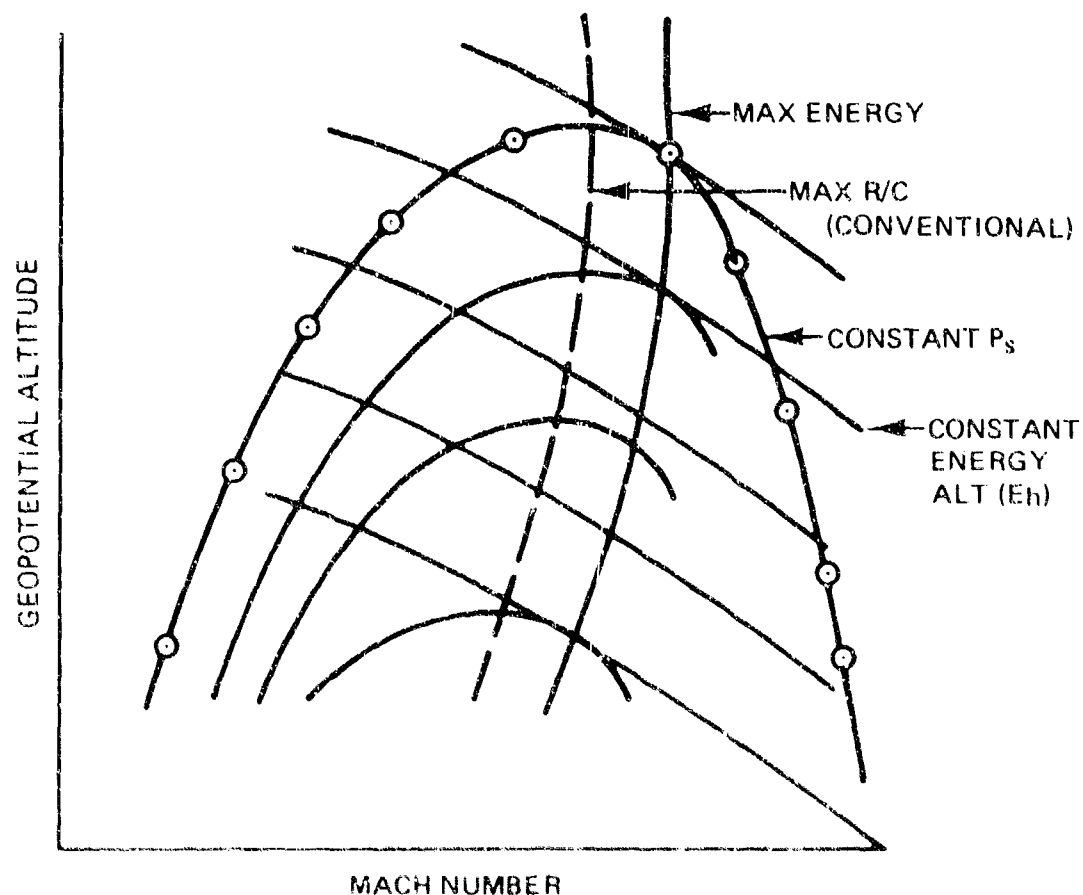


Figure 6  
H-V Diagram

The actual technique to determine areas of numerical specific excess power advantage involves the overlay of constant  $P_s$  lines of one aircraft over another. This technique is illustrated in Figure 7. The shaded area represents the flight region where aircraft A has maneuvering advantage over aircraft B. One step further in the analysis of energy maneuverability would be to determine contours of constant differential specific excess power ( $\Delta P_s$ ) as shown in Figure 8, the shaded area represents the flight region where aircraft A has its greatest potential for maneuvering advantage. These energy rate diagrams can also be used to determine areas of sustained maneuvering advantage as a function of g.

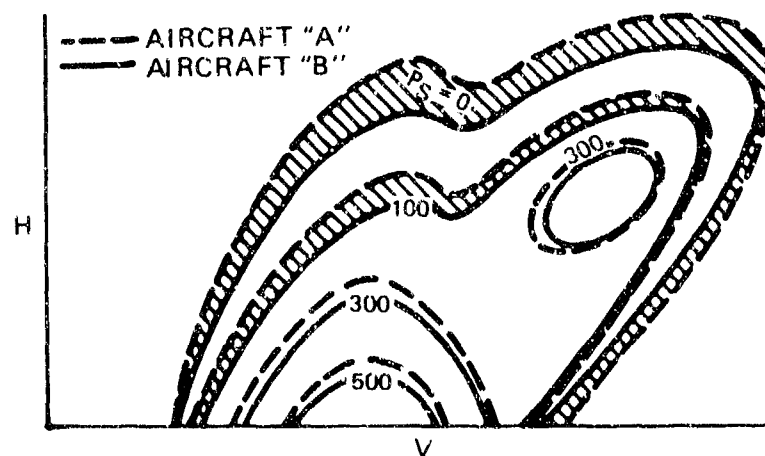


Figure 7  
Specific Excess Power Overlays

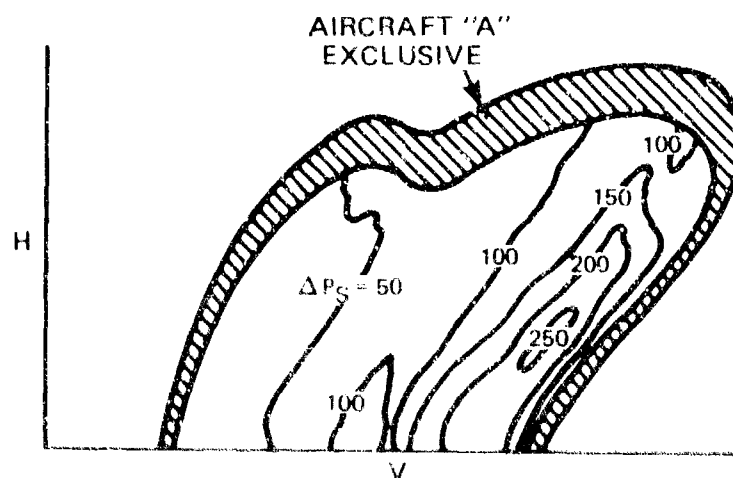


Figure 8  
Differential  $P_s$  Contours

Since energy rate diagrams ( $P_g$ ) are not limited to  $1g$ , a maximum maneuver corridor may be represented by an area between the maximum energy path at  $1g$  and the maximum energy path at the aerodynamic or structural  $g$  limit. By employing the overlay technique at various  $g$  levels, areas of tactical advantage for all flight conditions can be determined between adversary aircraft (Figure 9).

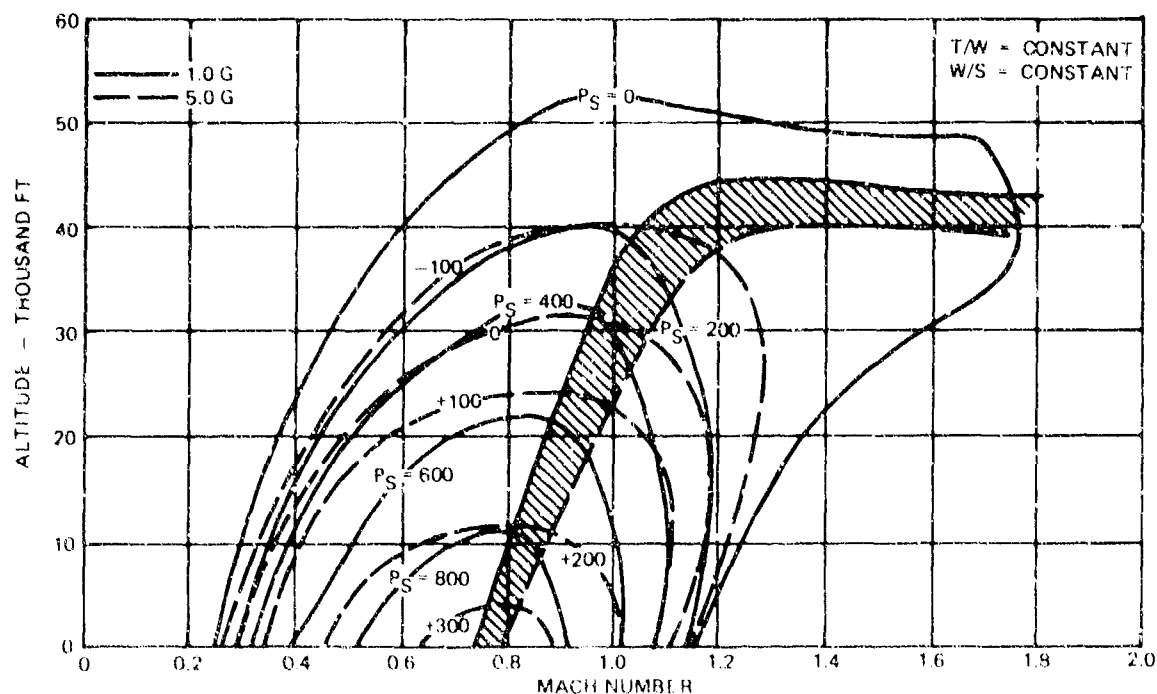


Figure 9  
Maneuver Corridor

The energy rate variation from a turn rate of zero (1g) to the maximum turn rate achievable (aerodynamic or structural g limit) characterizes the complete spectrum of energy maneuverability for any given airspeed and altitude combination. The typical variation illustrated in Figure 10 provides three useful measures of maneuvering performance. The 1g energy rate indicates a measure of acceleration and/or climb performance. The turn rate at zero energy rate ( $P_g = 0$ ) is that which can be sustained without energy loss. The maximum instantaneous turn rate corresponds to the highest energy loss rate. By improving the lift characteristics and/or reducing the drag characteristics, the energy rate variation with turn rate will be less pronounced and a general fanning out of the curve will result (Figure 11).

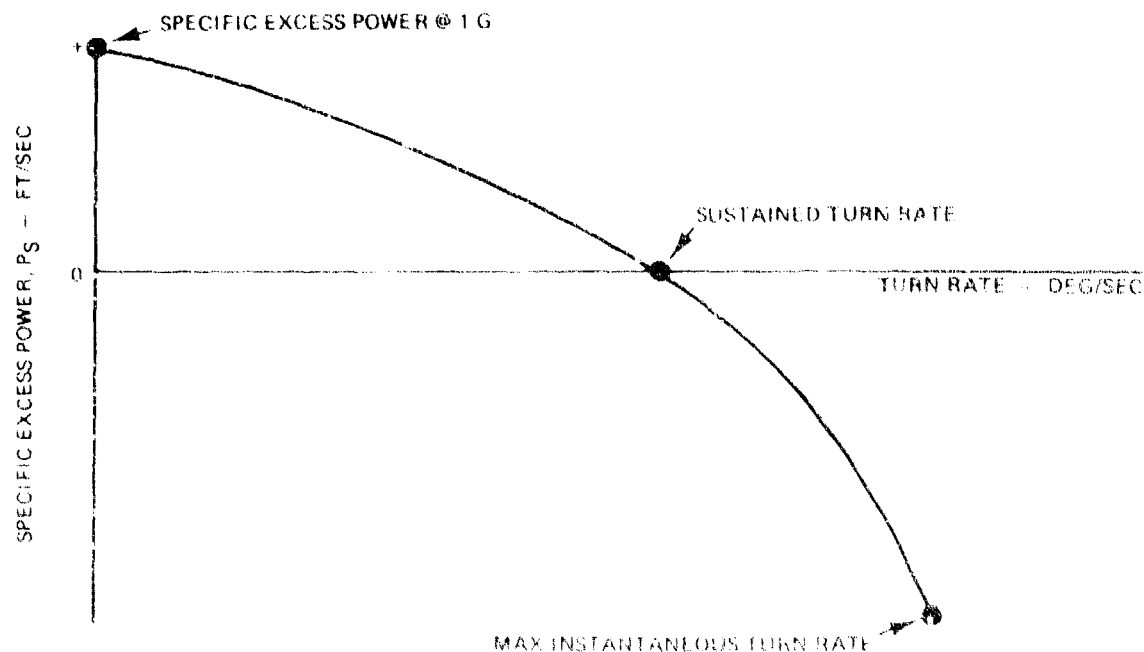


Figure 10  
Energy Maneuverability



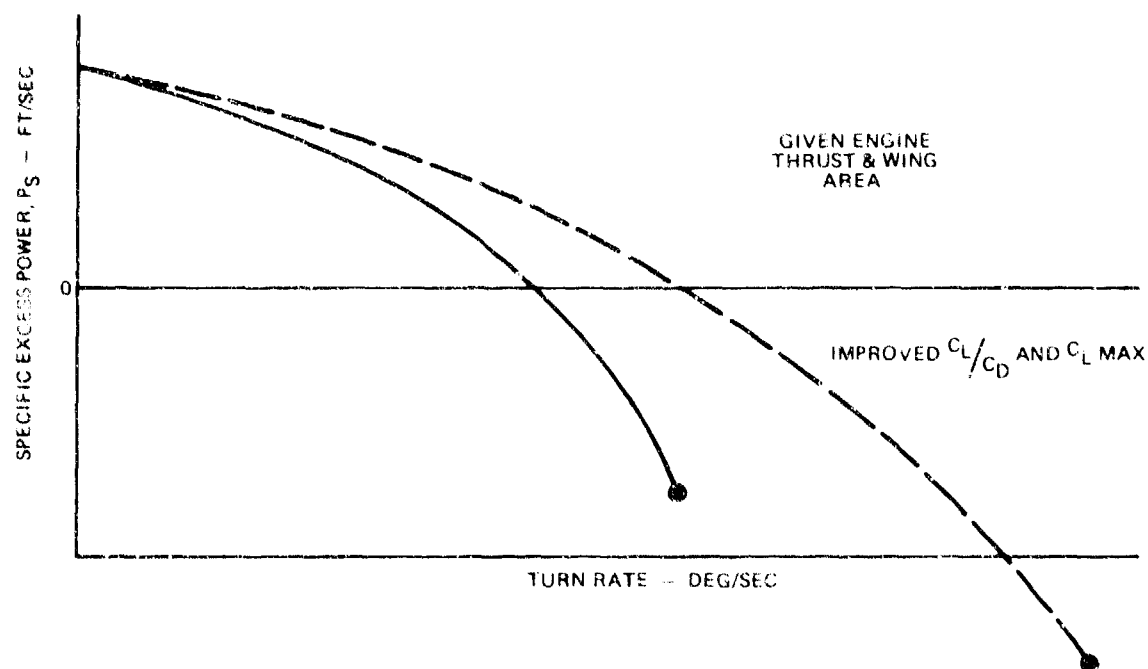


Figure 11  
Influence of Improved Aerodynamics

The effect of wing loading on the energy maneuverability curve is illustrated in Figure 12. The greatest benefit of reduced wing loading occurs at and near to the maximum instantaneous performance limit. The influence of thrust-to-weight on the energy maneuverability curve is illustrated in Figure 13. Increasing the thrust-to-weight ratio has no significant effect on the instantaneous turn performance, but significantly improves the lg and sustained maneuvering capabilities.

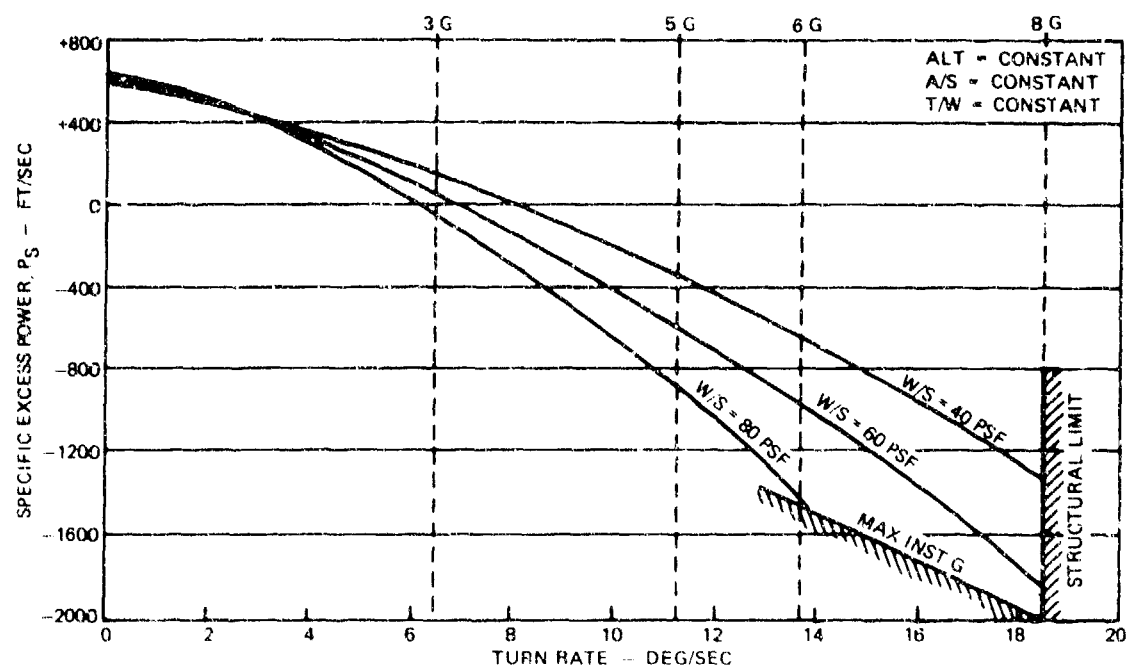


Figure 12  
Effect of Reduced Wing Loading

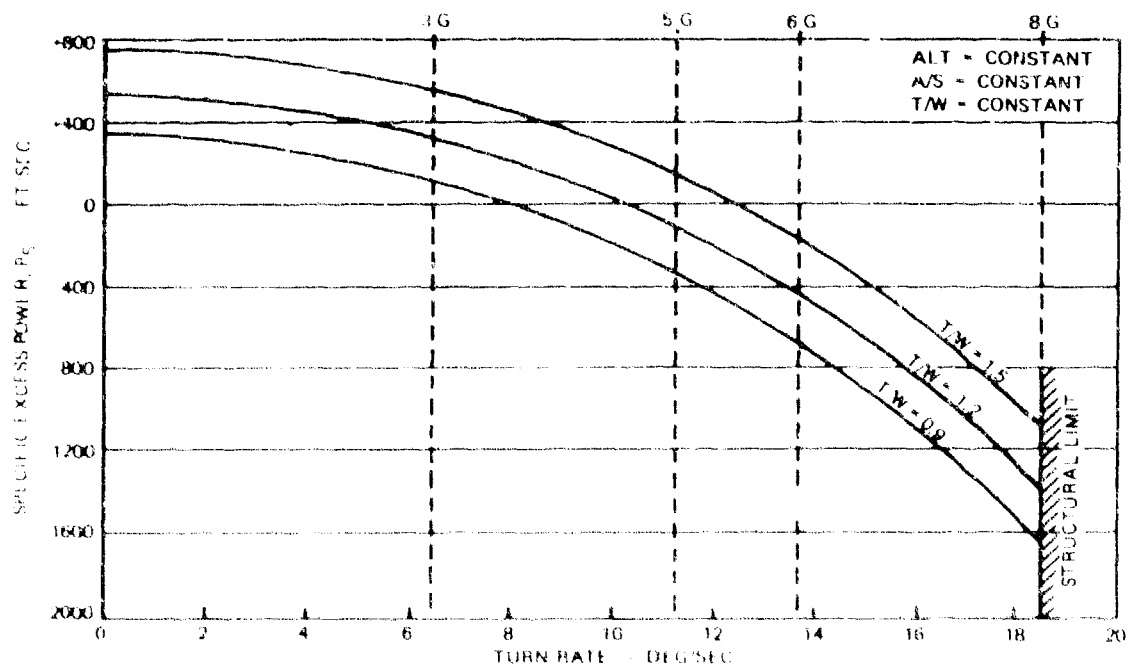


Figure 13  
Effect of Improved Thrust-to-Weight

### Maneuvering Performance Superiority

A conceptual evaluation of the relative maneuvering performance of two aircraft can be made by comparing their energy rate variations with turn rate characteristics as indicated in Figure 14. The friendly aircraft depicted has a margin in turn rate regardless of the energy rate and/or turn rate employed by the threat aircraft. The specific excess power margin at zero turn rate provides the friendly aircraft with the advantage of engaging or disengaging at will. The sustained turn rate margin enables the friendly aircraft to maintain an energy level above that of the threat aircraft. The maximum instantaneous turn rate and deceleration margins allow turning superiority for the friendly aircraft in defensive maneuvering situations.

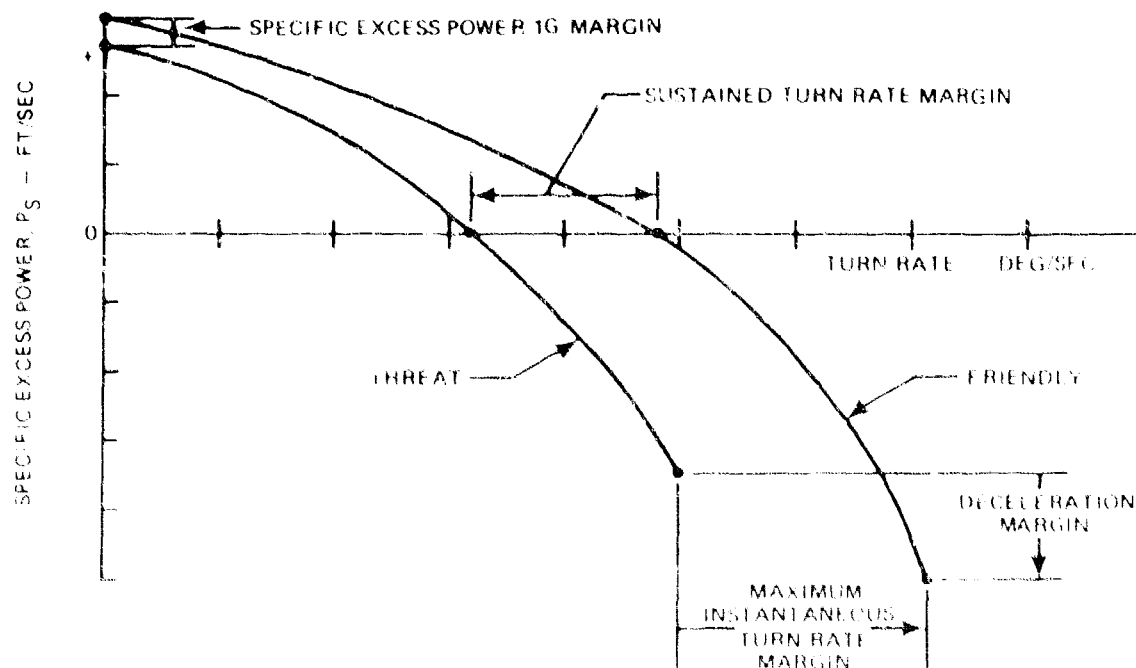


Figure 14  
Energy-Maneuverability Comparison

In summary, this analysis of aircraft maneuverability, particularly when supplemented by a specific excess power comparison, can be an extremely useful tool to the tactician seeking to determine flight envelopes of advantage or disadvantage. His ability to translate the results of an energy maneuverability analysis into tactical maneuvers will contribute to mission success. Of course, the ultimate measure of a pilot's achievement is still how well he takes advantage of his aircraft's capabilities, whatever they may be.

#### ANALYSIS OF SUSTAINED VS. INSTANTANEOUS CAPABILITIES

The major factors influencing sustained and instantaneous turning performance have been discussed. A combined graph such as Figure 10 cannot be constructed without  $P_s$  data, but much can be learned from analysis of a graph like Figure 15. A deficiency in sustained turning capability must be due to low  $T/W$  ratio and/or  $C_L/C_D$  characteristics, while a deficiency in (limit) instantaneous  $n_z$  available must be caused by low values of  $C_{L_{max}}$  and/or excessive wing loading. The onset of buffet may occur at low  $n_z$ , indicating airflow separation. In the example shown: (1) a low  $T/W$  ratio may limit sustained capabilities, (2) early onset of buffet may cause degraded flying qualities and high energy losses during maneuvering flight, while (3) the aerodynamic and structural  $n_z$  limits may provide adequate turn rates and radii for all required maneuvers. A contractor might correct the deficiencies by using maneuvering leading edge slats to improve  $C_L/C_D$  and delay airstream separation (onset buffet), and by increasing engine thrust to improve sustained capabilities.

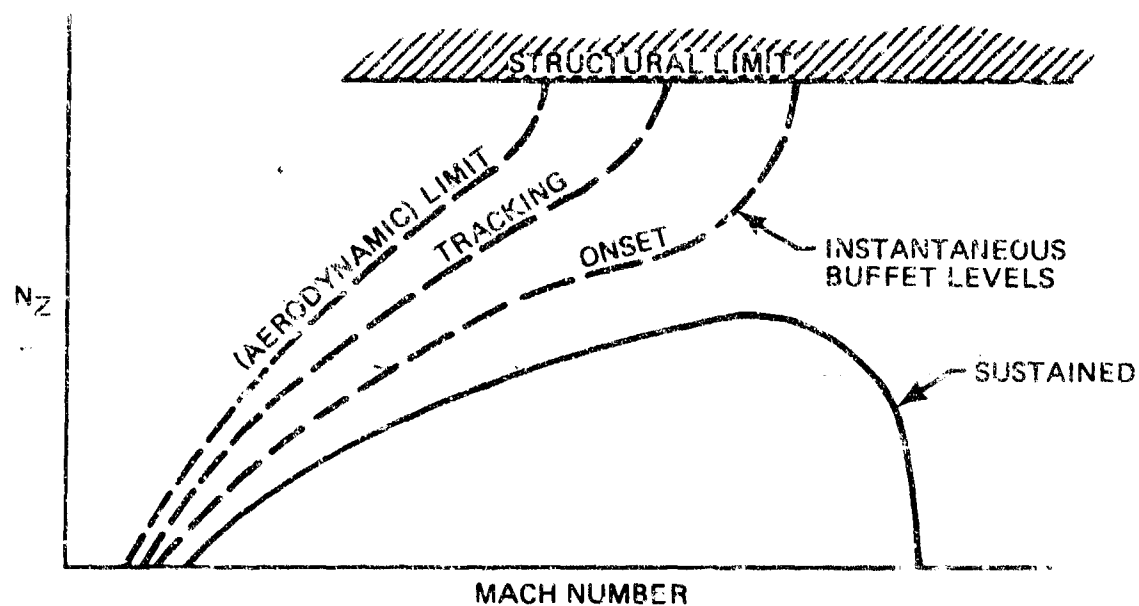


Figure 15  
Engine-Airframe Compatibility

### LEVEL FLIGHT TURNING PERFORMANCE

#### TEST PROCEDURES AND TECHNIQUES

##### Preflight Procedures

a. Prepare pilot's data card. In flight, in each stabilized turn, record the following cockpit data.

Airspeed.

Pressure altitude.

Normal acceleration.

Fuel used or fuel remaining.

OAT.

b. On the data card make a brief table of the variation of normal acceleration with bank angle and have a blank plot of  $n_z$  vs.  $V$  (or  $M$ ) to plot rough data as collected.

### Flight Procedures

a. Obtain  $V_{nrt}$  point first. Record the actual reading of the sensitive accelerometer at this 1g point. Any disparity from 1.0g should be noted as  $n_z$  tare.

b. Set a bank angle for the next data point. A  $30^\circ$  bank angle is recommended. Assume that stable equilibrium conditions can be obtained and use the constant altitude test technique. Maintain the  $30^\circ$  bank angle, test altitude, and MIL thrust until equilibrium airspeed is attained. Trim out stick forces as speed decreases. Record data when equilibrium conditions are achieved for 5 sec.

c. The constant altitude technique may be repeated for greater bank angles and higher normal acceleration until the maximum level flight, normal accelerations has been achieved. At bank angles greater than about  $60^\circ$ , the high variation of  $n_z$  with bank angle may make stabilization very difficult.

d. If desired, the constant airspeed technique may be used for these difficult test points. For airspeeds less than that for peak normal acceleration, the test technique will be that for unstable equilibrium conditions. The constant airspeed technique must be used. Fly smoothly and make small corrections as shown during the performance demo flight until equilibrium conditions have been achieved for at least 5 sec, then record data. Satisfactory data may be obtained within 1,000 ft of the desired test altitude.

e. Some airplanes (mainly propeller driven) exhibit different turning characteristics in left and right turns. The direction of turn should be alternated to establish whether the characteristics vary.

## DATA REDUCTION AND PRESENTATION

### General

The variation of maximum normal acceleration in level flight with Mach number will be presented for a standard airplane gross weight. From this characteristic (faired test data), the variation of radius of turn and rate of turn with Mach number will be computed.

### Computer

Usually, the appropriate Hewlett-Packard program will be used for data reduction. This program corrects the data to standard day temperature and standard gross weight conditions. Review necessary input parameters during preflight planning.

### Manual

- a. Correct observed airspeed for instrument and position error.
- b. Determine Mach number from the  $M - V_C - H_P$  chart.
- c. Correct observed normal acceleration for instrument error and for  $n_z$  tare.

Assume the tare correction is constant throughout the range of the sensitive accelerometer and apply it to all observed values.

$$n = n_o + n_{ic} + n_z \text{ tare}$$

- d. Compute airplane gross weight,  $W$ .

- e. Compute normal acceleration for the standard weight.

$$n_s = n_t \left( \frac{W_t}{W_s} \right)$$

- f. Construct a graph of the variation of standard weight normal acceleration with Mach number or velocity. From the faired curve, read values of  $n_s$  at various airspeeds.

- g. Correct observed OAT for instrument error to determine  $T_a$  (if necessary).

- h. Compute true airspeed

$$V_T = 65.8 M \sqrt{T_a} (^{\circ}K) \quad \text{fps}$$

- i. Calculate radius of turn

$$R = \frac{V_T^2}{g \sqrt{n_s^2 - 1}} \quad \text{ft}$$

- j. Calculate rate of turn

$$\omega = \frac{57.3 V_T}{R} = \frac{57.3 g \sqrt{n_s^2 - 1}}{V_T} \quad \text{deg/sec}$$

- k. Present the following graphs for each altitude tested:

Variation of sustained normal acceleration with Mach number/KCAS.

Variation of sustained radius of turn with Mach number/KCAS.

Variation of sustained rate of turn with Mach number/KCAS.



## BUFFET BOUNDARY

### TEST PROCEDURES AND TECHNIQUES

#### Preflight Procedure

a. Review stall/spin characteristics and recovery techniques. Prepare pilot's data card. For each altitude at several speeds, record the following cockpit data:

Mach number.

Pressure altitude.

Normal acceleration at onset buffet.

Normal acceleration at tracking buffet.

Normal acceleration at limit buffet.

Fuel used or fuel remaining.

AOA, remarks, etc.

#### Flight Procedure

a. The maneuver best suited to obtain the data required for this test is the wind-up turn in which normal acceleration is steadily increased while maintaining a constant Mach number. Stabilize at the desired airspeed with a mission-related power setting (climb may be necessary) and slightly above the assigned altitude. Smoothly roll into a turn and steadily increase  $n_z$ . When the Mach number shows a tendency to decrease, use more bank angle to increase nose down attitude. Note the normal accelerations when onset buffet and tracking buffet occur. Continue to increase  $n_z$  until limit buffet or the structural limit of the airplane is attained. Recover and record data. Satisfactory data may be obtained at any altitude within 1,000 ft of the assigned altitude. If possible, record altitude for each buffet level data point. This will provide the most accurate data and minimize altitude effects.

b. Repeat this procedure for other airspeeds until the buffet boundary for the entire maneuvering airspeed range is defined. For some Mach numbers it may be necessary to repeat the wind-up turn, obtaining onset and tracking buffet in the first turn, and limit buffet in the second. The use of augmented thrust may be useful in some airplanes to maintain speed or to reduce altitude loss during the maneuvers, but should be recorded and noted on the graphical data plot.

## DATA REDUCTION AND PRESENTATION

### General

The Hewlett-Packard programs will usually be used for reducing these data. Output is in the form of lift coefficient ( $C_L$ ) for the various buffet levels vs. Mach number. Once the variation of  $C_L$  with Mach number is smoothly faired for each buffet level,  $n_z$  vs.  $M$  curves can be constructed using the formula

$$n_{z \text{ inst}} = \frac{C_L \rho V_T^2 S}{2W}$$

where  $S$  is the wing area in ( $\text{ft}^2$ ). This procedure may indicate that  $C_L$  curves generalize between altitudes, which would allow generation of many curves after testing only two or three altitudes.

### Manual

- Correct observed Mach number for instrument and position error.
- Correct observed normal acceleration for instrument error, and for  $n_z$  tare.
- Compute airplane gross weight.
- Compute normal acceleration for a standard weight.

$$n_s = n \left( \frac{W}{W_s} \right)$$

e. Construct graphs of the variation of normal acceleration for onset, tracking, and limit buffet with Mach number for each test altitude.

f. Compute and construct graphs of:

Variation of instantaneous rate of turn with Mach number/KCAS.

Variation of instantaneous radius of turn with Mach number/KCAS.

SECTION IX

CLIMB PERFORMANCE

## REFERENCES

### SECTION IX

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2. Dommasch, Airplane Aerodynamics, Section 9:7
3. Petersen, Aircraft and Airplane Performance, Chapter 8, p. 156
4. Rutowski, DAC, "Energy Approach to the General Aircraft Performance Problem"
5. USAF, ARPS PTM, Sections 4.4 and 4.5 of Theory

## CLIMB PERFORMANCE

### PURPOSE

The purpose of this test is: (1) to determine the climb performance of a turbojet airplane; (2) to determine compliance with the combat ceiling (500 fpm rate of climb) or climb performance guarantee of the applicable Detail Specification; and (3) to check proposed climb schedules and obtain data for presentation in Flight Handbooks.

### DISCUSSION AND THEORY

The excess power (specific energy) characteristics of an airplane can be determined at several altitudes using the level acceleration run tests described in Section VI. Maximum rate of climb and maximum energy climb schedules can be derived from these data using the procedures set forth in Section XII. Since computation of time to climb data from acceleration runs (or sawtooth climbs) is tedious and does not always yield accurate results, time to climb data are normally determined by flying continuous climbs to service ceiling.

Normally, the contract guarantee ceiling would be checked using the contractor's recommended climb schedule. For the purposes of this evaluation, TPS derived schedules will be flown. The maximum energy climb schedule results in minimum time to a given energy altitude. Variations of this schedule may result in minimum time to a pressure altitude, maximum range or greater cruise ceiling. Close approximation may be obtained by using a constant KCAS/Mach schedule which is easier for a pilot to fly.

An important aspect of climb performance is the time and fuel required to accelerate from brake release to the climb schedule airspeed near sea level. Test climbs are usually divided into two phases: (1) the acceleration and transition to climb schedule and (2) the climb from near sea level to the ceiling.

In performing the climb test, it is important to fly the desired climb schedule precisely and smoothly. If the pilot strays from the schedule momentarily but corrects his airspeed smoothly, the average rate of climb for the interval will closely approximate the rate of climb which would have been achieved had the schedule been flown precisely. This is based on the principle of interchangeability of potential and kinetic energy and was applied in the acceleration runs test also. There are energy losses in this exchange, of course, and the effect of deviations from the desired climb schedule becomes more critical as altitude increases.

Another factor which must be considered in the climb test is the effect of wind velocity gradient with altitude. If the climb were flown at constant true airspeed into a headwind which increased with altitude, the velocity of the airplane (relative to the ground) would decrease as altitude increased. Some of the airplane's kinetic energy would be converted to potential energy (in effect, a zoom maneuver), and the observed rate of climb would be greater than that observed on a similar climb in still air conditions. Correction for horizontal wind gradients with altitude and for vertical movements of the air mass are tedious and beyond the scope of this exercise. Gradient effects can be minimized by performing the test climbs on a heading perpendicular to the average winds.

The correction of jet climb performance data to standard day conditions is similar to the corrections required for acceleration run data - temperature correction for thrust, momentum correction for weight, and induced drag correction. The thrust correction usually is the most significant although the weight and induced drag corrections must always be considered. An assumption in the correction of climb data is that the standard day calibrated airspeed at a given altitude is the same as the test day airspeed. When evaluating a climb performance guarantee which prescribes a standard rate of change of airspeed, an additional correction to rate of climb for nonstandard  $dV/dh$  or  $dV/dt$  must be made.

## TEST PROCEDURES AND TECHNIQUES

### Preflight Procedures

- a. Required data (Photopanel required - set on 1/2 frame/sec).

Time.

Pressure altitude.

Airspeed.

OAT.

Fuel used (or remaining).

RPM (for reference).

Take photopanel data every 4,000 ft interval for altitudes below 30,000 ft and every 2,000 ft for higher altitudes but no less frequently than every 2 min. For high performance airplanes, it is usually desirable to take continuous photopanel data to a medium altitude. Take correlating data on kneeboard.



- b. Prepare data card (see Figure 1).
- c. Obtain assigned climb schedule from group leader.
- d. Obtain upper air wind velocities from aerology. Estimate average direction weighting strong vertical gradients more heavily.
- e. Review takeoff technique.
- f. Carry a 60-sec stopwatch or navigator's hack watch.

WEATHER		CARD NO.	
CLIMB TO CRUISE CEILING		OUK-1	
AIRPLANE TYPE	BU. NO.	TIME T.O.	DATE
PBY-4	001394	TIME LAND	
PILOT		T.O.C.G.	
KIDD		GEAR DOWN	24.5
CONDITION		T.O. GROSS WEIGHT	
TRAINER		51,346 lb	
EXTERNAL CONFIGURATION			
2 OAT PROBES			
TO: TIME		SEC	DISTANCE
Mpa	Vo	Alt	P/P
FIELD ELEV		0	
ALT FOR			
INTERRUPTION			
3000		0	
7000			
11,000			
PARTICULAR CLIMB SCHEDULE			
CRUISE CLIMB			

Figure 1  
Sample Pilot's Data Card

### Flight Procedures

The climb performance test will be conducted in two parts - brake release to 3,000 ft (an arbitrary altitude) and climb to cruise ceiling from near sea level. Estimated takeoff performance will also be determined during the first part of the test.

a. Energize the photopanel and release brakes at a predetermined film counter number. The takeoff should commence at a runway marker so that a reasonable estimate of test day takeoff performance can be made. Record time and distance to lift-off. Start watch at brake release. Record time and fuel required for takeoff, acceleration, and climb to 3,000 ft above field elevation. The acceleration phase should be conducted at as low an altitude as possible commensurate with safety, course rules, and sound judgment.

b. Establish the airplane on a heading perpendicular to the average wind direction. Energize photopanel. Set MIL thrust and stabilize on the required climb schedule to pass through 2,500 ft pressure altitude in a stabilized, trimmed climb. When the takeoff heading is within 30 deg of the climb heading, the climb may be done in one continuous maneuver from brake release. Do not exceed 10 deg bank angle while adjusting heading in the climb.

c. Commence timing the climb at 3,000 ft pressure altitude. Record kneeboard and photopanel data at 4,000 ft intervals. Let the photopanel run continuously until well established in the climb. Thereafter, energize the photopanel for 10 to 20 sec while passing through each data altitude.

d. Above 30,000 ft record data at 2,000 ft intervals, but no less frequently than each 2 min until cruise ceiling is attained. Determine that the rate of climb is less than 300 fpm by observing a time history of altitude. It is important that this part of the climb schedule be flown precisely.

e. If it is necessary to reduce thrust to maintain EGT limit, note this on the data card.

f. Should it become necessary to interrupt the climb for any reason, the gross weight change must be kept to a minimum. At low altitude, this is best accomplished by a rapid descending turn at idle thrust supplemented by the use of speedbrakes, if necessary. At high altitudes, it may be necessary to make the turn at MIL or MAX thrust to prevent excessive altitude/airspeed loss and suffering the attendant increase in fuel used attempting to regain lost altitude and/or airspeed.

g. The airspeed indicator, vice the Machmeter, should be used for the climb schedule since the Machmeter is of insufficient accuracy.

#### MANUAL DATA REDUCTION

##### General

The climb data may be presented for test day conditions. For a more exact analysis, the data should be corrected for the effects of nonstandard thrust and gross weight. Record the following data from the photopanel film -- at brake release, at climb airspeed and at 3,000 ft above field elevation; in the climb at 4,000 ft intervals from 3,000 ft pressure altitude to 30,000 ft and at 2,000 ft intervals above 30,000 ft, but at least every 2 min.

Elapsed Time	H <sub>p</sub>	V	W <sub>F</sub>	OAT	N
--------------	----------------	---	----------------	-----	---

#### Data Reduction - Test Day Climb Performance to Cruise Ceiling

- a. Correct observed altitude and airspeed for instrument and position errors.
- b. Determine Mach number from  $V_c - M - H_p$  chart.
- c. Correct observed OAT for instrument error and determine the ambient temperature from Appendix IV.
- d. Calculate true airspeed:

$$V_T = 39.0 \sqrt{T_A} \quad (^\circ K) \quad \text{kt}$$

- e. Compute fuel used in lb.
- f. On a sheet of 16" x 20" graph paper, construct a plot of the variation of pressure altitude with the time for both parts of the climb test. Use scales of about 5,000 ft/in. and 5 min/in. Assume that the data recorded at field elevation and 3,000 ft above field elevation are equivalent to data which might have been recorded on another day at sea level and 3,000 ft pressure altitude. On the same graph sheet also plot the variation with altitude of calibrated airspeed or Mach number (depending on climb schedule used), both test day and standard day ambient temperature and fuel used. Fair a smooth curve through the time to climb and airspeed data. In fairing these curves, observe that when the airspeed was faster than the mean, the climb data points will fall below the faired climb curve and vice versa. There may be a slight discontinuity in the time to climb curve at an altitude when it was necessary to reduce thrust, or at an altitude where the temperature lapse rate changed abruptly. Fair a curve through the fuel used data. Extrapolate both the time to climb and fuel used curves to sea level.

g. From the faired time to climb curve, graphically measure the indicated rate of climb,  $(dH_p/dt)_i$  at 5,000 ft intervals beginning at sea level. Plot the variation of indicated rate of climb with altitude. Mark the combat ceiling (500 fpm) and cruise ceiling (300 fpm).

h. From the faired curves of time and fuel used, record the time to climb and fuel required at 5,000 ft intervals for a climb from sea level (at climb airspeed) to cruise ceiling.

i. Compute the time to climb through each 5,000 ft interval,  $\Delta t$ .

j. For each interval, compute the no-wind distance flown.

$$d = V_{T_{av}} \frac{\Delta t}{60} \quad \text{nmi}$$

k. Compute the distance flown from start of climb to each altitude.

$$\Delta d_H = \sum_{H=0}^H d \quad \text{nmi}$$

l. Determine the time and fuel required for takeoff and acceleration to climb airspeed by subtracting time and fuel required for climb from sea level to 3,000 ft (extrapolation of climb data) from the observed time and fuel required from brake release to 3,000 ft.

m. These results may be presented as the table of time, fuel used, and air distance (from sea level at climb airspeed) to cruise ceiling in 5,000 ft intervals of altitude. State the time and fuel required for takeoff and acceleration in a footnote to the table.

#### Data Reduction - Ceiling Guarantee

The rate of climb observed at some arbitrarily chosen altitude can be corrected to standard conditions of thrust and gross weight. If such corrections are made to rate of climb observed at several altitudes near the guarantee ceiling, a curve defining the variation of standard day rate of climb with altitude may be drawn, and the guarantee ceiling at standard conditions of thrust and gross weight established. The ceiling also is affected by airspeed. This test was flown at several climb schedules, one of which will result in the greatest ceiling for the conditions of the guarantee.

n. For the several data points obtained at 34,000 ft and above correct observed pressure altitude and airspeed for instrument and position errors.

o. Determine Mach number from  $V_c - M - H_p$  chart.

p. Correct observed OAT for instrument error and determine ambient temperature from Appendix IV (note - some of the ambient temperature data computed in step c may be used if suitable)

q. Calculate true airspeed.

$$V_T = 65.8 \sqrt{T_a} \text{ (°F)} \quad \text{fps}$$

r. Determine airplane gross weight from fuel used or fuel remaining.

s. Obtain pressure ratio,  $\delta$ , and standard day ambient temperature  $T_{as}$ , from Appendix VII.

t. Plot the variation of time to climb and true airspeed with altitude. Fair a curve through these data. Graphically determine indicated rate of climb,  $(dh_p/dt)_t$ , at 30,000 ft for at least three higher data points.

u. Calculate actual rate of climb for test day.

$$(R/C)_t = \left( \frac{dh_p}{dt} \right)_t \left( \frac{T_a}{T_{as}} \right) \quad \text{fpm}$$

v. Compute net thrust correction (nonstandard temperature).

$$\Delta(R/C)_1 = \left( \frac{60V_T}{W_S} \right) \left( \frac{dF_n/dT}{dT} \right) (T_{as} - T_a) \quad \text{fpm}$$

where

$$\begin{aligned} \frac{dF_n/dT}{dT} &= -70 \text{ lb/}^\circ\text{K} \quad (\text{J57-P-4/8/16 engine}) \\ &= -40 \text{ lb/}^\circ\text{K} \quad (\text{J-52-P8A engine}) \\ &= -46 \text{ lb/}^\circ\text{K} \quad (\text{J-85 engine}) \end{aligned}$$

w. Compute induced drag correction.

$$\Delta(R/C)_2 = \left( \frac{120gR}{P_{s1} \text{ eb}^2 W_S} \right) \frac{T_{as}}{V_T} (W^2 - W_S^2) \quad \text{fpm}$$

where

$$\begin{aligned} \frac{120gR}{P_{s1} \text{ eb}^2 W_S} &= 5.4 \times 10^{-6} \text{ for A-4} \\ &= 10.8 \times 10^{-6} \text{ for T-38} \\ &= 1.05 \times 10^{-6} \text{ for F-8K} \end{aligned}$$



x. Compute standard day rate of climb for the conditions of the guarantee.

$$(R/C)_S = \frac{W}{W_S} (R/C)_t \left[ \frac{T_{as}}{T_a} \right]^{1/2} + \frac{1}{2} (R/C)_1 + \frac{1}{2} (R/C)_2$$

y. Plot the variation of standard rate of climb with altitude. Extrapolate as necessary to determine service ceiling. On the same graph, show the test day rate of climb variation and the test conditions (average gross weight and average difference of ambient temperature from standard).

#### COMPUTER DATA REDUCTION

Climb data is normally reduced by CSD (Computer Services Division) with resultant plots reduced to standard day conditions. Determination of climb schedules from acceleration run data is contained in Section XII of this manual.

SECTION X  
DESCENT PERFORMANCE

## REFERENCES

### Section X

1. Petersen, Aircraft and Engine Performance, Chapter 10.
2. USAF, ARPS PTM, Section 4.7 of Theory.

## DESCENT PERFORMANCE

### PURPOSE

The purpose of this test is to determine the idle thrust descent performance - time, fuel used, and range - in a descent from 40,000 ft to near sea level at landing gross weight. Such descent tests are required to determine the most desirable airspeed for descent and to obtain data for presentation in Flight Handbooks.

### DISCUSSION AND THEORY

For stabilized, level flight, engine thrust is adjusted to balance the airplane drag. In terms of power, we say that engine power equals power required ( $FV = DV$ , ft-lb/sec). If the thrust were reduced to zero, then the power required to maintain the airplane's airspeed must come from the airplane's time rate of change of kinetic and potential energy. The rate of expenditure of this energy varies directly with the rate of descent and linear acceleration. The minimum rate of descent will occur at the airspeed for minimum power required. The maximum gliding range will be obtained at a faster airspeed where the maximum ratio of true airspeed to power required occurs.

The airspeed for minimum rate of descent at idle thrust in the jet airplane is approximately the airspeed which results in the maximum ratio of  $C_L^{3/2}/C_D$ . The airspeed for maximum range in the idle thrust glide is approximately the airspeed for maximum ratio of  $C_L/C_D$ . The idle thrust descent airspeed schedule can be estimated on the basis of these theoretical ratios; however, for a more precise result, the effects of residual thrust also must be considered. The estimated idle descent airspeed schedule can be refined by performing a series of descents at several constant airspeeds. Usually, a constant calibrated airspeed descent

schedule which will give near optimum performance can be determined. For this test, idle thrust descents will be made at one gross weight, and the effects of weight changes will not be considered. Qualitatively, an increase in airplane gross weight has the effect of increasing the airspeed required to maintain the desired lift coefficient which is stated above. The rate of descent observed at the heavier gross weight will be greater because the power required is greater; however, the glide angle will not change, and the range in the descent will be unchanged by an increase in weight. This statement must be qualified in that it does not consider the effect of residual thrust at the idle thrust setting or steady winds. Generally, the residual thrust effects result in increased descent range as airplane weight decreases. Because a change in airplane kinetic and potential energy is involved in the idle thrust descent test, wind gradients will affect the observed rate of descent. The descent must be flown on a heading perpendicular to the average wind direction to minimize vertical wind gradient effects.

## TEST PROCEDURES AND TECHNIQUES

### Preflight Procedures

- a. Required Data (photopanel desired - 1/2 frame/sec)
  - Time
  - Pressure altitude
  - Airspeed
  - OAT
  - Fuel used
  - RPM (for reference only)

d. Carry a 60-sec stopwatch or navigator's back watch.

Figure 1  
Sample Pilot's Data Card

### Flight Procedures

Normally, the idle thrust descent test will be performed in the same flight as the service ceiling climb test. It is desirable to perform the descent along a reciprocal flight path of that used for the climb because it will probably allow the ambient temperature information obtained in one of the tests to be used for both.

a. Proceed to the required starting point at about 42,000 ft. When fuel remaining is 2,000 lb (1,600 lb in T-38A) establish a heading inbound to the field and perpendicular to the average wind direction. Stabilize in level flight at the required airspeed. Set idle thrust and quickly retrim so as to be stabilized in the glide at 40,000 ft.

b. Energize the photopanel briefly and record required data at 40,000 ft and at 5,000 ft intervals in the descent. Record last data at 3,000 ft pressure altitude.

c. At 3,000 ft after mentally noting required data and advancing the throttle, record the data on the kneeboard. Be alert to the possibility of fuel control icing and be prepared to take proper corrective action immediately if the engine fails to accelerate.

### MANUAL DATA REDUCTION

#### General

The descent data will be presented for test day conditions. For a more exact analysis, the rate of descent should be corrected to a standard gross weight. Corrections to thrust are not significant in this test. Record the following data from the photopanel film at 5,000 ft intervals beginning at 40,000 ft and also for 3,000 ft:

Elapsed time       $H_p$        $V$        $W_F$       OAT       $w_{f*}$        $N^*$

\*for reference only

#### Data Reduction - Idle Thrust Descent

- a. Correct observed altitude and airspeed for instrument and position errors.
- b. Determine Mach number from  $V_c - M - H_p$  chart.
- c. If the idle descent was performed over the same flight path as the service ceiling climb and the weather conditions were stable, use ambient temperature calculated in step c of the climb data reduction. Otherwise, correct observed OAT for instrument error and determine the ambient temperature from Appendix IV.

- d. Calculate true airspeed

$$V_T = 39.0 M \sqrt{T_a} \text{ (}^\circ\text{K)} \quad \text{kt}$$

- e. Compute fuel used or fuel remaining in pounds.
- f. On a sheet of 16" x 20" graph paper, construct a graph of the variation of pressure altitude with time in the descent. Use scales of about 5,000 ft/in and 5 min/in. Fair a smooth curve through the data and extrapolate to sea level. On the time to descent curve, note the CAS schedule flown.
- g. From the faired time to descent curve, graphically measure indicated rate of descent,  $(dH_p/dt)_t$  at 5,000 ft intervals beginning at 40,000 ft. Also plot this data on the large graph.
- h. From the faired curves of time and fuel used, record the time to descend and the fuel used in a descent from 40,000 ft to sea level. Repeat at 5,000 ft intervals.



i. Compute time to descend through each 5,000 ft interval,  $\Delta t$ .

j. For each interval, compute the air distance flown.

$$\Delta d = V_{T_{av}} \frac{\Delta t}{60} \quad \text{nmi}$$

k. Compute the glide distance from each 5,000 ft interval to sea level (note that in steps h and k that the reference point is the terminus of the glide - sea level).

$$d_H = \sum_{S-L} d \quad \text{nmi}$$

l. The results may be presented as a table of time, fuel used, and air distance for an idle thrust descent to sea level from 5,000 ft intervals of altitude (to 40,000 ft).

m. Note - had fuel flow information been desired, more reliable data are obtained from differentiation of the variation of fuel used with time than from the fuel flow indicator, which generally is inaccurate at low flow rates.

$$\dot{w}_{av} = \frac{W_{F_b} - W_{F_a}}{\Delta t} \quad \text{lb}$$

#### COMPUTER DATA REDUCTION

See Section XIII.

SECTION XI

STOL PERFORMANCE

NOTATIONS INTRODUCED IN THIS SECTION

$R/C$	Rate of Climb
$\gamma$	Flight path angle
$V_{\gamma \max}$	Airspeed for greatest flight path climb angle
$V_{R/C \max}$	Airspeed for greatest rate of climb
$Q$	Engine torque pressure
$N_1$	Engine speed - percent

## INVESTIGATION OF STOL PERFORMANCE

### PURPOSE

The purpose of this test is to determine the STOL performance of a fixed incidence turboprop airplane. Airspeed for stall, best climb angle, best rate of climb, and best approach angle will be determined. Test day takeoff and landing distances will be determined.

### DISCUSSION AND THEORY

The performance of a STOL airplane will depend greatly upon its engine thrust line angle, its variable geometry or incidence features, and its gross weight. In the case of the fixed incidence STOL airplanes, the principle source of lift will be the wing, therefore, the excess power characteristics will dictate its potential takeoff and landing performance. Additional limitations may be imposed by flying quality deficiencies.

The sawtooth climb method can be used to determine excess power characteristics and airspeeds for best angle and rate of climb. The best approach configuration and airspeed can be determined by a sawtooth descent method. The effect of gross weight on these airspeeds cannot always be calculated by the classical square root weight ratio method since the airplane pitch angle may be so large that the thrust component supplies a significant amount of lift. This means that test data points must be taken at both heavy and light gross weights. Relatively small variations in gross weight can be corrected by the classical method in order to correct test data to one weight. After best airspeeds have been determined, actual takeoff and landing distances must be measured. A number of special test recording devices are available for takeoff and landing data collection

and analysis. The Fairchild flight analyzer is a typical device used for determining takeoff and landing performance. The touchdown rate of descent indicator (TROD) is a typical carrier suitability test device which can be used to rapidly determine sink rate.

Special pilot techniques may be required for STOL operation and performance testing. An airplane with a fixed incidence wing and a fixed thrust axis will require particular pilot attention to ensure that the angle of attack for maximum lift is established at the earliest possible point in the takeoff roll. If sufficient longitudinal control is available, simply holding full aft stick will achieve the desired angle of attack before minimum lift-off speed is attained. If either the elevator control power is marginal or the location of the main landing gear precludes the airplane's achieving the necessary angle of attack prior to attaining minimum lift-off speed, it may be possible to "lounge" the airplane to the desired attitude. This is done by compressing the nose strut with a full forward stick input followed by a full aft stick input. This technique can result in attaining the desired airplane attitude at slower airspeeds than from static control inputs. The airplane will usually become airborne with full aft stick. After becoming airborne, it may be necessary to ease the stick forward slowly to accelerate to the airspeed for best climb angle, or it may be possible to simply hold full aft stick while the airplane climbs to some desired altitude without accelerating to a higher airspeed.

Unfortunately, the takeoff airspeed and possibly the airspeed for best climb angle will usually be less than the minimum single engine control airspeed. The pilot must always be prepared to abort a takeoff at the first sign of significant loss of power from an engine. This possibility must be considered when recommending a landing gear up or down climb configuration.

The importance of best climb angle for clearing an obstacle after takeoff is obvious. The airspeed for best climb angle will be lower than the airspeed for best rate of climb. While there will be a best climb airspeed for every airplane, the method of accelerating to that speed, or the requirement for actually accelerating to that airspeed will depend upon the airplane's performance and mission.

The short field landing also requires some special pilot techniques. Minimum landing distance will depend upon a combination of reverse pitch and brakes. Reverse pitch is usually the primary means of controlling landing roll distance. As a general rule, if full power is available in reverse pitch, little or no braking will be required. When less than 50% power is available in reverse pitch, brakes will be required, and special pilot precautions must be taken to avoid blown tires. Whenever any amount of reverse pitch is used, it must be applied while the engine power is still relatively high. This means the approach should be made with the highest possible power requirement. This may result in airspeeds that are on the unstable side of the power required curve. Again, this airspeed will usually be lower than the minimum single engine airspeed. The approach may be a constant rate of descent (mirror type) or a level approach with a fixed throttle high dip and flare to touchdown (flap paddles type). The type of approach will be dictated by the requirement to maintain sufficiently high power for effective reverse pitch operation and a sufficiently safe airspeed-power-altitude relationship to avoid single engine catastrophe. As soon as the airplane touches down, full available reverse power must be applied. If nose wheel steering is available, the nose wheel must be held on deck during reverse pitch operation. In some short field work (C-130 carrier trials for instance), reverse pitch may be applied prior to actual touchdown. This technique usually requires an outside observer or LSO, and will not be used at USNTPS.

## TEST PROCEDURES AND TECHNIQUES

### Pertinent Particulars

- a. At the time these tests are performed, the student test pilot should have completed both the performance and the longitudinal, lateral-directional syllabus test flights.
- b. Determine stall characteristics before conducting low speed tests near the ground.
- c. Ensure that autofeather feature operates properly prior to conducting actual takeoff and landing tests.

### Preflight Procedures

- a. First flight required data.
  1. Stall airspeed at heavy and light weights.
  2. Sawtooth climb information,  $V_o$ ,  $R/C_o$ ,  $H_{p_o}$ ,  $W_R$ , Configuration, and Remarks.
  3. Sawtooth descent information,  $V_o$ ,  $R/C_o$ ,  $H_p$ ,  $W_R$ , Configuration, Torque, RPM,  $N_1$ .
- b. Second flight required data.
  1. Takeoff data. Configuration,  $V_{o_{NLO}}$ ,  $V_{o_{TO}}$ ,  $t_{TO}$ ,  $t_{NLO}$ ,  $H_p$ ,  $S_{TO}$ .
  2. Landing data.  $V_{o_{PA}}$ ,  $V_{o_L}$ ,  $V_{o_R}$ ,  $t$ ,  $S_L$ , Configuration  $Q_{PA}$ ,  $RPM_{PA}$ .
  3. Prepare pilot's data cards (see Figures 3, 4, and 5).

## Flight Procedures

### First Flight:

Determine stall airspeeds, airspeeds for best climb angle and rate of climb, and best approach airspeed and configuration for approach. The airspeed range for this investigation should be from  $V_{\text{stall}}$  to a midcruise airspeed. Configuration takeoff should be investigated with LG down and retracted. Power approach configuration should be investigated with speedbrakes in and out, full flaps, gear down. At least two constant power settings should be investigated. The most desirable approach configuration will be one that requires sufficient power to ensure that engine RPM is high, but does not require a power setting that would preclude successful recovery from a sudden engine failure. Data must be obtained at a relatively heavy and a relatively light weight.

Sawtooth climbs and descents will use indicated rate of climb or descent in lieu of a timed climb. Indicated rate of climb information will be sufficiently accurate for STOL evaluation purposes. If refinement of results is required, the time method could be employed.

The recommended sequence of events on the first flight is as follows: After a normal takeoff, climb to 7,000 ft  $H_p$  and determine stall airspeeds in each configuration to be tested. Then descend to 3,000  $H_p$  and conduct sawtooth climbs and descents. Start at the high airspeed and collect data by alternating climbs and descents. Take test data from base altitude of 3,000 ft  $H_p$  and ensure that data taken near stall airspeed is taken at an altitude near 7,000 ft  $H_p$ . Data will be recorded by the acting copilot when the pilot has established equilibrium conditions for at least 30 sec. Determine stall airspeeds at 7,000 ft and with approximately 400 lb of fuel remaining. Make a normal landing.



### Second Flight:

The second flight will be used to determine takeoff and landing distances and proper pilot techniques for minimum STOL distances. Takeoffs should be investigated by using neutral stick and then rotating normally at lift-off airspeed by using full aft stick and by using a dynamic lift-off technique. The dynamic technique is full back stick at  $V_{STO-10}$  kt. At this airspeed, apply full forward stick followed by full aft stick. The effect of cocking the airplane to overcome initial torque should also be investigated. Nose wheel steering should be used for all takeoffs. STOL's are not to be conducted with autofeather inoperative. Takeoff time, distance, and lift-off speed will be recorded.

Landing approach and landing will be evaluated using both flat and descending approaches. Reverse pitch and nose wheel steering will be used. Brakes should be used only after the landing roll is essentially complete. Landing distance, approach airspeed, touchdown airspeed, and configuration will be recorded.

### DATA REDUCTION AND PRESENTATION

#### First Flight

a. Correct stall data to indicated stall airspeed at one heavy and one light standard weight.

$$V_O + V_{IC} = V_i$$

$$V_{IS} = V_{IT} \left( \frac{W_T}{W_S} \right)^{1/2}$$

b. Correct sawtooth climb data to indicated airspeed and observed rate of climb at heavy standard weight for heavy weight data points, and at light standard weight for light weight data points.

$$V_o + V_{ic} = V_i$$

$$V_{is} = V_{it} \left( \frac{W_s}{W_t} \right)^{1/2}$$

$$R/C_{os} = R/C_{ot} \left( \frac{W_s}{W_t} \right)^{1/2}$$

c. Plot variation of observed rate of climb with indicated airspeed, in configuration TO at both standard weights (see Figure 1).

d. Determine airspeed for best rate of climb.

e. Determine airspeed for greatest climb angle by plotting a tangent as shown in Figure 1. The airspeed at the tangent point represents best climb angle airspeed since

$$\left( \frac{R/C}{V} \right)_{\max} = \sin_{\max}$$

f. Determine whether there is any advantage in retracting the main landing gear to achieve best climb angle after takeoff from an analysis of sawtooth climb data. Consider operational aspects. Figure 2 is an example of results which would indicate gear should be kept down.

g. Correct sawtooth descent data to indicated airspeed and observed rate of descent at 11,000 lb standard gross weight and plot results as shown in Figure 1.

h. Determine airspeeds for minimum rate of descent approach and descent angle at that airspeed for both power settings investigated. See example in Figure 1.

i. Plan second flight from test results determined in steps a through h. Include evaluation of at least two airspeeds and configurations for landing evaluations.

Second Flight

j. Discuss the most successful STOL pilot techniques which you determine from these tests.

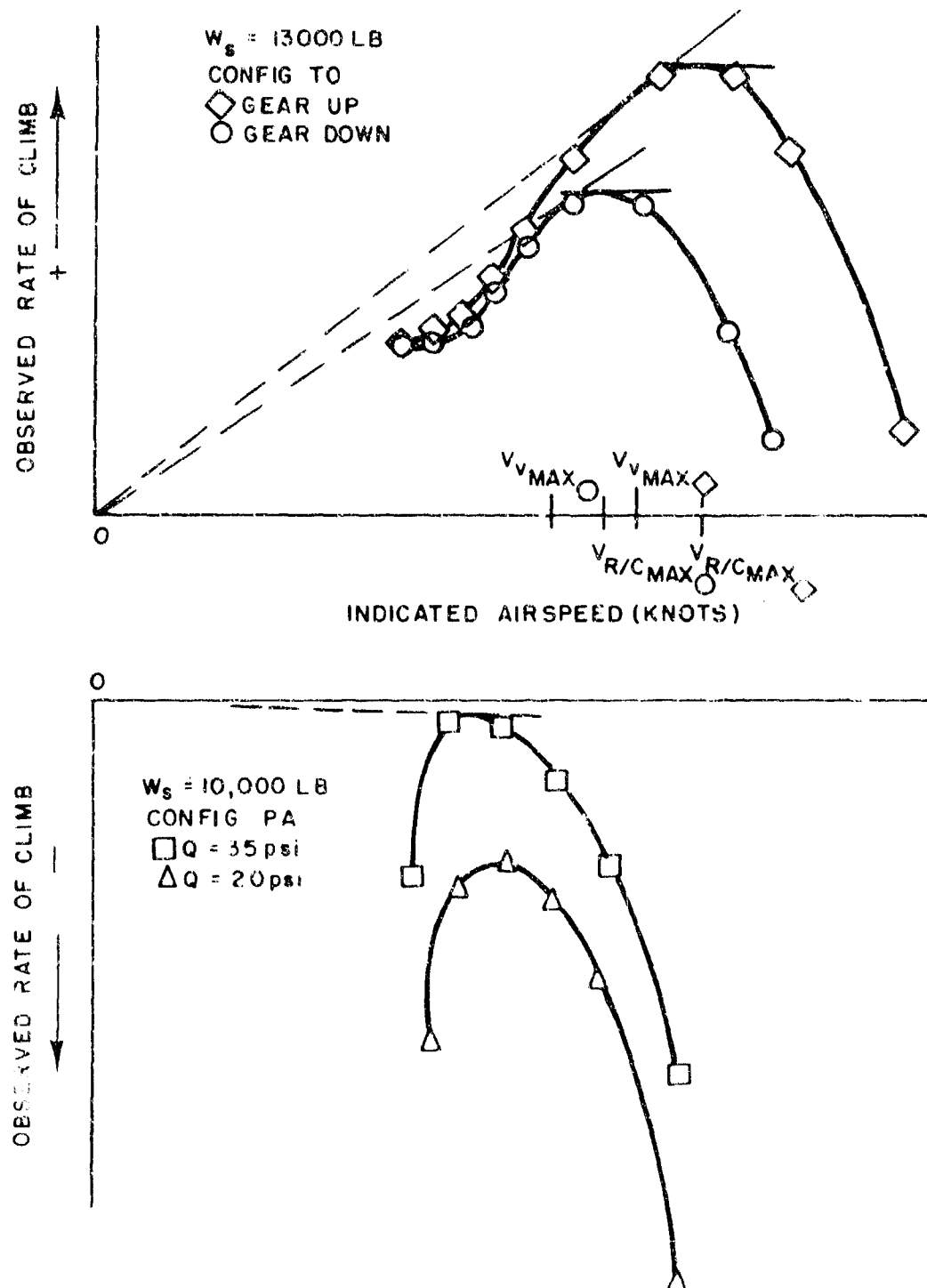


Figure 1  
Sample Data Presentation

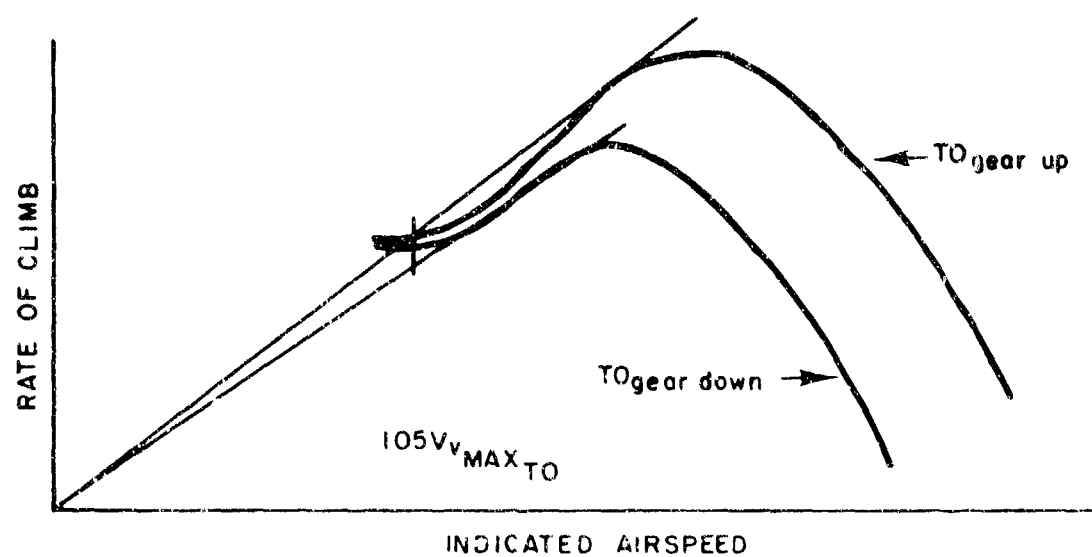


Figure 2  
Sample Data Presentation

WEATHER				CARD NO.			
				PIW BIS			
AIRPLANE TYPE	BU. NO.	TIME T.O.	DATE				
OV-1A	137815	TIME LAND	9 Sept 1978				
PILOT		T.O. C.G.					
		GEAR DOWN	UP				
CONDITION				T.O. GROSS WEIGHT			
NORMAL OBSERVATION							
EXTERNAL CONFIGURATION							
WING TANKS				S. E Speed TO PA			
STALL EVALUATION							
CONFIG	V <sub>0</sub>	H <sub>0</sub>	WR	Q <sub>0</sub>	RPM <sub>0</sub>	Q <sub>0</sub>	Remarks
CR					1450		
TO					1690		
PA					1690		
TO,					1690		
L					1690		
SAWTOOTH DATA - CONFIG TO PA							
V <sub>0</sub>	110	120	110	100	115	100	115
R <sub>0</sub>							
H <sub>0</sub>							
WR							
SAWTOOTH DATA - CONFIG PA							
V <sub>0</sub>	110	105	115	100	115	100	115
R <sub>0</sub>							
H <sub>0</sub>							
WR							
						RPM - 1690	
						Q <sub>0</sub>	
						N <sub>0</sub>	

Figure 3  
Pilot's Kneeboard Card  
First Flight







SECTION XII  
DETERMINATION OF CLIMB SCHEDULES  
FROM ACCELERATION RUN DATA

## DATA REDUCTION PROCEDURE FOR DETERMINING CLIMB SCHEDULES

### FROM ACCELERATION RUN DATA

#### GENERAL

The variation of rate of change of specific excess power or specific energy ( $P_s$  or  $dE_h/dt$ ) at MIL thrust at several altitudes was determined from acceleration runs. To determine a climb schedule from these results, the acceleration run data must be plotted in a single graph, faired as a family, and cross-plotted as lines of constant rate of change of  $P_s$  on coordinates of altitude and airspeed.

In reducing the acceleration run data, corrections were made for nonstandard airplane weight. The corrections for gross thrust and ram drag variations because of nonstandard temperature were assumed to be negligible and were not made for manual data reduction. If automatic data reduction was employed, these corrections were included. In the preliminary fairing of manually reduced data, it may be necessary to consider the effect of nonstandard temperature.

#### DATA REDUCTION

In the acceleration run test, the results were presented in the form of variation of  $P_s = dE_h/dt$  with Mach number or CAS for several altitudes. Further data reduction will commence with those results.

Using a 16" x 20" graph sheet, plot the variation of  $P_s$  with  $M$  for each altitude. Lightly sketch a curve through the data and label with the altitude and  $T_a$  (if reduced manually). When the curves for all test altitudes are plotted on this graph, they should form a family similar to Figure 1. It may be necessary to adjust some of the curves to fit this family. Two effects may be noted:

a. Warmer than standard ambient air temperature will tend to shift the curves downward in relation with those obtained at standard conditions. This feature is more apparent at high altitudes.

b. The curves will tend to break downward sharply at the same Mach number.

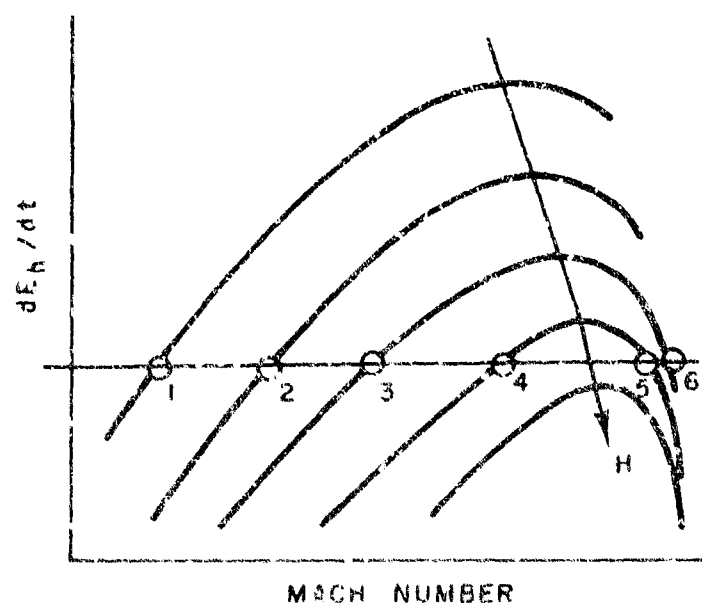


Figure 1  
Specific Excess Power Working Plot

When the fairing of the working plot is complete, cross-plot the data as lines of constant  $P_s$  on coordinates of altitude and true airspeed - use the prepared Specific Energy Analysis graph.

a. At several values of  $P_s$ , record the  $M$  corresponding to each altitude, (points 1 through 6 in Figure 1).

b. Compute  $V_T$  for each value of  $M$  obtained.

$$V_T = 38.9 \sqrt{\rho_{sl} / \rho_{sl_0}}$$

at

c. Plot these data as lines of constant  $P_s$  on the Specific Energy Analysis graph. This plot should form a family similar to Figure 2. If unusual bends or anomalies occur in these curves, refer to the working plot. The fairing of that data may be at fault.

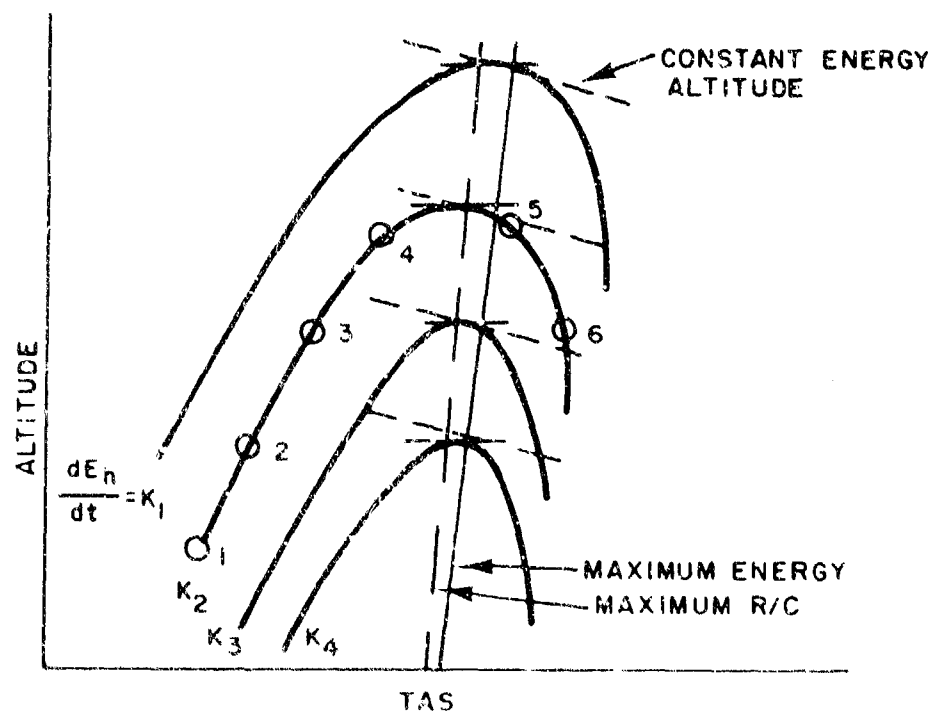


Figure 2  
Specific Energy Analysis Plot

A climb schedule may be determined now. A line connecting the peak values of  $P_s$  is approximately a maximum rate of climb schedule. A line connecting the points at which the constant  $P_s$  lines are tangent with the energy height lines (i.e., points of peak energy height) is the maximum energy climb schedule. For the purpose of interpreting the climb schedule, it is convenient to scribe lines of constant Mach number on the Specific Energy Analysis plot.

Additional discussion of the maximum energy climb schedule is found on pages 156 through 165 of the Aircraft and Engine Performance manual.

SECTION XIII  
COMPUTER DATA REDUCTION

## COMPUTER DATA REDUCTION

### ACCELERATION RUNS, CLIMB AND DESCENT PERFORMANCE

#### GENERAL

Analysis of level acceleration, climb, and descent data by digital computer routines is common practice at both military and contractor test facilities. The automatic data reduction process provides a rapid and accurate method of obtaining corrected performance results from raw recorded data.

#### DATA REDUCTION ROUTINE

The data reduction routines used by the NATC Computer Services Division (CSD) are mathematically similar to the methods described in applicable sections. The net thrust and induced drag corrections are added for level accelerations. The method requires inputs of observed airspeed, altitude, elapsed time, OAT, and fuel remaining or fuel use. Instrument corrections and positive error corrections can be automatically applied from a correction subroutine. The engine net thrust correction is provided by a subroutine that is based on engine manufacturer's data.

#### DATA INPUT

Position error correction for each type airplane must be sent to CSD. Instrument corrections for each instrument used during the test must be submitted for each airplane for altitudes up to 50,000 ft. This information need only be submitted once for each airplane (if no instruments are changed between flights). See Figure 1 for sample format.

AIRPLANE TYPE _____		BuNo _____	
(TYPE INSTRUMENT)/ (DATE OF CAL)		(TYPE INSTRUMENT)/ (DATE OF CAL)	
OBSERVED INST READING	INSTRUMENT CORRECTION	OBSERVED INST READING	INSTRUMENT CORRECTION

Figure 1  
Sample Format for Instrument Calibration Submission

A performance data sheet for each acceleration run, climb, or descent must be submitted in the format shown in Figure 2. Data should be recorded every 4 sec (every other frame) for acceleration runs. For climbs it should be recorded every 4,000 ft (2,000 ft above 30,000 ft) or every 2 min, whichever comes first. For descents, record data every 5,000 ft. The elapsed time ( $\Delta t$ ) is from start time ( $t_0$ ) and should be measured in minutes and seconds.

Prior to submitting data to CSD for processing, a rough plot of  $V_o$  vs. time for acceleration runs or average rate of climb (or descent) vs. altitude for climbs (or descents) should be made to ensure you are submitting your data correctly. Any gross errors (such as failing to account for climb interruption) will be immediately obvious from these plots and unnecessary time delays will be avoided.

AIRPLANE TYPE \_\_\_\_\_ BuNo \_\_\_\_\_  
 PILOT \_\_\_\_\_ DATE OF FLIGHT \_\_\_\_\_

Standard Gross Weight \_\_\_\_\_  
 Start Gross Weight (fuel used counter) \_\_\_\_\_  
 Empty Gross Weight (fuel remaining system) \_\_\_\_\_  
 Operational Weight for Climb Intercept \_\_\_\_\_  
 Counts Per Gallon (6.8 lb/gal) \_\_\_\_\_

t min + seconds	$V_o$ (kt)	$H_o$ Po	Fuel used or Fuel remaining (counts)	OAT <sub>o</sub> (°C)	Fuel Flow (gpm or pph)

Figure 2  
 Sample Format for Computer Data Submission

### DATA REDUCTION AND OUTPUT

After both data sheets (Figures 1 and 2) have been completed, these sheets will be delivered to CSD. Punched cards are then prepared by data reduction personnel; these cards are used for computer inputs. After the computer is programmed and the data reduction is completed, the results will be printed out automatically. The results will be presented in tabular form and will be plotted automatically. A typical automatic plot is shown in Figure 3.



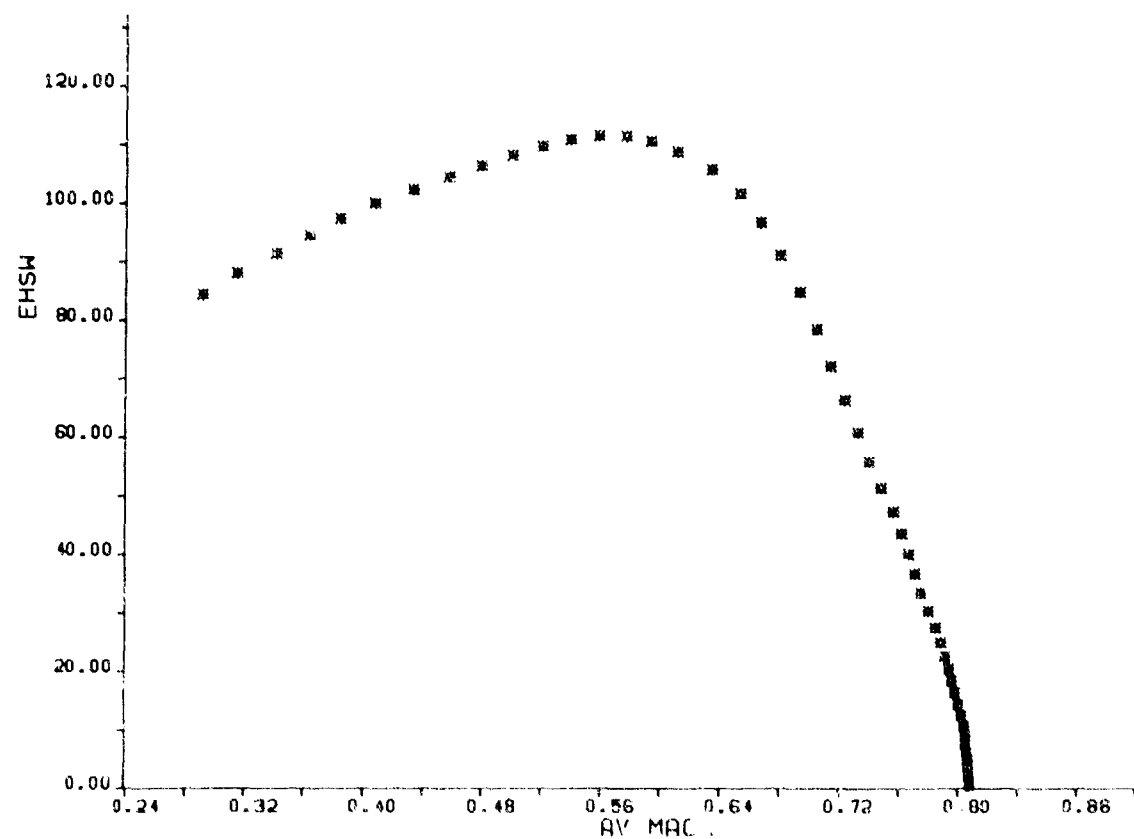


Figure 3  
Typical Computer Printout from Accel Run

#### NOTATIONS

The notations used in the CSD data printout are:

HPO	- Observed Altitude - Feet
HPI	- Corrected for Instrument Error - Feet
HPC	- Corrected for Position Error - Feet
PAI	- Static Pressure from $h_{P_i}$ - lbs/ft <sup>2</sup>
PA	- Static Pressure from $h_{P_c}$ - lbs/ft <sup>2</sup>
VO	- Observed Airspeed - knots - $V_o$

VI	- Corrected for Instrument Error - knots - $V_i$
VC	- Corrected for Position Error - knots - $V_c$
QC	- Impact pressure from $V_c$ - $\text{lbs/ft}^2$
QCI	- Impact pressure from $V_i$ - $\text{lbs/ft}^2$
OAT	- Outside Air Temperature - $^{\circ}\text{C}$ or $^{\circ}\text{F}$
TI	- Corrected for Instrument Error
MI	- $f(q_{c_i}/p_{a_i})$
MACH	- $f(q_c/p_a)$
TA	- Temperature Ambient in $^{\circ}\text{C}$
TAS	- Temperature for Standard Day in $^{\circ}\text{C}$
DT	- $T_a - T_{a_s}$ $^{\circ}\text{C}$
VTs	- True Airspeed for Standard Day - $V_{T_s}$
VT	- True Airspeed for Test Day - $V_{T_t}$
SRT (TA/TAS)	- Square Root of $T_a/T_{a_s}$
FT	- Observed Fuel Temperature
FTI	- Corrected for Instrument Error
WF	- Observed Fuel Flow
WFI	- Corrected for Instrument Error
FUEL CN	- Observed fuel used or fuel remaining (gals or lbs)
FCI	- Corrected for Instrument Error
FUEL CN LB	- Fuel counter converted to lbs
WT	- TO Gross weight - Fuel used (lbs) - empty weight + fuel remaining
RPM	- Observed RPM
RPMI	- Corrected for Instrument Error

RPMC	- Corrected for Tachometer Error
DFN	- Thrust Correction - $\Delta F_n$
DR/C 1	- $(F_n \times V_{TS} \times 101.3) / W_1$
R/CO	- Observed rate of climb
(COSO)SQ	- Cosine of the climb angle squared
$C_L$	- Lift Coefficient
(B)SQE	- Wing span <sup>2</sup> x Efficiency Factor ( $b^2 e$ )
WS	- Varying weight standard - calculated for fuel used or constant $W_s$
DR/C 2	- $\frac{69877.0 \times V_{TS} \times \cos^2 \times (W_s^2 - W_T^2)}{b^2 e \times p_a \times V_{TS} \times W_s}$ - induced drag correction
RC/S	- $RCO + (DRC\ 1 + DRC\ 2) \times .5$
AV HPC	- $(HPC_n + HPC_{n+1}) \times .5$
RCSW	- $RC/S \times (WT/WS) - DR/C\ 2 - \text{weight correction}$

SECTION XIV  
TPS AIRPLANE DETAILS

AIRPLANE DETAILS

XIV - 2	T-2C
4	TA-4J
6	T-38A
8	A-7C
10	OV-1B

## MODEL T-2C AIRPLANE

### GENERAL

Loading Condition: Normal Trainer (one or two pilots)

T.O. Gross Weight: Approx 13,180 lb

Standard Gross Weight: 11,200 lb

Fuel Capacity: 696 gal/4,700 lb JP-5

Wing Span: 38.1 ft Area: 254.9 ft<sup>2</sup> Aspect Ratio: 5.07

### POWER PLANT INFORMATION

Engine: Two J85-GE-4

Ratings:

Condition.	Design Static SL Thrust (uninstalled-lb)	RP-1 %	Max EGT °C
MIL	2,950 (per engine)	101.7 maximum	732
NORM		99	718

### CONFIGURATIONS

#### Cruise (CR)

Landing gear and flaps UP, speedbrakes CLOSED, thrust for level flight at trim airspeed.

#### Power (P)

Landing gear and flaps UP, speedbrakes CLOSED, MIL thrust.

Glide (G)

Landing gear and flaps UP, speedbrakes CLOSED, Idle thrust.

Power Approach (PA)

Landing gear DOWN, flaps FULL, speedbrakes EXTENDED, thrust for level flight at trim airspeed.

Landing (L)

Landing gear DOWN, flaps FULL, speedbrakes EXTENDED, Idle thrust.

Wave-off (WO)

Landing gear DOWN, flaps FULL, speedbrakes CLOSED, MIL thrust.

Takeoff (TO)

Landing gear DOWN, flaps 1/2, speedbrakes CLOSED, MIL thrust (takeoff trim).

## MODEL TA-4J AIRPLANE

### GENERAL

Internal Fuel Capacity: 660 gal/4,490 lb (JP-5)

T.O. Gross Weight: Approx 18,000 lb

Standard Gross Weight: 15,500 lb

### POWER PLANT INFORMATION

Engine: Pratt and Whitney J-52-P6A/E

Condition	Design Static SL Thrust (uninstalled)-lb	RPM %	EGT °C
ALL	8,500	97-100	610
GORI	7,450	MIL-3	590

Engine: Pratt and Whitney J-52 - P-8B

Condition	Static Thrust (lb)	RPM %	EGT °C
ALL	9,300	97-100	650
GORI	8,250	MIL-3	595

### CONFIGURATIONS

#### Cruise (CR)

Landing gear and flaps UP, speedbrakes CLOSED, thrust for level flight at trim airspeed.



Power (P)

Landing gear and flaps UP, speedbrakes CLOSED, MIL thrust.

Glide (G)

Landing gear and flaps UP, speedbrakes CLOSED, Idle thrust.

Power Approach (PA)

Landing gear DOWN, flaps FULL, speedbrakes EXTENDED, thrust for level flight at trim airspeed.

Wave-off (WO)

Landing gear DOWN, flaps FULL, speedbrakes CLOSED, MIL thrust.

Takeoff (TO)

Landing gear DOWN, flaps 1/2, speedbrakes CLOSED, MIL thrust (takeoff trim).

## MODEL T-38A AIRPLANE

### GENERAL

Loading Condition: Normal Trainer (one or more pilots)

T.O. Gross Weight: Approx 12,000 lb

Standard Gross Weight: 9,950 lb

Fuel Capacity: 583 gal/3,750 lb (JP-4)

Wing Span: 25.25 ft Area (S): 170 ft<sup>2</sup> Aspect Ratio: 3.75

### POWER PLANT INFORMATION

Engine: Two J85-GE-5

Ratings:

Condition	Design Static SL Thrust/Engine (uninstalled/ installed) lb	Maximum Steady State EGT °C	RPM %
MAX A/B	3,850/2,900	645	99.5-104
MIL	2,680/2,050	645	99.5-104

### CONFIGURATIONS

#### Cruise (CR)

Landing gear and flaps UP, speedbrakes CLOSED, thrust for level flight at trim airspeed.

#### Power (P)

Landing gear and flaps UP, speedbrakes CLOSED, MIL thrust.

Combat (CO)

Landing gear and flaps UP, speedbrakes CLOSED, MAX A/B thrust.

Glide (G)

Landing gear and flaps UP, speedbrakes CLOSED, Idle thrust.

Power Approach (PA)

Landing gear DOWN, flaps 100%, speedbrakes CLOSED, thrust for level flight at trim airspeed.

Landing (L)

Landing gear DOWN, flaps 100%, speedbrakes CLOSED, Idle thrust.

Wave-off (WO)

Landing gear DOWN, flaps 100%, speedbrakes CLOSED, MIL thrust.

Takeoff (TO)

Landing gear DOWN, flaps 60%, speedbrakes CLOSED, MAX A/B thrust.

## MODEL A-7C AIRPLANE

### GENERAL

Internal Fuel Capacity: 10,172 lb/1,496 gal (JP-5)

T.O. Gross Weight: Approx 29,500 lb

Standard Gross Weight: 26,000 lb

### POWER PLANT INFORMATION

Engine: Pratt and Whitney TF-30-P408

Condition	Design Static SL Thrust (uninstalled)-lb	RPM	EGT °C
MIL	13,900	97.3	1160
N 80	12,500	-	1077

### CONFIGURATIONS

#### Cruise (CR)

Landing gear and flaps UP, speedbrakes CLOSED, thrust for level flight at trim airspeed.

#### Power (P)

Landing gear and flaps UP, speedbrakes CLOSED, MIL thrust.

Glide (G)

Landing gear and flaps UP, speedbrakes CLOSED, Idle thrust.

Power Approach (PA)

Landing gear DOWN, flaps FULL, speedbrakes CLOSED, thrust for level flight (or 3 deg glide slope) at trim airspeed.

Wave-off (WO)

Landing gear DOWN, flaps FULL, speedbrakes CLOSED, MIL thrust.

Takeoff (TO)

Landing gear DOWN, flaps 25 deg, speedbrakes CLOSED, MIL thrust (takeoff trim).

## MODEL OV-1B AIRPLANE

### GENERAL

Loading Condition: Normal Observation

T.O. Gross Weight: Approx 13,300 lb

Standard Gross Weight: 12,500 lb

Fuel Capacity: (JP-4) 1,930 lb (no external)

3,880 lb (internal and external)

Wing Span: 48 ft

### POWER PLANT INFORMATION

Engine: Two Lycoming T-53-L-7

Ratings:

Condition	SHP	RPM	Torque	EGT max	N <sub>1</sub> max
Takeoff	1,160	1,700	97 (max)	640°	101.5
MIL	1,160	1,700	97 (max)	640°	
CORE	900	1,600	80	620°	

Difference in TO and MIL is in throttle detent only. Five min max operation in TO, 30 min max operation in MIL.

### CONFIGURATIONS

#### Cruise (CR)

Landing gear and flaps retracted, power for level flight.

Power (P)

Landing gear and flaps retracted, MIL power.

Glide (G)

Landing gear and flaps retracted, flight idle detent, min RPM.

Power Approach (PA)

Landing gear DOWN, flaps 45 deg, max RPM, torque as required for level flight or fixed as required.

Landing (L)

Landing gear DOWN, flaps 45 deg, max RPM, flight idle detent.

Wave-off (WO)

Landing gear DOWN, flaps 45 deg, TO power

Takeoff (TO)

Landing gear DOWN, flaps 15 deg, TO power, trim 1 deg, ND, 5 deg RWD, 5 deg NR.

SECTION XV  
COCKPIT EVALUATION



## REFERENCES

### Section XV

1. Anthropometry of Naval Aviators, 1964, NAEC-ACEL-533.
2. Cockpit Anthropometric Survey of Model A-4C, A-6A, A-7E, AV-8A, F-4J, F-8D, and OV-10A Airplanes, ST-120R-71.
3. Investigation of A-4 Aircraft Escape System Clearance Envelope, ST-53R-72.
4. MIL-STD-1472A, Human Engineering Design Criteria for Military Systems, Equipment, and Facilities.
5. MIL-A-8806, General Specification for Acoustical Noise Levels in Aircraft.

## COCKPIT EVALUATION

### INTRODUCTION

The purpose of conducting a cockpit evaluation at TPS is to acquaint the student with the evaluation of human engineering design requirements as related to aircraft testing. Because the cockpit is the focal point of the man-machine interface, it is the area which should receive maximum human engineering design emphasis. Unfortunately, many individual cockpit items receive little or no emphasis concerning human accommodation or compatibility. It is often not until all cockpit items are assembled into a mockup or actual cockpit during design and testing evaluations that human engineering design deficiencies become identifiable. Not until the working relationships of controls, displays, lighting, and cockpit environment are analyzed can an intelligent evaluation be conducted. Regardless of airplane performance and flying quality potential, the man in the cockpit must accomplish the transfer function of changing airplane potential into reality. Important variables which control this transfer function are (1) capabilities and limitations of the aviator and (2) the cockpit design which is the link between action and reaction performed by the pilot as a result of information received and processed.

The primary airplane mission must be emphasized during human engineering evaluation. Additionally, "worst case" events must be considered such as degraded system operation and in-flight emergencies.

### TEST PROCEDURES

The most important general principle to keep in mind when performing any human engineering evaluation is the concept of individual differences. We all tend

to evaluate items such as controls and displays from a very subjective viewpoint, i.e., our own capabilities, limitations, and experience. The fact that pilots are all different must be thoroughly understood when evaluating cockpits. Test pilots, in particular, are perhaps not always entirely objective when it comes to admitting that particular human engineering design deficiencies are a problem as far as they are concerned. The following differences are among the most important in cockpit evaluation.

### ANTHROPOMETRY

Body sizes vary considerably. Reference 1 is a compilation of 96 body dimensions based on measurements of 1,549 Naval Aviators. The data are presented in inches and centimeters as well as percentiles.

Detail airplane specifications generally require that cockpits accommodate 5th through 95th percentile sized aviators for older airplane cockpits and crew stations (prior to 1970) and that 3rd through 98th percentile be accommodated in newer cockpits (since 1970). It is generally assumed that if one's body measurements, such as height and weight, are 50th percentile (average) that all his dimensions will be 50th percentile; this is not true. Uniformity in body dimensions is very rare; e.g., it is doubtful that if a person's sitting eye height is 70th percentile that his functional reach will also be 70th percentile. It is important to know one's own percentile ranks of body dimensions.

Physiological Training Units (altitude chambers) are equipped with anthropometric measuring devices where you can be measured and have your measurements translated into percentile ranks. The most important dimensions relative to aircrew station design are:

1. Total sitting height.
2. Sitting eye height.
3. Sitting shoulder height.
4. Biddetold diameter (shoulder width).
5. Functional reach (grasp between thumb and forefinger).
6. Fingertip reach ("push-button" reach with extended forefinger).
7. Buttocks-to-knee length (sitting).

Only by knowing your own various percentile ranks can you make relative judgments as to the overall anthropometric accommodation of a particular cockpit; e.g., if you know your functional reach is 35th percentile and that you cannot reach a particular control when fully restrained, you therefore know that anyone with a functional reach less than 35th percentile, when fully restrained, also cannot reach the control.

Equipments exist which can objectively measure anthropometric parameters such as reach distances, angles of vision, and ejection seat egress clearances (References 2 and 3).

Egress clearances in ejection seat cockpits are often jeopardized when modifications such as cameras, control boxes, or other equipments, are added to canopy rails, glare shields, etc. Human engineering personnel are prepared to use particular equipments to attain quantitative data in cockpit anthropometry evaluations.

NOTE: It is critical that the Design Eye Position (DEP) be the source of measurements for anthropometric evaluations. The DEP is the point in space where the pilot's eyes should be to see all displays and have adequate exterior

vision. To further define the DEP, other preliminary definitions are in order and are presented below.

a. Seat Reference Point is a center line intersection of the seat back tangent line and seat surface.

b. Neutral Seat Reference Point (NSRP) is the location of the seat reference point when the seat is adjusted to the mid-point of vertical adjustment; e.g., with 5 in. of vertical seat travel available, the seat would be adjusted to 2.5 in. above the lower limit.

The DEP is then defined as the point in space located at the sitting eye height dimension of the 50th percentile average aviator (31.5 in.) measured vertically above the NSRP and 13 in. measured horizontally forward of the seat back tangent line. All anthropometric evaluations must originate at the DEP. Whatever the size of the individual evaluating items, such as control reach, display visibility, or cockpit space accommodation, the seat must be adjusted to place his eyes at the DEP. The necessity of adjusting the eyes to the DEP when making anthropometric evaluations is more critical now than ever with the increasing emphasis on heads-up displays and other optical devices which require strict adherence to line-of-sight criterion.

As with all human engineering evaluations, anthropometry must be checked against "worst case" conditions. An example of a "worst case" condition would be reaching for a critical control such as the emergency stores jettison when fully restrained (shoulder harness locked) and when under a high-g condition such as a catapult launch.

Additional items of anthropometric deficiency include insufficient sitting height, inability to reach rudder pedals or foot controls, inability to fit through emergency egress openings, etc.

## CONTROL DESIGN

Controls must meet various criteria to be satisfactory; these criteria include:

### 1. Proper Location

The criticality of control function establishes the priority of location within a crew station. The most important controls should be the easiest to reach and manipulate. Controls should never be located such that the hand or arm manipulating the control is in the line of sight required to see the display effect or setting of the control.

### 2. Natural Direction-of-Motion Relationships

Actuating controls such as toggle switches forward or up should turn systems on. Turning rotary controls clockwise should increase system output. Standard direction-of-motion relationships should be adhered to in cockpit control actuation.

### 3. Shape Coding

Controls which may require manipulation without direct visual monitoring should feel different to the touch if they are near controls of dissimilar systems.

### 4. Inadvertent Actuation

Controls which can be activated incorrectly should be designed to prevent such activation either by electronic circuitry or mechanical guards; e.g., forward wing-sweep actuation during supersonic flight regimes which would potentially be damaging to airplane components should be electrically or mechanically prevented.

Care should be exercised in determining proper safeguards to prevent inadvertent actuation of controls, switches, etc., which might be actuated by flight clothing or items of personal equipment, such as survival vests, flotation devices, anti-exposure garments, etc.

#### 5. Actuation Feedback

Controls should have proper tactile cues relative to actuation. One should "feel" the click of a toggle switch or push button without necessarily hearing it. Controls should have the proper resistance and range of displacement as specified by Reference 4.

### DISPLAYS

The content and format of displays should be limited to essential information but should not require mental computation or translation to be usable. There must be provisions within displays to alert an operator to display failure if the failure is not immediately obvious.

The location and arrangement of displays shall be assigned priority relative to importance for normal and emergency operations. Other criteria, such as viewing distances, grouping, and transilluminated display requirements, are specified in Reference 4.

### LABELING

Items of equipment which must be identified, manipulated, or located should be adequately labeled to permit efficient human performance. Blueprints which illustrate control panels often portray a straight-on-view. However, when the control panel is installed in a cockpit, it is often offset from direct line of sight. The three dimensional line-of-sight offset often results in labels (numbers, ON/OFF

legends, or other nomenclature) being obscured by the very controls to which they are related.

It is important to evaluate labeling legibility in low ambient light (dark) conditions as well as in daylight. If an item must be labeled for normal daylight use, it should be legible at night.

### ENVIRONMENT

Heating, ventilation, and air conditioning shall be evaluated and compared with the criterion specified in Reference 4 or in the applicable specification listed in the detail specification of the airplane being evaluated. Hand-held instruments are available to measure temperature as well as relative humidity. Specifications generally require an Environmental Control System (ECS) to maintain between 60 and 80°F ambient temperature in a crew station and 10°F maximum differential between head and foot level.

Interior ambient air should also be sampled throughout the flight regime or mission profile of any aircraft. Carbon monoxide or other toxic fumes can be potential hazards, particularly during operations such as taxiing downwind, gun or rocket firing, and during refueling operations when directly behind a tanker.

Noise is the most serious and persistent problem among those associated with aircraft environment. The maximum allowable noise limits relative to aircraft type are described in Reference 5. It should be recognized that high noise levels of less intensity than those specified as physically damaging to hearing can produce human fatigue and degrade an aviator's effectiveness.



As a project officer, you should evaluate ambient exterior noise which maintenance or deck personnel are exposed to as a result of being in the immediate vicinity of the aircraft during ground operations. Maintenance personnel often neglect the required ear protection because hearing loss is a slow insidious process.

Various levels of instrumentation are available in evaluating the acoustical environment ranging from small pocket sized decibel meters to sophisticated type recording devices that record noise samples which can be analyzed in detail for various frequency bands.

NOTE: When conducting an interior noise survey, exercise any additional equipments which may increase the acoustic level, such as air conditioning or defogging systems, heater blowers, ambient air vents, and the extended configuration of in-flight refueling probes.

### LIGHTING

A concerted effort in the evaluation of cockpit lighting usually identifies numerous lighting deficiencies. Often there is little emphasis on lighting evaluation. Typically, during night flights general lighting observations are made by crewmen who are busy flying or conducting other airborne tasks, thereby overlooking numerous lighting deficiencies.

A particular procedure which has been effective in static lighting evaluation is described below.

1. Get into the airplane attired in the complete complement of proper flight clothing and equipment (take a tape recorder with you).

2. Have the canopy covered with an opaque cover preventing any ambient light from entering. This allows you to conduct the evaluation day or night.

3. Have electrical power supplied to the aircraft to enable interior light actuation.

4. Adjust your seat to place your eyes in the design eye position (or where you normally fly).

5. Allow your eyes to become adjusted to the dark (10 to 15 min).

6. Begin by locating the auxiliary light (if you can find it in the dark) and see if it is suitable for minimum illumination if all other lights were lost.

7. After the auxiliary light evaluation, systematically exercise all light controls in the cockpit. Vary the intensity, look for instrument lights on a particular rheostat which extinguish before others when adjusting from bright to OFF. Look for brightness imbalance such that, at a given light adjustment, some instruments may be too bright or too dim when most other instruments on that particular lighting control are at a reasonable intensity. Identify any glare or reflection which might possibly be shielded.

8. Verbally record on your tape recorder any deficiencies noted; this allows evaluation uninterrupted by turning on floodlights or flashlights to write down deficiencies which in turn would require readjustment of the eyes to low ambient light.

9. Adjust your seat to various positions to determine if lighting is sufficient throughout a typical range of particular eye locations. This may be one of the few times you evaluate lighting strictly for its own sake. Take as much time as is required to evaluate all lighting variations, legibility of labels, visibility of controls, etc.

## SPECIFICATIONS

A partial listing of Human Engineering Specifications is listed below which may be of assistance in establishing Military Specification noncompliance.

MIL-STD-1472	Human Engineering Design Criteria for Systems, Equipment, and Facilities
MIL-STD-203E	Aircrew Station Controls and Displays for Fixed Wing Aircraft
MIL-STD-250C	Aircrew Station Controls and Displays for Rotary Wing Aircraft
MIL-STD-411D	Aircrew Station Signals
SD24J Vol. 1	General Specification for Design and Construction of Aircraft Weapons Systems, Fixed Wing Aircraft
SD24J Vol. 2	General Specification for Design and Construction of Aircraft Weapons Systems, Rotary Wing Aircraft

**APPENDIXES**

## APPENDIXES

- I. ICAO Standard Atmosphere Tables
- II. Compressibility Correction
- III. Airspeed - Altitude - Mach Number Crossplot
- IV. Mach Number - Indicated Temperature - Recovery Factor - Ambient Temperature
- V. Centigrade - Fahrenheit Temperature Conversion
- VI. Calibrated Airspeed - Energy Relations
- VII. Static Pressure versus Pressure Altitude\*
- VIII. Impact Pressure ( $q_c$ ) versus Calibrated Airspeed\*

\*Derived from NASA TN D 822

## STANDARD ATMOSPHERE TABLES

### Sea Level Values

Pressure: 14.7 psi, 2,116 psf, 29.92 in Hg, 1,013 mb

Temperature: 59°F    15°C    518.69°Rankine    288.16°K

Density: .0023769 slugs/cu ft

Sonic Velocity: 661.48 kt, 1,116.45 fps

Gravitational Acceleration: 32.174 ft/sec/sec

Temperature lapse rate to 36,089 ft

-3.57°F per 1,000 ft

-1.98°C per 1,000 ft

Sonic Velocity:

$$a = a_{sl} \left( \frac{T}{T_{sl}} \right)^{1/2}$$

Tropopause - at 36,089 ft geopotential altitude, T = 216.66°K

Density Ratio

$$\sigma = \frac{\rho}{\rho_{sl}}$$

Temperature Ratio

$$\theta = \frac{T}{T_{sl}}$$

Pressure Ratio

$$\delta = \frac{P}{P_{sl}}$$

Viscosity Ratio

$$\eta = \frac{\mu}{\mu_{sl}}$$

# STANDARD ATMOSPHERE

Altitude (ft)	Pressure (in Hg)	Temperature (°F)	Temperature (°C)	ρ (g/cm³)	σ (lb/ft²)	√σ	1/√σ	√g/σ	1/√g/σ	ρ/σ	ρ/φ√g
0	30.10	59.0	15.0	0.001225	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
100	29.92	58.8	14.9	0.001224	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
200	29.74	58.6	14.8	0.001223	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
300	29.56	58.4	14.7	0.001222	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
400	29.38	58.2	14.6	0.001221	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
500	29.20	58.0	14.5	0.001220	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
600	29.02	57.8	14.4	0.001219	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
700	28.84	57.6	14.3	0.001218	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
800	28.66	57.4	14.2	0.001217	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
900	28.48	57.2	14.1	0.001216	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
1000	28.30	57.0	14.0	0.001215	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
1100	28.12	56.8	13.9	0.001214	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
1200	27.94	56.6	13.8	0.001213	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
1300	27.76	56.4	13.7	0.001212	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
1400	27.58	56.2	13.6	0.001211	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
1500	27.40	56.0	13.5	0.001210	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
1600	27.22	55.8	13.4	0.001209	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
1700	27.04	55.6	13.3	0.001208	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
1800	26.86	55.4	13.2	0.001207	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
1900	26.68	55.2	13.1	0.001206	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
2000	26.50	55.0	13.0	0.001205	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
2100	26.32	54.8	12.9	0.001204	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
2200	26.14	54.6	12.8	0.001203	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
2300	25.96	54.4	12.7	0.001202	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
2400	25.78	54.2	12.6	0.001201	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
2500	25.60	54.0	12.5	0.001200	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
2600	25.42	53.8	12.4	0.001199	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
2700	25.24	53.6	12.3	0.001198	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
2800	25.06	53.4	12.2	0.001197	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
2900	24.88	53.2	12.1	0.001196	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
3000	24.70	53.0	12.0	0.001195	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
3100	24.52	52.8	11.9	0.001194	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
3200	24.34	52.6	11.8	0.001193	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
3300	24.16	52.4	11.7	0.001192	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
3400	23.98	52.2	11.6	0.001191	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
3500	23.80	52.0	11.5	0.001190	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
3600	23.62	51.8	11.4	0.001189	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
3700	23.44	51.6	11.3	0.001188	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
3800	23.26	51.4	11.2	0.001187	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
3900	23.08	51.2	11.1	0.001186	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
4000	22.90	51.0	11.0	0.001185	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
4100	22.72	50.8	10.9	0.001184	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
4200	22.54	50.6	10.8	0.001183	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
4300	22.36	50.4	10.7	0.001182	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
4400	22.18	50.2	10.6	0.001181	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
4500	22.00	50.0	10.5	0.001180	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
4600	21.82	49.8	10.4	0.001179	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
4700	21.64	49.6	10.3	0.001178	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
4800	21.46	49.4	10.2	0.001177	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
4900	21.28	49.2	10.1	0.001176	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
5000	21.10	49.0	10.0	0.001175	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
5100	20.92	48.8	9.9	0.001174	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
5200	20.74	48.6	9.8	0.001173	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
5300	20.56	48.4	9.7	0.001172	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
5400	20.38	48.2	9.6	0.001171	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
5500	20.20	48.0	9.5	0.001170	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
5600	20.02	47.8	9.4	0.001169	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
5700	19.84	47.6	9.3	0.001168	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
5800	19.66	47.4	9.2	0.001167	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
5900	19.48	47.2	9.1	0.001166	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
6000	19.30	47.0	9.0	0.001165	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
6100	19.12	46.8	8.9	0.001164	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
6200	18.94	46.6	8.8	0.001163	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
6300	18.76	46.4	8.7	0.001162	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
6400	18.58	46.2	8.6	0.001161	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
6500	18.40	46.0	8.5	0.001160	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
6600	18.22	45.8	8.4	0.001159	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
6700	18.04	45.6	8.3	0.001158	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
6800	17.86	45.4	8.2	0.001157	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
6900	17.68	45.2	8.1	0.001156	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
7000	17.50	45.0	8.0	0.001155	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
7100	17.32	44.8	7.9	0.001154	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
7200	17.14	44.6	7.8	0.001153	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
7300	16.96	44.4	7.7	0.001152	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
7400	16.78	44.2	7.6	0.001151	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
7500	16.60	44.0	7.5	0.001150	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
7600	16.42	43.8	7.4	0.001149	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
7700	16.24	43.6	7.3	0.001148	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
7800	16.06	43.4	7.2	0.001147	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
7900	15.88	43.2	7.1	0.001146	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
8000	15.70	43.0	7.0	0.001145	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
8100	15.52	42.8	6.9	0.001144	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
8200	15.34	42.6	6.8	0.001143	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
8300	15.16	42.4	6.7	0.001142	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
8400	14.98	42.2	6.6	0.001141	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
8500	14.80	42.0	6.5	0.001140	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
8600	14.62	41.8	6.4	0.001139	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
8700	14.44	41.6	6.3	0.001138	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
8800	14.26	41.4	6.2	0.001137	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
8900	14.08	41.2	6.1	0.001136	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
9000	13.90	41.0	6.0	0.001135	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
9100	13.72	40.8	5.9	0.001134	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
9200	13.54	40.6	5.8	0.001133	1.0000	1.0000	1.0000	1.0000			





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APPENDIX I





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APPENDIX Y

H (FEET)	P <sub>0</sub> (W.G.)	δ	1/δ	T <sub>0</sub> (°K)	v (KNOTS)	θ	√θ	σ	√σ	1/√σ	√δ/8	1/√δ	1/8√δ	φ	δ/φ√θ
21000	12.107	4045	2.471	242.60	606.90	8415	9175	4806	6933	1.4424	2.2574	1.4424	2.693	.870	.5055
21100	12.253	4025	2.431	242.45	606.70	8412	9172	4789	6921	1.4419	2.2564	1.4419	2.706	.860	.5052
21200	12.403	4011	2.422	242.20	606.44	8405	9168	4773	6909	1.4414	2.2553	1.4414	2.719	.857	.5052
21300	12.552	3994	2.503	242.00	606.18	8398	9164	4756	6897	1.4409	2.2941	1.4409	2.731	.857	.5052
21400	12.701	3977	2.514	241.80	605.91	8391	9160	4740	6885	1.4404	2.3030	1.4404	2.744	.857	.5052
21500	12.850	3960	2.515	241.59	605.71	8384	9157	4723	6873	1.4400	2.3122	1.4400	2.757	.857	.5052
21600	12.998	3943	2.515	241.39	605.45	8377	9153	4707	6861	1.4395	2.3212	1.4395	2.770	.865	.4959
21700	13.148	3925	2.516	241.22	605.18	8371	9149	4690	6849	1.4391	2.3302	1.4391	2.783	.864	.4957
21800	13.297	3908	2.517	241.02	604.92	8364	9145	4674	6837	1.4386	2.3392	1.4386	2.797	.864	.4957
21900	13.446	3892	2.517	240.82	604.72	8357	9142	4657	6825	1.4382	2.3486	1.4382	2.810	.863	.4957
22000	13.595	3875	2.518	240.61	604.46	8350	9138	4641	6813	1.4378	2.3578	1.4378	2.823	.863	.4957
22100	13.744	3859	2.518	240.41	604.19	8343	9134	4625	6801	1.4374	2.3669	1.4374	2.837	.862	.4956
22200	13.893	3842	2.519	240.21	603.91	8335	9130	4609	6789	1.4370	2.3762	1.4370	2.850	.862	.4956
22300	14.042	3825	2.520	240.01	603.60	8328	9126	4593	6777	1.4366	2.3854	1.4366	2.864	.861	.4956
22400	14.191	3809	2.521	239.81	603.46	8322	9123	4577	6765	1.4362	2.3941	1.4362	2.877	.861	.4956
22500	14.340	3792	2.522	239.61	603.20	8315	9119	4560	6753	1.4358	2.4034	1.4358	2.891	.860	.4956
22600	14.489	3775	2.523	239.41	602.93	8307	9115	4544	6742	1.4354	2.4132	1.4354	2.905	.859	.4956
22700	14.638	3758	2.524	239.21	602.67	8302	9112	4528	6730	1.4350	2.4231	1.4350	2.919	.859	.4956
22800	14.787	3742	2.525	239.01	602.47	8295	9108	4512	6718	1.4346	2.4333	1.4346	2.933	.859	.4956
22900	14.936	3727	2.525	238.81	602.21	8288	9104	4497	6706	1.4342	2.4442	1.4342	2.947	.858	.4956
23000	15.085	3710	2.526	238.61	601.94	8281	9100	4481	6694	1.4339	2.4552	1.4339	2.961	.858	.4956
23100	15.234	3693	2.527	238.41	601.68	8274	9096	4465	6682	1.4335	2.4661	1.4335	2.975	.857	.4956
23200	15.383	3676	2.528	238.22	601.48	8267	9093	4449	6671	1.4331	2.4771	1.4331	2.990	.857	.4956
23300	15.532	3662	2.529	238.02	601.22	8261	9089	4433	6659	1.4327	2.4881	1.4327	3.004	.855	.4956
23400	15.681	3646	2.530	237.85	600.95	8254	9085	4418	6647	1.4324	2.4991	1.4324	3.018	.855	.4956
23500	15.830	3630	2.531	237.65	600.69	8247	9081	4402	6635	1.4320	2.5101	1.4320	3.033	.854	.4956
23600	15.979	3614	2.532	237.44	600.42	8240	9077	4387	6623	1.4317	2.5211	1.4317	3.047	.854	.4956
23700	16.128	3598	2.533	237.24	600.22	8233	9074	4371	6612	1.4314	2.5321	1.4314	3.062	.853	.4956
23800	16.277	3583	2.534	237.04	599.96	8226	9070	4355	6600	1.4311	2.5431	1.4311	3.077	.853	.4956
23900	16.426	3567	2.535	236.84	599.69	8219	9066	4340	6588	1.4307	2.5541	1.4307	3.092	.852	.4956
24000	16.575	3551	2.536	236.64	599.43	8212	9062	4324	6576	1.4304	2.5651	1.4304	3.106	.852	.4956
24100	16.724	3536	2.537	236.45	599.17	8205	9058	4309	6565	1.4301	2.5761	1.4301	3.121	.850	.4956
24200	16.873	3520	2.538	236.26	598.97	8199	9055	4294	6553	1.4298	2.5871	1.4298	3.137	.850	.4956
24300	17.022	3505	2.539	236.06	598.70	8192	9051	4278	6541	1.4295	2.5981	1.4295	3.152	.849	.4956
24400	17.171	3490	2.540	235.86	598.44	8185	9047	4263	6530	1.4292	2.6091	1.4292	3.167	.848	.4956
24500	17.320	3474	2.541	235.66	598.17	8178	9043	4248	6518	1.4289	2.6201	1.4289	3.182	.848	.4956
24600	17.469	3459	2.542	235.46	597.91	8171	9039	4233	6506	1.4286	2.6311	1.4286	3.197	.847	.4956
24700	17.618	3443	2.543	235.25	597.71	8164	9036	4218	6495	1.4283	2.6421	1.4283	3.212	.847	.4956
24800	17.767	3428	2.544	235.05	597.45	8157	9032	4203	6483	1.4280	2.6531	1.4280	3.227	.847	.4956
24900	17.916	3413	2.545	234.84	597.18	8151	9028	4188	6472	1.4277	2.6641	1.4277	3.242	.847	.4956

M (FEET)	P <sub>0</sub> (W.G.)	δ	V <sub>0</sub>	T <sub>0</sub> (°C)	ρ (GRAMS)	δ	√δ	ρ	√ρ	1/√ρ	√δ/ρ	1/δ√ρ	φ	δ/φ√ρ
27000	10.156	3.194	2.942	234.68	596.92	8144	9024	4173	6460	1.5480	2.655	3.263	.846	.4459
27100	10.161	3.195	2.955	234.48	596.62	8137	9020	4157	6448	1.5509	2.646	3.276		.4434
27200	10.167	3.196	2.968	234.27	596.45	8130	9017	4151	6437	1.5535	2.637	3.292	.845	.4419
27300	10.173	3.197	2.982	234.07	596.19	8123	9013	4128	6425	1.5564	2.627	3.308		.4399
27400	10.178	3.198	2.995	233.87	595.92	8116	9009	4113	6412	1.5593	2.618	3.325	.844	.4378
27500	10.184	3.199	3.008	233.67	595.66	8109	9005	4098	6400	1.5620	2.609	3.341		.4367
27600	10.190	3.200	3.022	233.47	595.40	8102	9001	4083	6390	1.5649	2.600	3.357	.842	.4346
27700	10.196	3.201	3.035	233.27	595.14	8096	8998	4068	6379	1.5676	2.591	3.374		.4326
27800	10.202	3.202	3.049	233.07	594.91	8089	8994	4054	6367	1.5706	2.582	3.390	.841	.4305
27900	10.208	3.203	3.063	232.89	594.67	8082	8990	4040	6356	1.5733	2.573	3.407		.4285
28000	10.214	3.204	3.076	232.69	594.40	8075	8986	4025	6344	1.5761	2.564	3.423	.840	.4264
28100	10.220	3.205	3.090	232.49	594.14	8068	8982	4010	6333	1.5790	2.555	3.440		.4243
28200	10.226	3.206	3.104	232.29	593.88	8061	8978	3996	6321	1.5820	2.546	3.457	.839	.4222
28300	10.232	3.207	3.118	232.08	593.62	8054	8975	3981	6310	1.5848	2.537	3.474		.4201
28400	10.238	3.208	3.132	231.88	593.41	8047	8971	3967	6298	1.5878	2.528	3.491	.838	.4180
28500	10.244	3.209	3.146	231.67	593.15	8041	8967	3952	6287	1.5906	2.519	3.507		.4159
28600	10.250	3.210	3.160	231.47	592.88	8034	8963	3938	6276	1.5934	2.510	3.526	.837	.4138
28700	10.256	3.211	3.175	231.27	592.62	8027	8959	3923	6264	1.5964	2.501	3.543		.4117
28800	10.262	3.212	3.189	231.07	592.35	8020	8955	3909	6253	1.5992	2.492	3.561	.836	.4096
28900	10.268	3.213	3.203	230.90	592.15	8013	8952	3895	6241	1.6021	2.483	3.579		.4075
29000	10.274	3.214	3.218	230.70	591.89	8006	8948	3881	6230	1.6051	2.474	3.596	.835	.4054
29100	10.280	3.215	3.232	230.50	591.62	7999	8944	3867	6219	1.6080	2.465	3.614		.4033
29200	10.286	3.216	3.247	230.30	591.35	7992	8940	3853	6207	1.6111	2.456	3.632	.834	.4012
29300	10.292	3.217	3.262	230.12	591.09	7986	8936	3839	6196	1.6141	2.447	3.650		.4001
29400	10.298	3.218	3.276	229.92	590.82	7979	8932	3825	6184	1.6171	2.438	3.668	.833	.3980
29500	10.304	3.219	3.291	229.72	590.56	7972	8928	3811	6173	1.6200	2.429	3.686		.3959
29600	10.310	3.220	3.306	229.52	590.30	7965	8925	3797	6162	1.6228	2.420	3.705	.832	.3938
29700	10.316	3.221	3.321	229.32	590.10	7958	8921	3782	6150	1.6258	2.411	3.723		.3917
29800	10.322	3.222	3.337	229.12	589.85	7951	8917	3769	6139	1.6289	2.402	3.742	.831	.3896
29900	10.328	3.223	3.352	228.91	589.57	7944	8913	3755	6128	1.6319	2.393	3.761		.3875
30000	10.334	3.224	3.367	228.71	589.30	7937	8909	3741	6117	1.6348	2.384	3.779	.830	.3854
30100	10.340	3.225	3.382	228.54	589.02	7931	8905	3727	6105	1.6380	2.375	3.798		.3833
30200	10.346	3.226	3.398	228.34	588.76	7924	8901	3713	6094	1.6410	2.366	3.817	.829	.3812
30300	10.352	3.227	3.413	228.14	588.50	7917	8898	3699	6082	1.6441	2.357	3.836		.3791
30400	10.358	3.228	3.429	227.93	588.24	7910	8894	3685	6071	1.6471	2.348	3.855	.828	.3770
30500	10.364	3.229	3.445	227.73	587.98	7903	8890	3671	6060	1.6502	2.339	3.874		.3749
30600	10.370	3.230	3.460	227.53	587.72	7896	8886	3657	6049	1.6532	2.330	3.893	.827	.3728
30700	10.376	3.231	3.476	227.33	587.45	7889	8882	3643	6038	1.6562	2.321	3.912		.3707
30800	10.382	3.232	3.492	227.13	587.19	7882	8878	3629	6027	1.6592	2.312	3.931	.826	.3686
30900	10.388	3.233	3.508	226.95	586.95	7875	8874	3615	6016	1.6622	2.303	3.950		.3665
31000	10.394	3.234	3.524	226.75	586.70	7868	8870	3601	6005	1.6652	2.294	3.969	.825	.3644

$\mu$ c.c.t.	$\mu$ (M)	$\delta$	$\frac{1}{\delta}$	$T_0$ (°K)	$\theta$	$\sqrt{\theta}$	$\sigma$	$\sqrt{\sigma}$	$\frac{1}{\sqrt{\sigma}}$	$\frac{\sqrt{\theta}}{\sigma}$	$\frac{1}{\sqrt{\theta}}$	$\frac{1}{\sqrt{\sigma}}$
1.000	8.484	2.836	3.524	224.75	586.80	.8871	3.605	.6004	1.6656	3.127	3.974	.822
1.100	8.449	2.871	3.541	226.53	586.53	.8867	3.592	.5993	1.6686	3.149	3.994	.821
1.200	8.410	2.911	3.557	228.35	586.27	.8863	3.578	.5982	1.6717	3.153	4.016	.821
1.300	8.371	2.956	3.573	230.21	586.01	.8859	3.565	.5971	1.6746	3.166	4.034	.819
1.400	8.331	2.998	3.590	232.09	585.74	.8855	3.551	.5960	1.6779	3.179	4.054	.819
1.500	8.295	3.037	3.607	234.00	585.48	.8851	3.538	.5949	1.6810	3.192	4.075	.818
1.600	8.259	3.074	3.623	235.94	585.21	.8847	3.525	.5938	1.6841	3.205	4.095	.818
1.700	8.218	3.108	3.640	237.91	584.95	.8843	3.512	.5926	1.6875	3.219	4.116	.817
1.800	8.181	3.134	3.657	239.91	584.68	.8839	3.499	.5915	1.6907	3.232	4.137	.817
1.900	8.143	3.159	3.674	241.97	584.41	.8835	3.486	.5904	1.6938	3.246	4.158	.816
2.000	8.105	3.184	3.691	244.06	584.15	.8832	3.473	.5893	1.6969	3.260	4.180	.815
2.100	8.068	3.208	3.708	246.16	583.89	.8828	3.460	.5882	1.7001	3.274	4.201	.814
2.200	8.030	3.232	3.725	248.26	583.64	.8824	3.447	.5871	1.7033	3.287	4.222	.814
2.300	7.993	3.256	3.742	250.36	583.39	.8820	3.434	.5860	1.7065	3.301	4.244	.813
2.400	7.956	3.279	3.759	252.46	583.14	.8816	3.421	.5849	1.7097	3.315	4.265	.813
2.500	7.919	3.302	3.776	254.56	582.89	.8812	3.408	.5838	1.7129	3.329	4.287	.812
2.600	7.882	3.325	3.793	256.66	582.64	.8808	3.395	.5827	1.7161	3.343	4.309	.812
2.700	7.845	3.348	3.810	258.76	582.39	.8804	3.382	.5816	1.7193	3.357	4.331	.811
2.800	7.808	3.371	3.827	260.86	582.14	.8800	3.370	.5805	1.7225	3.371	4.353	.811
2.900	7.771	3.394	3.844	262.96	581.89	.8797	3.357	.5794	1.7257	3.386	4.376	.810
3.000	7.734	3.417	3.861	265.06	581.64	.8793	3.344	.5783	1.7289	3.400	4.398	.809
3.100	7.697	3.440	3.878	267.16	581.39	.8789	3.332	.5772	1.7321	3.414	4.421	.809
3.200	7.660	3.463	3.895	269.26	581.14	.8785	3.319	.5761	1.7353	3.429	4.443	.808
3.300	7.623	3.486	3.912	271.36	580.89	.8781	3.306	.5750	1.7385	3.443	4.466	.807
3.400	7.586	3.509	3.929	273.46	580.64	.8777	3.294	.5739	1.7417	3.458	4.489	.807
3.500	7.549	3.532	3.946	275.56	580.39	.8773	3.281	.5728	1.7449	3.473	4.512	.806
3.600	7.512	3.555	3.963	277.66	580.14	.8769	3.269	.5717	1.7481	3.487	4.535	.806
3.700	7.475	3.578	3.980	279.76	579.89	.8765	3.256	.5706	1.7513	3.502	4.559	.805
3.800	7.438	3.601	4.000	281.86	579.64	.8761	3.244	.5695	1.7545	3.517	4.582	.805
3.900	7.401	3.624	4.019	283.96	579.39	.8757	3.232	.5685	1.7577	3.532	4.606	.804
4.000	7.364	3.647	4.038	286.06	579.14	.8753	3.219	.5674	1.7609	3.546	4.630	.803
4.100	7.327	3.670	4.057	288.16	578.89	.8749	3.207	.5663	1.7641	3.561	4.654	.803
4.200	7.290	3.693	4.076	290.26	578.64	.8745	3.195	.5652	1.7673	3.576	4.678	.802
4.300	7.253	3.716	4.095	292.36	578.39	.8741	3.183	.5642	1.7705	3.591	4.703	.801
4.400	7.216	3.739	4.114	294.46	578.14	.8737	3.171	.5631	1.7737	3.606	4.727	.801
4.500	7.179	3.762	4.133	296.56	577.89	.8733	3.159	.5620	1.7769	3.621	4.752	.800
4.600	7.142	3.785	4.152	298.66	577.64	.8729	3.147	.5609	1.7801	3.636	4.776	.800
4.700	7.105	3.808	4.171	300.76	577.39	.8725	3.135	.5598	1.7833	3.651	4.801	.799
4.800	7.068	3.831	4.190	302.86	577.14	.8721	3.123	.5587	1.7865	3.666	4.826	.799
4.900	7.031	3.854	4.209	304.96	576.89	.8717	3.111	.5576	1.7897	3.681	4.851	.798
5.000	6.994	3.877	4.228	307.06	576.64	.8713	3.100	.5565	1.7929	3.696	4.876	.798

$M$ (FEET)	$\frac{M}{1000}$	$\delta$	$\frac{1}{\delta}$	$\frac{1}{\delta^2}$	$\frac{1}{\delta^3}$	$\frac{1}{\delta^4}$	$\frac{1}{\delta^5}$	$\frac{1}{\delta^6}$	$\frac{1}{\delta^7}$	$\frac{1}{\delta^8}$	$\frac{1}{\delta^9}$	$\frac{1}{\delta^{10}}$	$\frac{1}{\delta^{11}}$	$\frac{1}{\delta^{12}}$	$\frac{1}{\delta^{13}}$	$\frac{1}{\delta^{14}}$	$\frac{1}{\delta^{15}}$	$\frac{1}{\delta^{16}}$	$\frac{1}{\delta^{17}}$	$\frac{1}{\delta^{18}}$	$\frac{1}{\delta^{19}}$	$\frac{1}{\delta^{20}}$	$\frac{1}{\delta^{21}}$	$\frac{1}{\delta^{22}}$	$\frac{1}{\delta^{23}}$	$\frac{1}{\delta^{24}}$	$\frac{1}{\delta^{25}}$	$\frac{1}{\delta^{26}}$	$\frac{1}{\delta^{27}}$	$\frac{1}{\delta^{28}}$	$\frac{1}{\delta^{29}}$	$\frac{1}{\delta^{30}}$	$\frac{1}{\delta^{31}}$	$\frac{1}{\delta^{32}}$	$\frac{1}{\delta^{33}}$	$\frac{1}{\delta^{34}}$	$\frac{1}{\delta^{35}}$	$\frac{1}{\delta^{36}}$	$\frac{1}{\delta^{37}}$	$\frac{1}{\delta^{38}}$	$\frac{1}{\delta^{39}}$	$\frac{1}{\delta^{40}}$	$\frac{1}{\delta^{41}}$	$\frac{1}{\delta^{42}}$	$\frac{1}{\delta^{43}}$	$\frac{1}{\delta^{44}}$	$\frac{1}{\delta^{45}}$	$\frac{1}{\delta^{46}}$	$\frac{1}{\delta^{47}}$	$\frac{1}{\delta^{48}}$	$\frac{1}{\delta^{49}}$	$\frac{1}{\delta^{50}}$	$\frac{1}{\delta^{51}}$	$\frac{1}{\delta^{52}}$	$\frac{1}{\delta^{53}}$	$\frac{1}{\delta^{54}}$	$\frac{1}{\delta^{55}}$	$\frac{1}{\delta^{56}}$	$\frac{1}{\delta^{57}}$	$\frac{1}{\delta^{58}}$	$\frac{1}{\delta^{59}}$	$\frac{1}{\delta^{60}}$	$\frac{1}{\delta^{61}}$	$\frac{1}{\delta^{62}}$	$\frac{1}{\delta^{63}}$	$\frac{1}{\delta^{64}}$	$\frac{1}{\delta^{65}}$	$\frac{1}{\delta^{66}}$	$\frac{1}{\delta^{67}}$	$\frac{1}{\delta^{68}}$	$\frac{1}{\delta^{69}}$	$\frac{1}{\delta^{70}}$	$\frac{1}{\delta^{71}}$	$\frac{1}{\delta^{72}}$	$\frac{1}{\delta^{73}}$	$\frac{1}{\delta^{74}}$	$\frac{1}{\delta^{75}}$	$\frac{1}{\delta^{76}}$	$\frac{1}{\delta^{77}}$	$\frac{1}{\delta^{78}}$	$\frac{1}{\delta^{79}}$	$\frac{1}{\delta^{80}}$	$\frac{1}{\delta^{81}}$	$\frac{1}{\delta^{82}}$	$\frac{1}{\delta^{83}}$	$\frac{1}{\delta^{84}}$	$\frac{1}{\delta^{85}}$	$\frac{1}{\delta^{86}}$	$\frac{1}{\delta^{87}}$	$\frac{1}{\delta^{88}}$	$\frac{1}{\delta^{89}}$	$\frac{1}{\delta^{90}}$	$\frac{1}{\delta^{91}}$	$\frac{1}{\delta^{92}}$	$\frac{1}{\delta^{93}}$	$\frac{1}{\delta^{94}}$	$\frac{1}{\delta^{95}}$	$\frac{1}{\delta^{96}}$	$\frac{1}{\delta^{97}}$	$\frac{1}{\delta^{98}}$	$\frac{1}{\delta^{99}}$	$\frac{1}{\delta^{100}}$																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																		
35000	7.040	2353	4.249	218.83	576.42	7594	8714	3098	5567	1.7923	3.703	4.877	5.932	6.902	7.815	8.683	9.511	10.305	11.071	11.811	12.527	13.221	13.894	14.547	15.181	15.797	16.396	16.979	17.547	18.101	18.642	19.171	19.688	20.194	20.689	21.174	21.649	22.115	22.572	23.021	23.462	23.896	24.323	24.743	25.157	25.565	25.967	26.363	26.753	27.137	27.515	27.888	28.256	28.619	28.977	29.330	29.678	30.021	30.359	30.692	31.020	31.343	31.661	31.975	32.284	32.589	32.890	33.187	33.480	33.769	34.054	34.335	34.612	34.885	35.154	35.419	35.680	35.937	36.190	36.439	36.684	36.925	37.162	37.395	37.624	37.850	38.073	38.292	38.508	38.721	38.931	39.138	39.342	39.543	39.741	39.936	40.128	40.317	40.503	40.686	40.866	41.043	41.217	41.388	41.556	41.721	41.884	42.044	42.201	42.356	42.509	42.660	42.809	42.955	43.100	43.242	43.382	43.520	43.656	43.790	43.922	44.052	44.180	44.306	44.430	44.552	44.672	44.790	44.906	45.021	45.134	45.245	45.355	45.463	45.569	45.673	45.776	45.877	45.977	46.075	46.172	46.268	46.363	46.457	46.549	46.640	46.730	46.819	46.907	47.093	47.278	47.461	47.643	47.824	48.004	48.182	48.359	48.535	48.710	48.884	49.057	49.229	49.400	49.570	49.739	49.906	50.072	50.237	50.401	50.564	50.726	50.887	51.047	51.206	51.363	51.519	51.674	51.828	51.981	52.133	52.284	52.434	52.583	52.731	52.878	53.024	53.169	53.313	53.456	53.598	53.739	53.879	54.018	54.156	54.293	54.429	54.564	54.698	54.831	54.963	55.094	55.224	55.353	55.481	55.608	55.734	55.859	55.983	56.106	56.228	56.349	56.469	56.588	56.706	56.823	56.939	57.054	57.168	57.281	57.393	57.505	57.616	57.726	57.835	57.943	58.050	58.156	58.261	58.366	58.470	58.573	58.676	58.778	58.879	58.980	59.080	59.179	59.277	59.374	59.470	59.566	59.661	59.755	59.849	59.942	60.035	60.127	60.218	60.309	60.400	60.490	60.580	60.669	60.757	60.845	60.932	61.019	61.105	61.191	61.276	61.361	61.445	61.529	61.612	61.695	61.777	61.859	61.940	62.021	62.102	62.182	62.262	62.341	62.420	62.499	62.577	62.655	62.733	62.810	62.887	62.964	63.040	63.116	63.192	63.268	63.343	63.418	63.492	63.566	63.640	63.714	63.787	63.860	63.933	64.006	64.078	64.150	64.222	64.294	64.365	64.436	64.507	64.577	64.647	64.717	64.786	64.855	64.924	64.993	65.061	65.129	65.197	65.265	65.332	65.399	65.466	65.532	65.599	65.665	65.731	65.797	65.862	65.927	65.992	66.056	66.120	66.184	66.248	66.311	66.374	66.437	66.499	66.561	66.623	66.685	66.746	66.807	66.868	66.928	66.988	67.048	67.107	67.166	67.225	67.284	67.342	67.400	67.458	67.516	67.573	67.630	67.687	67.744	67.800	67.857	67.913	67.969	68.025	68.081	68.136	68.191	68.246	68.301	68.356	68.410	68.464	68.518	68.572	68.626	68.679	68.733	68.786	68.839	68.892	68.945	68.998	69.051	69.103	69.156	69.208	69.260	69.312	69.364	69.416	69.467	69.518	69.569	69.620	69.671	69.722	69.773	69.823	69.874	69.924	69.974	70.024	70.074	70.124	70.174	70.224	70.273	70.323	70.372	70.421	70.470	70.519	70.568	70.617	70.665	70.714	70.762	70.810	70.858	70.906	70.954	71.002	71.050	71.097	71.145	71.192	71.239	71.286	71.333	71.380	71.426	71.473	71.519	71.565	71.611	71.657	71.703	71.748	71.794	71.839	71.884	71.929	71.974	72.019	72.064	72.108	72.153	72.197	72.242	72.286	72.330	72.374	72.418	72.462	72.506	72.550	72.594	72.637	72.681	72.724	72.767	72.810	72.853	72.896	72.939	72.982	73.025	73.067	73.110	73.152	73.195	73.237	73.279	73.321	73.363	73.405	73.447	73.488	73.530	73.571	73.613	73.654	73.695	73.736	73.777	73.818	73.858	73.899	73.939	73.979	74.019	74.059	74.099	74.139	74.179	74.218	74.258	74.297	74.337	74.376	74.415	74.454	74.493	74.532	74.571	74.610	74.649	74.688	74.727	74.765	74.804	74.843	74.881	74.920	74.958	74.997	75.035	75.074	75.112	75.150	75.188	75.226	75.264	75.302	75.340	75.378	75.416	75.454	75.492	75.530	75.567	75.605	75.642	75.680	75.717	75.754	75.791	75.828	75.865	75.902	75.939	75.976	76.013	76.050	76.086	76.123	76.159	76.196	76.232	76.268	76.304	76.340	76.376	76.412	76.447	76.483	76.518	76.554	76.589	76.624	76.659	76.694	76.729	76.764	76.799	76.834	76.868	76.903	76.937	76.972	77.006	77.040	77.074	77.108	77.142	77.176	77.210	77.244	77.278	77.312	77.345	77.379	77.412	77.446	77.479	77.512	77.545	77.578	77.611	77.644	77.677	77.710	77.743	77.776	77.809	77.842	77.874	77.907	77.939	77.972	78.004	78.036	78.068	78.100	78.132	78.164	78.196	78.228	78.260	78.291	78.323	78.354	78.385	78.416	78.447	78.478	78.509	78.540	78.570	78.601	78.631	78.661	78.691	78.721	78.751	78.781	78.811	78.841	78.871	78.901	78.931	78.961	78.991	79.021	79.051	79.081	79.111	79.141	79.171	79.201	79.231	79.261	79.291	79.321	79.351	79.381	79.411	79.441	79.471	79.501	79.531	79.561	79.591	79.621	79.651	79.681	79.711	79.741	79.771	79.801	79.831	79.861	79.891	79.921	79.951	79.981	80.011	80.041	80.071	80.101	80.131	80.161	80.191	80.221	80.251	80.281	80.311	80.341	80.371	80.401	80.431	80.461	80.491	80.521	80.551	80.581	80.611	80.641	80.671	80.701	80.731	80.761	80.791	80.821	80.851	80.881	80.911	80.941	80.971	81.001	81.031	81.061	81.091	81.121	81.151	81.181	81.211	81.241	81.271	81.301	81.331	81.361	81.391	81.421	81.451	81.481	81.511	81.541	81.571	81.601	81.631	81.661	81.691	81.721	81.751	81.781	81.811	81.841	81.871	81.901	81.931	81.961	81.991	82.021	82.051	82.081	82.111	82.141	82.171	82.201	82.231	82.261	82.291	82.321	82.351	82.381	82.411	82.441	82.471	82.501	82.531	82.561	82.591	82.621	82.651	82.681	82.711	82.741	82.771	82.801	82.831	82.861	82.891	82.921	82.951	82.981	83.011	83.041	83.071	83.101	83.131	83.161	83.191	83.221	83.251	83.281	83.311	83.341	83.371	83.401	83.431	83.461	83.491	83.521	83.551	83.581	83.611	83.641	83.671	83.701	83.731	83.761	83.791	83.821	83.851	83.881	83.911	83.941	83.971	84.001	84.031	84.061	84.091	84.121	84.151	84.181	84.211	84.241	84.271	84.301	84.331	84.361	84.391	84.421	84.451	84.481	84.511	84.541	84.571	84.601	84.631	84.661	84.691	84.721	84.751	84.781	84.811	84.841	84.871	84.901	84.931	84.961	84.991	85.021	85.051	85.081	85.111	85.141	85.171	85.201	85.231	85.261	85.291	85.321	85.351	85.381	85.411	85.441	85.471	85.501	85.531	85.561	85.591	85.621	85.651	85.681	85.711	85.741	85.771	85.801	85.831	85.861	85.891	85.921	85.951	85.981	86.011	86.041	86.071	86.101	86.131	86.161	86.191	86.221	86.251	86.281	86.311	86.341	86.371	86.401	86.431	86.461	86.491	86.521	86.551	86.581	86.611	86.641	86.671	86.701	86.731	86.761	86.791	86.821	86.851	86.881	86.911	86.941	86.971	87.001	87.031	87.061	87.091	87.121	87.151	87.181	87.211	87.241	87.271	87.301	87.331	87.361	87.391	87.421	87.451	87.481	87.511	87.541	87.571	87.601	87.631	87.661	87.691	87.721	87.751	87.781	87.811	87.841	87.871	87.901	87.931	87.961	87.991	88.021	88.051	88.081	88.111	88.141	88.171	88.201	88.231	88.261	88.291	88.321	88.351	88.381	88.411	88.441	88.471	88.501	88.531	88.561	88.591	88.621	88.651	88.681	88.711	88.741



$\frac{P_0}{\text{FEET}}$	$\frac{P_0}{\text{psi}}$	$\delta$	$U_0$	$T_0$ (°F)	$\theta$	$\sqrt{\theta}$	$\sigma$	$\sqrt{\sigma}$	$\frac{1}{\sqrt{\sigma}}$	$\frac{\sqrt{\theta}}{\sigma}$	$\phi$	$\frac{\delta}{\sqrt{\theta}}$
30000	5.810	19319	5.169	216.66	7519	8671	2582	5082	1.9677	4.465	790	2821
31000	5.782	19326	5.174	216.66	7519	8671	2570	5070	1.9724	4.468	790	2821
32000	5.754	19331	5.179	216.66	7519	8671	2559	5058	1.9772	4.471	790	2821
33000	5.727	19336	5.184	216.66	7519	8671	2548	5046	1.9819	4.474	790	2821
34000	5.699	19341	5.189	216.66	7519	8671	2537	5033	1.9867	4.477	790	2821
35000	5.672	19346	5.194	216.66	7519	8671	2526	5021	1.9915	4.480	790	2821
36000	5.644	19351	5.199	216.66	7519	8671	2515	5009	1.9962	4.483	790	2821
37000	5.617	19356	5.204	216.66	7519	8671	2504	4997	2.0010	4.486	790	2821
38000	5.589	19361	5.209	216.66	7519	8671	2493	4985	2.0057	4.489	790	2821
39000	5.562	19366	5.214	216.66	7519	8671	2482	4973	2.0105	4.492	790	2821
40000	5.534	19371	5.219	216.66	7519	8671	2471	4961	2.0152	4.495	790	2821
41000	5.507	19376	5.224	216.66	7519	8671	2460	4949	2.0199	4.498	790	2821
42000	5.479	19381	5.229	216.66	7519	8671	2449	4937	2.0247	4.501	790	2821
43000	5.452	19386	5.234	216.66	7519	8671	2438	4925	2.0294	4.504	790	2821
44000	5.424	19391	5.239	216.66	7519	8671	2427	4914	2.0342	4.507	790	2821
45000	5.397	19396	5.244	216.66	7519	8671	2416	4902	2.0389	4.510	790	2821
46000	5.369	19401	5.249	216.66	7519	8671	2405	4890	2.0437	4.513	790	2821
47000	5.342	19406	5.254	216.66	7519	8671	2394	4878	2.0484	4.516	790	2821
48000	5.314	19411	5.259	216.66	7519	8671	2383	4867	2.0532	4.519	790	2821
49000	5.287	19416	5.264	216.66	7519	8671	2372	4855	2.0579	4.522	790	2821
50000	5.259	19421	5.269	216.66	7519	8671	2361	4844	2.0627	4.525	790	2821
51000	5.232	19426	5.274	216.66	7519	8671	2350	4832	2.0674	4.528	790	2821
52000	5.204	19431	5.279	216.66	7519	8671	2339	4820	2.0722	4.531	790	2821
53000	5.177	19436	5.284	216.66	7519	8671	2328	4809	2.0769	4.534	790	2821
54000	5.149	19441	5.289	216.66	7519	8671	2317	4797	2.0817	4.537	790	2821
55000	5.122	19446	5.294	216.66	7519	8671	2306	4786	2.0864	4.540	790	2821
56000	5.094	19451	5.299	216.66	7519	8671	2295	4774	2.0912	4.543	790	2821
57000	5.067	19456	5.304	216.66	7519	8671	2284	4763	2.0959	4.546	790	2821
58000	5.039	19461	5.309	216.66	7519	8671	2273	4751	2.1007	4.549	790	2821
59000	5.012	19466	5.314	216.66	7519	8671	2262	4740	2.1054	4.552	790	2821
60000	4.984	19471	5.319	216.66	7519	8671	2251	4729	2.1102	4.555	790	2821
61000	4.957	19476	5.324	216.66	7519	8671	2240	4717	2.1149	4.558	790	2821
62000	4.929	19481	5.329	216.66	7519	8671	2229	4706	2.1197	4.561	790	2821
63000	4.902	19486	5.334	216.66	7519	8671	2218	4695	2.1244	4.564	790	2821
64000	4.874	19491	5.339	216.66	7519	8671	2207	4683	2.1292	4.567	790	2821
65000	4.847	19496	5.344	216.66	7519	8671	2196	4672	2.1339	4.570	790	2821
66000	4.819	19501	5.349	216.66	7519	8671	2185	4661	2.1387	4.573	790	2821
67000	4.792	19506	5.354	216.66	7519	8671	2174	4650	2.1434	4.576	790	2821
68000	4.764	19511	5.359	216.66	7519	8671	2163	4639	2.1482	4.579	790	2821
69000	4.737	19516	5.364	216.66	7519	8671	2152	4627	2.1529	4.582	790	2821
70000	4.709	19521	5.369	216.66	7519	8671	2141	4616	2.1577	4.585	790	2821
71000	4.682	19526	5.374	216.66	7519	8671	2130	4605	2.1624	4.588	790	2821
72000	4.654	19531	5.379	216.66	7519	8671	2119	4594	2.1672	4.591	790	2821
73000	4.627	19536	5.384	216.66	7519	8671	2108	4583	2.1719	4.594	790	2821
74000	4.599	19541	5.389	216.66	7519	8671	2097	4572	2.1767	4.597	790	2821
75000	4.572	19546	5.394	216.66	7519	8671	2086	4561	2.1814	4.600	790	2821
76000	4.544	19551	5.399	216.66	7519	8671	2075	4550	2.1862	4.603	790	2821
77000	4.517	19556	5.404	216.66	7519	8671	2064	4539	2.1909	4.606	790	2821
78000	4.489	19561	5.409	216.66	7519	8671	2053	4528	2.1957	4.609	790	2821
79000	4.462	19566	5.414	216.66	7519	8671	2042	4517	2.2004	4.612	790	2821
80000	4.434	19571	5.419	216.66	7519	8671	2031	4506	2.2052	4.615	790	2821
81000	4.407	19576	5.424	216.66	7519	8671	2020	4495	2.2099	4.618	790	2821
82000	4.379	19581	5.429	216.66	7519	8671	2009	4484	2.2147	4.621	790	2821
83000	4.352	19586	5.434	216.66	7519	8671	1998	4473	2.2194	4.624	790	2821
84000	4.324	19591	5.439	216.66	7519	8671	1987	4462	2.2242	4.627	790	2821
85000	4.297	19596	5.444	216.66	7519	8671	1976	4451	2.2289	4.630	790	2821
86000	4.269	19601	5.449	216.66	7519	8671	1965	4440	2.2337	4.633	790	2821
87000	4.242	19606	5.454	216.66	7519	8671	1954	4429	2.2384	4.636	790	2821
88000	4.214	19611	5.459	216.66	7519	8671	1943	4418	2.2432	4.639	790	2821
89000	4.187	19616	5.464	216.66	7519	8671	1932	4407	2.2479	4.642	790	2821
90000	4.159	19621	5.469	216.66	7519	8671	1921	4396	2.2527	4.645	790	2821

H (FEET)	P <sub>h</sub> (W.G.)	B	b	T <sub>h</sub> (°C)	θ (KNOTS)	g	√g	σ	√σ	1/√σ	√g/σ	1/√g	φ	g/φ√g
41800	4.794	14023	6.231	216.64	573.58	7519	8671	2131	4616	2.1662	3.411	7.197	.790	.2336
41900	4.771	13944	6.231	216.64	573.58	7519	8671	2120	4605	2.1715	3.417	7.232	.790	.2327
42000	4.748	13865	6.301	216.64	573.58	7519	8671	2110	4594	2.1767	3.423	7.264	.790	.2316
42100	4.725	13784	6.371	216.64	573.58	7519	8671	2100	4583	2.1819	3.430	7.296	.790	.2305
42200	4.702	13703	6.441	216.64	573.58	7519	8671	2090	4572	2.1872	3.437	7.327	.790	.2294
42300	4.680	13622	6.511	216.64	573.58	7519	8671	2080	4561	2.1924	3.443	7.357	.790	.2283
42400	4.657	13541	6.581	216.64	573.58	7519	8671	2070	4550	2.1977	3.450	7.388	.790	.2272
42500	4.634	13460	6.651	216.64	573.58	7519	8671	2060	4539	2.2030	3.457	7.418	.790	.2261
42600	4.611	13379	6.721	216.64	573.58	7519	8671	2050	4528	2.2083	3.463	7.449	.790	.2250
42700	4.588	13298	6.791	216.64	573.58	7519	8671	2040	4517	2.2136	3.470	7.479	.790	.2239
42800	4.565	13217	6.861	216.64	573.58	7519	8671	2030	4506	2.2189	3.477	7.510	.790	.2228
42900	4.542	13136	6.931	216.64	573.58	7519	8671	2020	4495	2.2242	3.484	7.540	.790	.2217
43000	4.519	13055	7.001	216.64	573.58	7519	8671	2010	4484	2.2295	3.491	7.571	.790	.2206
43100	4.496	12974	7.071	216.64	573.58	7519	8671	2000	4473	2.2348	3.498	7.601	.790	.2195
43200	4.473	12893	7.141	216.64	573.58	7519	8671	1990	4462	2.2401	3.505	7.632	.790	.2184
43300	4.450	12812	7.211	216.64	573.58	7519	8671	1980	4451	2.2454	3.512	7.662	.790	.2173
43400	4.427	12731	7.281	216.64	573.58	7519	8671	1970	4440	2.2507	3.519	7.693	.790	.2162
43500	4.404	12650	7.351	216.64	573.58	7519	8671	1960	4429	2.2560	3.526	7.723	.790	.2151
43600	4.381	12569	7.421	216.64	573.58	7519	8671	1950	4418	2.2613	3.533	7.754	.790	.2140
43700	4.358	12488	7.491	216.64	573.58	7519	8671	1940	4407	2.2666	3.540	7.784	.790	.2129
43800	4.335	12407	7.561	216.64	573.58	7519	8671	1930	4396	2.2719	3.547	7.815	.790	.2118
43900	4.312	12326	7.631	216.64	573.58	7519	8671	1920	4385	2.2772	3.554	7.845	.790	.2107
44000	4.289	12245	7.701	216.64	573.58	7519	8671	1910	4374	2.2825	3.561	7.876	.790	.2096
44100	4.266	12164	7.771	216.64	573.58	7519	8671	1900	4363	2.2878	3.568	7.906	.790	.2085
44200	4.243	12083	7.841	216.64	573.58	7519	8671	1890	4352	2.2931	3.575	7.937	.790	.2074
44300	4.220	12002	7.911	216.64	573.58	7519	8671	1880	4341	2.2984	3.582	7.967	.790	.2063
44400	4.197	11921	7.981	216.64	573.58	7519	8671	1870	4330	2.3037	3.589	7.998	.790	.2052
44500	4.174	11840	8.051	216.64	573.58	7519	8671	1860	4319	2.3090	3.596	8.028	.790	.2041
44600	4.151	11759	8.121	216.64	573.58	7519	8671	1850	4308	2.3143	3.603	8.059	.790	.2030
44700	4.128	11678	8.191	216.64	573.58	7519	8671	1840	4297	2.3196	3.610	8.089	.790	.2019
44800	4.105	11597	8.261	216.64	573.58	7519	8671	1830	4286	2.3249	3.617	8.120	.790	.2008
44900	4.082	11516	8.331	216.64	573.58	7519	8671	1820	4275	2.3302	3.624	8.150	.790	.1997
45000	4.059	11435	8.401	216.64	573.58	7519	8671	1810	4264	2.3355	3.631	8.181	.790	.1986
45100	4.036	11354	8.471	216.64	573.58	7519	8671	1800	4253	2.3408	3.638	8.211	.790	.1975
45200	4.013	11273	8.541	216.64	573.58	7519	8671	1790	4242	2.3461	3.645	8.242	.790	.1964
45300	3.990	11192	8.611	216.64	573.58	7519	8671	1780	4231	2.3514	3.652	8.272	.790	.1953
45400	3.967	11111	8.681	216.64	573.58	7519	8671	1770	4220	2.3567	3.659	8.303	.790	.1942
45500	3.944	11030	8.751	216.64	573.58	7519	8671	1760	4209	2.3620	3.666	8.333	.790	.1931
45600	3.921	10949	8.821	216.64	573.58	7519	8671	1750	4198	2.3673	3.673	8.364	.790	.1920
45700	3.898	10868	8.891	216.64	573.58	7519	8671	1740	4187	2.3726	3.680	8.394	.790	.1909
45800	3.875	10787	8.961	216.64	573.58	7519	8671	1730	4176	2.3779	3.687	8.425	.790	.1898
45900	3.852	10706	9.031	216.64	573.58	7519	8671	1720	4165	2.3832	3.694	8.455	.790	.1887
46000	3.829	10625	9.101	216.64	573.58	7519	8671	1710	4154	2.3885	3.701	8.486	.790	.1876
46100	3.806	10544	9.171	216.64	573.58	7519	8671	1700	4143	2.3938	3.708	8.516	.790	.1865
46200	3.783	10463	9.241	216.64	573.58	7519	8671	1690	4132	2.3991	3.715	8.547	.790	.1854
46300	3.760	10382	9.311	216.64	573.58	7519	8671	1680	4121	2.4044	3.722	8.577	.790	.1843
46400	3.737	10301	9.381	216.64	573.58	7519	8671	1670	4110	2.4097	3.729	8.608	.790	.1832
46500	3.714	10220	9.451	216.64	573.58	7519	8671	1660	4099	2.4150	3.736	8.638	.790	.1821
46600	3.691	10139	9.521	216.64	573.58	7519	8671	1650	4088	2.4203	3.743	8.669	.790	.1810
46700	3.668	10058	9.591	216.64	573.58	7519	8671	1640	4077	2.4256	3.750	8.699	.790	.1799
46800	3.645	9977	9.661	216.64	573.58	7519	8671	1630	4066	2.4309	3.757	8.730	.790	.1788
46900	3.622	9896	9.731	216.64	573.58	7519	8671	1620	4055	2.4362	3.764	8.760	.790	.1777
47000	3.599	9815	9.801	216.64	573.58	7519	8671	1610	4044	2.4415	3.771	8.791	.790	.1766
47100	3.576	9734	9.871	216.64	573.58	7519	8671	1600	4033	2.4468	3.778	8.821	.790	.1755
47200	3.553	9653	9.941	216.64	573.58	7519	8671	1590	4022	2.4521	3.785	8.852	.790	.1744
47300	3.530	9572	10.011	216.64	573.58	7519	8671	1580	4011	2.4574	3.792	8.882	.790	.1733
47400	3.507	9491	10.081	216.64	573.58	7519	8671	1570	4000	2.4627	3.799	8.913	.790	.1722
47500	3.484	9410	10.151	216.64	573.58	7519	8671	1560	3989	2.4680	3.806	8.943	.790	.1711
47600	3.461	9329	10.221	216.64	573.58	7519	8671	1550	3978	2.4733	3.813	8.974	.790	.1700
47700	3.438	9248	10.291	216.64	573.58	7519	8671	1540	3967	2.4786	3.820	9.004	.790	.1689
47800	3.415	9167	10.361	216.64	573.58	7519	8671	1530	3956	2.4839	3.827	9.035	.790	.1678
47900	3.392	9086	10.431	216.64	573.58	7519	8671	1520	3945	2.4892	3.834	9.065	.790	.1667
48000	3.369	9005	10.501	216.64	573.58	7519	8671	1510	3934	2.4945	3.841	9.096	.790	.1656
48100	3.346	8924	10.571	216.64	573.58	7519	8671	1500	3923	2.4998	3.848	9.126	.790	.1645
48200	3.323	8843	10.641	216.64	573.58	7519	8671	1490	3912	2.5051	3.855	9.157	.790	.1634
48300	3.300	8762	10.711	216.64	573.58	7519	8671	1480	3901	2.5104	3.862	9.187	.790	.1623
48400	3.277	8681	10.781	216.64	573.58	7519	8671	1470	3890	2.5157	3.869	9.218	.790	.1612
48500	3.254	8600	10.851	216.64	573.58	7519	8671	1460	3879	2.5210	3.876	9.248	.790	.1601
48600	3.231	8519	10.921	216.64	573.58	7519	8671	1450	3868	2.5263	3.883	9.279	.790	.1590
48700	3.208	8438	10.991	216.64	573.58	7519	8671	1440	3857	2.5316	3.890	9.309	.790	.1579
48800	3.185	8357	11.061	216.64	573.58	7519	8671	1430	3846	2.5369	3.897	9.340	.790	.1568
48900	3.162	8276	11.131	216.64	573.58	7519	8671	1420	3835	2.5422	3.904	9.370	.790	.1557
49000	3.139	8195	11.201	216.64	573.58	7519	8671	1410	3824	2.5475	3.911	9.401	.790	.1546
49100	3.116	8114	11.271	216.64	573.58	7519	8671	1400	3813	2.5528	3.918	9.431	.790	.1535
49200	3.093	8033	11.341	216.64	573.58	7519	8671	1390	3802	2.5581	3.925	9.462	.790	.1524
49300	3.070	7952	11.411	216.64	573.58	7519	8671	1380	3791	2.5634	3.932	9.492	.790	.1513
49400	3.047	7871	11.481	216.64	573.58	7519	8671	1370	3780	2.5687	3.939	9.523	.790	.1502
49500	3.024	7790	11.551	216.64	573.58	7519	8671	1360	3769	2.5740	3.946	9.553	.790	.1491
49600	3.001	7709	11.621	216.64	573.58	7519	8671	1350	3758	2.5793	3.953	9.584	.790	.1480
49700	2.978	7628	11.691	216.64	573.58	7519	8671	1340	3747	2.5846	3.960	9.614	.790	.1469
49800	2.955	7547	11.761	216.64	573.58	7519	8671							

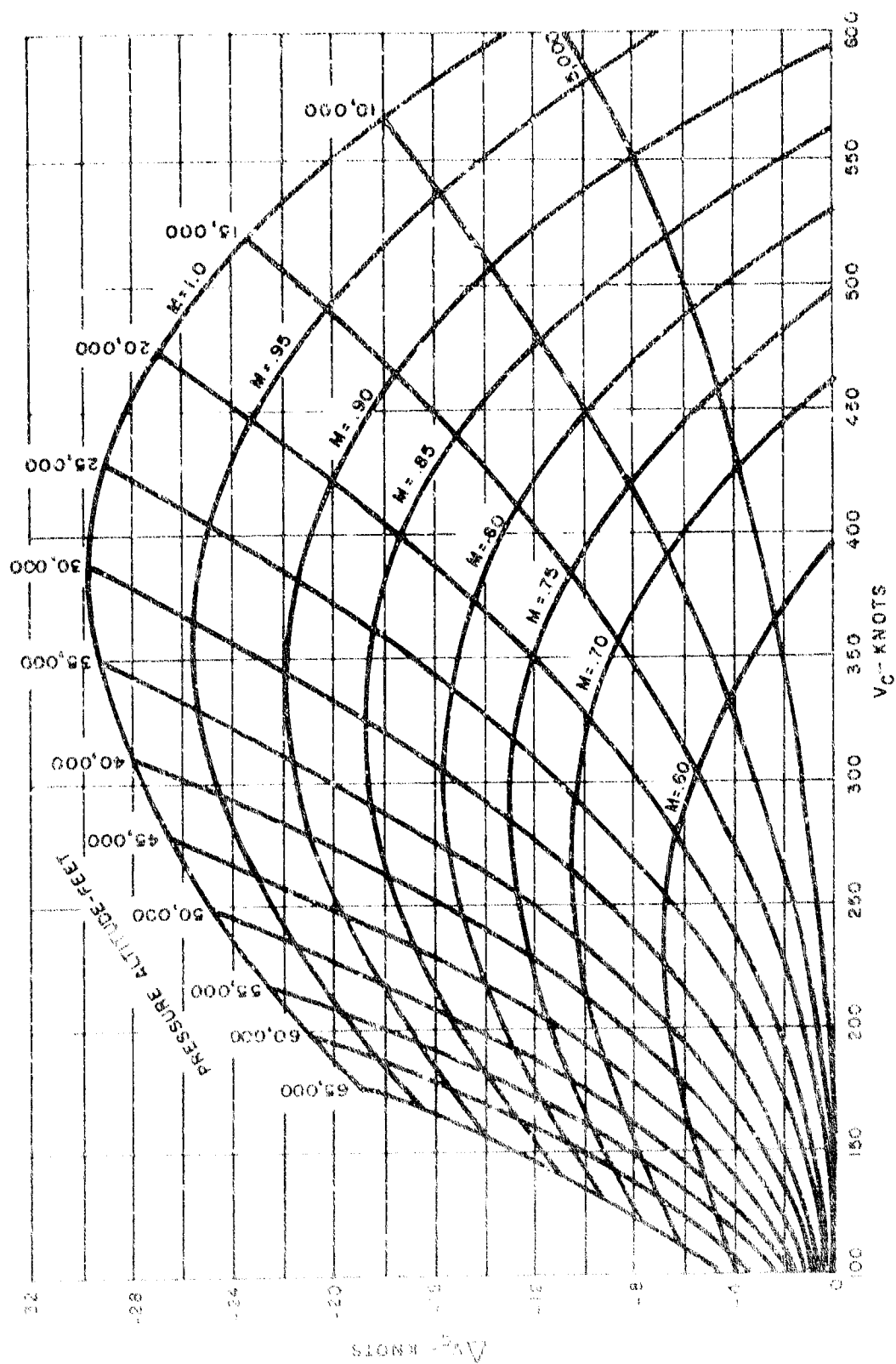
H FEET	M M	b	1/b	T <sub>b</sub> (°K)	n (NOTES)	θ	√θ	σ	√σ	1/√σ	√θ/σ	1/σ√θ	φ	φ√θ
47000	3.555	1322	7.565	216.66	573.58	7519	8671	17584	4193	2.3818	6.558	8.722	790	1929
47100	3.538	13157	7.562	216.66	573.58	7519	8671	17492	4183	2.3905	6.599	8.765	790	1919
47200	3.521	13094	7.559	216.66	573.58	7519	8671	17401	4173	2.3993	6.642	8.807	790	1910
47300	3.504	13031	7.556	216.66	573.58	7519	8671	17310	4163	2.4081	6.684	8.850	790	1902
47400	3.487	12968	7.553	216.66	573.58	7519	8671	17220	4153	2.4169	6.726	8.892	790	1894
47500	3.470	12905	7.550	216.66	573.58	7519	8671	17129	4143	2.4257	6.768	8.935	790	1886
47600	3.453	12842	7.547	216.66	573.58	7519	8671	17039	4133	2.4345	6.810	8.978	790	1878
47700	3.436	12779	7.544	216.66	573.58	7519	8671	16948	4123	2.4433	6.852	9.021	790	1870
47800	3.419	12716	7.541	216.66	573.58	7519	8671	16857	4113	2.4521	6.894	9.063	790	1862
47900	3.402	12653	7.538	216.66	573.58	7519	8671	16766	4103	2.4609	6.936	9.105	790	1854
48000	3.385	12590	7.535	216.66	573.58	7519	8671	16675	4093	2.4697	6.978	9.147	790	1846
48100	3.368	12527	7.532	216.66	573.58	7519	8671	16584	4083	2.4785	7.020	9.189	790	1838
48200	3.351	12464	7.529	216.66	573.58	7519	8671	16493	4073	2.4873	7.062	9.231	790	1830
48300	3.334	12401	7.526	216.66	573.58	7519	8671	16402	4063	2.4961	7.094	9.273	790	1822
48400	3.317	12338	7.523	216.66	573.58	7519	8671	16311	4053	2.5049	7.136	9.315	790	1814
48500	3.300	12275	7.520	216.66	573.58	7519	8671	16220	4043	2.5137	7.178	9.357	790	1806
48600	3.283	12212	7.517	216.66	573.58	7519	8671	16129	4033	2.5225	7.220	9.399	790	1798
48700	3.266	12149	7.514	216.66	573.58	7519	8671	16038	4023	2.5313	7.262	9.441	790	1790
48800	3.249	12086	7.511	216.66	573.58	7519	8671	15947	4013	2.5401	7.304	9.483	790	1782
48900	3.232	12023	7.508	216.66	573.58	7519	8671	15856	4003	2.5489	7.346	9.525	790	1774
49000	3.215	11960	7.505	216.66	573.58	7519	8671	15765	3993	2.5577	7.388	9.567	790	1766
49100	3.198	11897	7.502	216.66	573.58	7519	8671	15674	3983	2.5665	7.430	9.609	790	1758
49200	3.181	11834	7.499	216.66	573.58	7519	8671	15583	3973	2.5753	7.472	9.651	790	1750
49300	3.164	11771	7.496	216.66	573.58	7519	8671	15492	3963	2.5841	7.514	9.693	790	1742
49400	3.147	11708	7.493	216.66	573.58	7519	8671	15401	3953	2.5929	7.556	9.735	790	1734
49500	3.130	11645	7.490	216.66	573.58	7519	8671	15310	3943	2.6017	7.598	9.777	790	1726

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H (FEET)	$P_0$ (in. Hg.)	$\delta$	$V$ (ft./sec.)	$T_a$ (°C.)	$\theta$ (KNOTS)	$\sqrt{\theta}$	$\sigma$	$\sqrt{\sigma}$	$\frac{\sqrt{\theta}}{\sigma}$	$\frac{1}{\sqrt{\theta}}$	$\phi$	$\frac{\delta}{\phi\sqrt{\theta}}$
30000	2.221	07426	13.466	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0894
32100	2.213	07390	13.331	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0890
34200	2.205	07355	13.196	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0886
36300	2.197	07320	13.061	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0882
38400	2.189	07285	12.926	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0878
40500	2.181	07250	12.791	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0874
42600	2.173	07215	12.656	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0870
44700	2.165	07180	12.521	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0866
46800	2.157	07145	12.386	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0862
48900	2.149	07110	12.251	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0858
51000	2.141	07075	12.116	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0854
53100	2.133	07040	11.981	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0850
55200	2.125	07005	11.846	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0846
57300	2.117	06970	11.711	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0842
59400	2.109	06935	11.576	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0838
61500	2.101	06900	11.441	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0834
63600	2.093	06865	11.306	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0830
65700	2.085	06830	11.171	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0826
67800	2.077	06795	11.036	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0822
69900	2.069	06760	10.901	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0818
72000	2.061	06725	10.766	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0814
74100	2.053	06690	10.631	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0810
76200	2.045	06655	10.496	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0806
78300	2.037	06620	10.361	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0802
80400	2.029	06585	10.226	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0798
82500	2.021	06550	10.091	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0794
84600	2.013	06515	9.956	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0790
86700	2.005	06480	9.821	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0786
88800	2.000	06445	9.686	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0782
90900	2.000	06410	9.551	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0778
93000	2.000	06375	9.416	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0774
95100	2.000	06340	9.281	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0770
97200	2.000	06305	9.146	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0766
99300	2.000	06270	9.011	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0762
101400	2.000	06235	8.876	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0758
103500	2.000	06200	8.741	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0754
105600	2.000	06165	8.606	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0750
107700	2.000	06130	8.471	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0746
109800	2.000	06095	8.336	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0742
111900	2.000	06060	8.201	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0738
114000	2.000	06025	8.066	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0734
116100	2.000	05990	7.931	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0730
118200	2.000	05955	7.796	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0726
120300	2.000	05920	7.661	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0722
122400	2.000	05885	7.526	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0718
124500	2.000	05850	7.391	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0714
126600	2.000	05815	7.256	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0710
128700	2.000	05780	7.121	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0706
130800	2.000	05745	6.986	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0702
132900	2.000	05710	6.851	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0698
135000	2.000	05675	6.716	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0694
137100	2.000	05640	6.581	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0690
139200	2.000	05605	6.446	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0686
141300	2.000	05570	6.311	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0682
143400	2.000	05535	6.176	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0678
145500	2.000	05500	6.041	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0674
147600	2.000	05465	5.906	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0670
149700	2.000	05430	5.771	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0666
151800	2.000	05395	5.636	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0662
153900	2.000	05360	5.501	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0658
156000	2.000	05325	5.366	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0654
158100	2.000	05290	5.231	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0650
160200	2.000	05255	5.096	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0646
162300	2.000	05220	4.961	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0642
164400	2.000	05185	4.826	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0638
166500	2.000	05150	4.691	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0634
168600	2.000	05115	4.556	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0630
170700	2.000	05080	4.421	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0626
172800	2.000	05045	4.286	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0622
174900	2.000	05010	4.151	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0618
177000	2.000	04975	4.016	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0614
179100	2.000	04940	3.881	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0610
181200	2.000	04905	3.746	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0606
183300	2.000	04870	3.611	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0602
185400	2.000	04835	3.476	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0598
187500	2.000	04800	3.341	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0594
189600	2.000	04765	3.206	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0590
191700	2.000	04730	3.071	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0586
193800	2.000	04695	2.936	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0582
195900	2.000	04660	2.801	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0578
198000	2.000	04625	2.666	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0574
200100	2.000	04590	2.531	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0570
202200	2.000	04555	2.396	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0566
204300	2.000	04520	2.261	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0562
206400	2.000	04485	2.126	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0558
208500	2.000	04450	1.991	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0554
210600	2.000	04415	1.856	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0550
212700	2.000	04380	1.721	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0546
214800	2.000	04345	1.586	216.66	573.58	7519	.09877	3143	3.1420	11.676	.790	.0542
216900	2.000	04310	1.451	216.66	573.58	7519	.09877	3143	3.142			

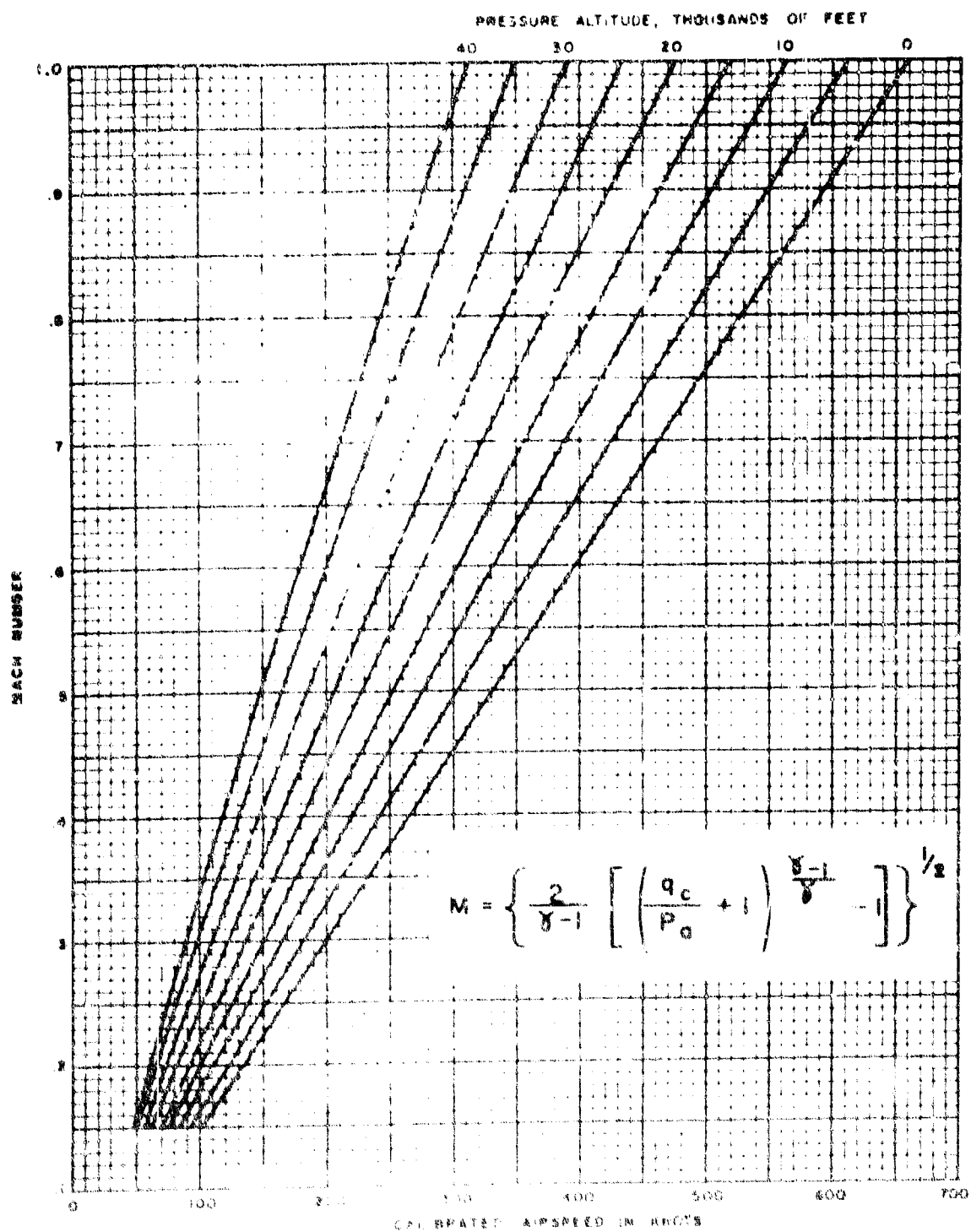
$\frac{1}{\sqrt{g}}$	$\frac{1}{g}$	$\frac{1}{g^2}$	$\frac{1}{g^3}$	$\frac{1}{g^4}$	$\frac{1}{g^5}$	$\frac{1}{g^6}$	$\frac{1}{g^7}$	$\frac{1}{g^8}$	$\frac{1}{g^9}$	$\frac{1}{g^{10}}$	$\frac{1}{g^{11}}$	$\frac{1}{g^{12}}$	$\frac{1}{g^{13}}$	$\frac{1}{g^{14}}$	$\frac{1}{g^{15}}$	$\frac{1}{g^{16}}$	$\frac{1}{g^{17}}$	$\frac{1}{g^{18}}$	$\frac{1}{g^{19}}$	$\frac{1}{g^{20}}$
1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
1.0001	1.0002	1.0004	1.0006	1.0008	1.0010	1.0012	1.0014	1.0016	1.0018	1.0020	1.0022	1.0024	1.0026	1.0028	1.0030	1.0032	1.0034	1.0036	1.0038	1.0040
1.0002	1.0004	1.0008	1.0012	1.0016	1.0020	1.0024	1.0028	1.0032	1.0036	1.0040	1.0044	1.0048	1.0052	1.0056	1.0060	1.0064	1.0068	1.0072	1.0076	1.0080
1.0003	1.0006	1.0012	1.0018	1.0024	1.0030	1.0036	1.0042	1.0048	1.0054	1.0060	1.0066	1.0072	1.0078	1.0084	1.0090	1.0096	1.0102	1.0108	1.0114	1.0120
1.0004	1.0008	1.0016	1.0024	1.0032	1.0040	1.0048	1.0056	1.0064	1.0072	1.0080	1.0088	1.0096	1.0104	1.0112	1.0120	1.0128	1.0136	1.0144	1.0152	1.0160
1.0005	1.0010	1.0020	1.0030	1.0040	1.0050	1.0060	1.0070	1.0080	1.0090	1.0100	1.0110	1.0120	1.0130	1.0140	1.0150	1.0160	1.0170	1.0180	1.0190	1.0200
1.0006	1.0012	1.0024	1.0036	1.0048	1.0060	1.0072	1.0084	1.0096	1.0108	1.0120	1.0132	1.0144	1.0156	1.0168	1.0180	1.0192	1.0204	1.0216	1.0228	1.0240
1.0007	1.0014	1.0028	1.0042	1.0056	1.0070	1.0084	1.0098	1.0112	1.0126	1.0140	1.0154	1.0168	1.0182	1.0196	1.0210	1.0224	1.0238	1.0252	1.0266	1.0280
1.0008	1.0016	1.0032	1.0048	1.0064	1.0080	1.0096	1.0112	1.0128	1.0144	1.0160	1.0176	1.0192	1.0208	1.0224	1.0240	1.0256	1.0272	1.0288	1.0304	1.0320
1.0009	1.0018	1.0036	1.0054	1.0072	1.0090	1.0108	1.0126	1.0144	1.0162	1.0180	1.0198	1.0216	1.0234	1.0252	1.0270	1.0288	1.0306	1.0324	1.0342	1.0360
1.0010	1.0020	1.0040	1.0060	1.0080	1.0100	1.0120	1.0140	1.0160	1.0180	1.0200	1.0220	1.0240	1.0260	1.0280	1.0300	1.0320	1.0340	1.0360	1.0380	1.0400
1.0011	1.0022	1.0044	1.0066	1.0088	1.0110	1.0132	1.0154	1.0176	1.0198	1.0220	1.0242	1.0264	1.0286	1.0308	1.0330	1.0352	1.0374	1.0396	1.0418	1.0440
1.0012	1.0024	1.0048	1.0072	1.0096	1.0120	1.0144	1.0168	1.0192	1.0216	1.0240	1.0264	1.0288	1.0312	1.0336	1.0360	1.0384	1.0408	1.0432	1.0456	1.0480
1.0013	1.0026	1.0052	1.0078	1.0104	1.0130	1.0156	1.0182	1.0208	1.0234	1.0260	1.0286	1.0312	1.0338	1.0364	1.0390	1.0416	1.0442	1.0468	1.0494	1.0520
1.0014	1.0028	1.0056	1.0084	1.0112	1.0140	1.0168	1.0196	1.0224	1.0252	1.0280	1.0308	1.0336	1.0364	1.0392	1.0420	1.0448	1.0476	1.0504	1.0532	1.0560
1.0015	1.0030	1.0060	1.0090	1.0120	1.0150	1.0180	1.0210	1.0240	1.0270	1.0300	1.0330	1.0360	1.0390	1.0420						

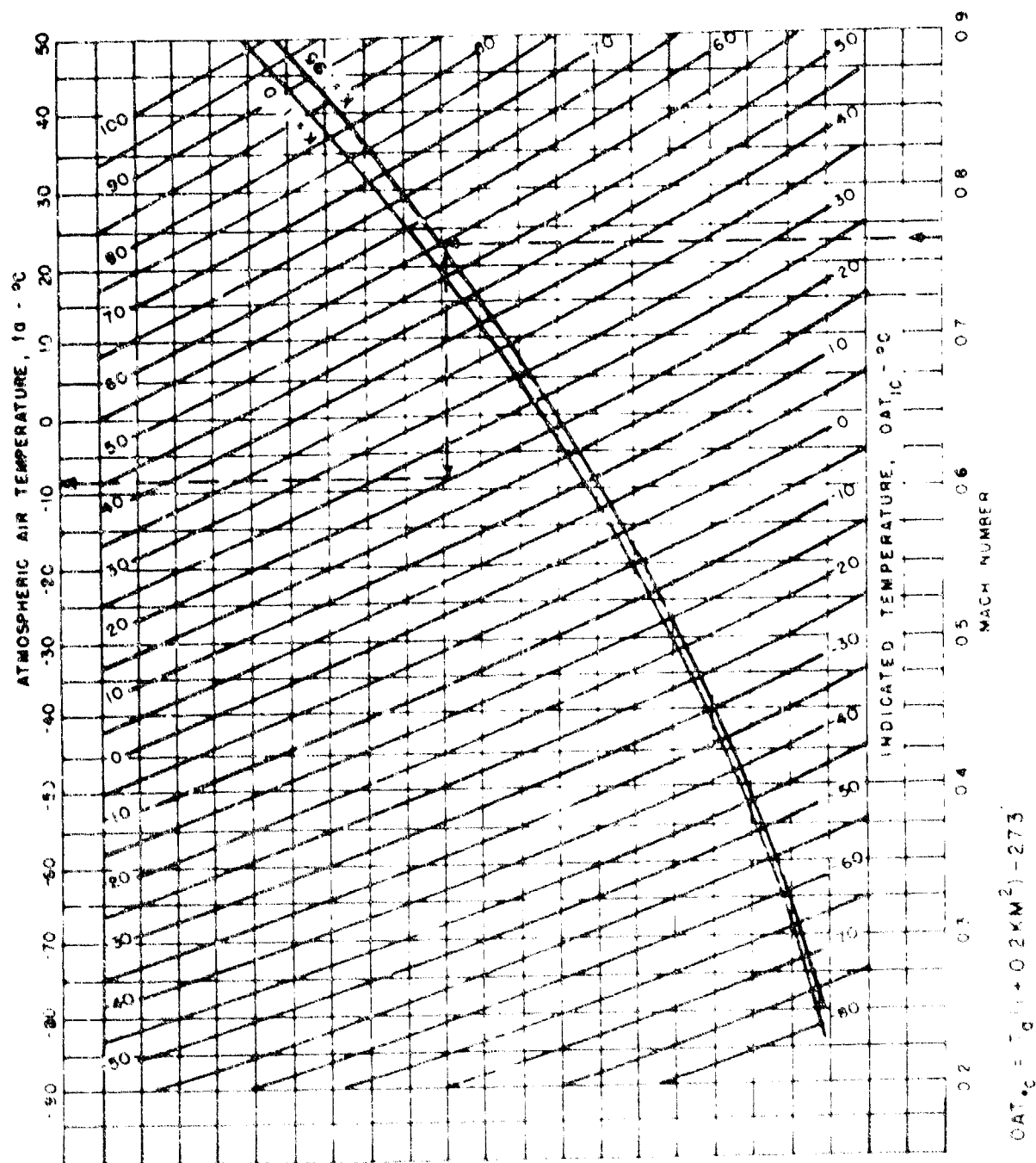


COMPRESSIBILITY CORRECTION  
 $V_e = V_c + \Delta V_c$



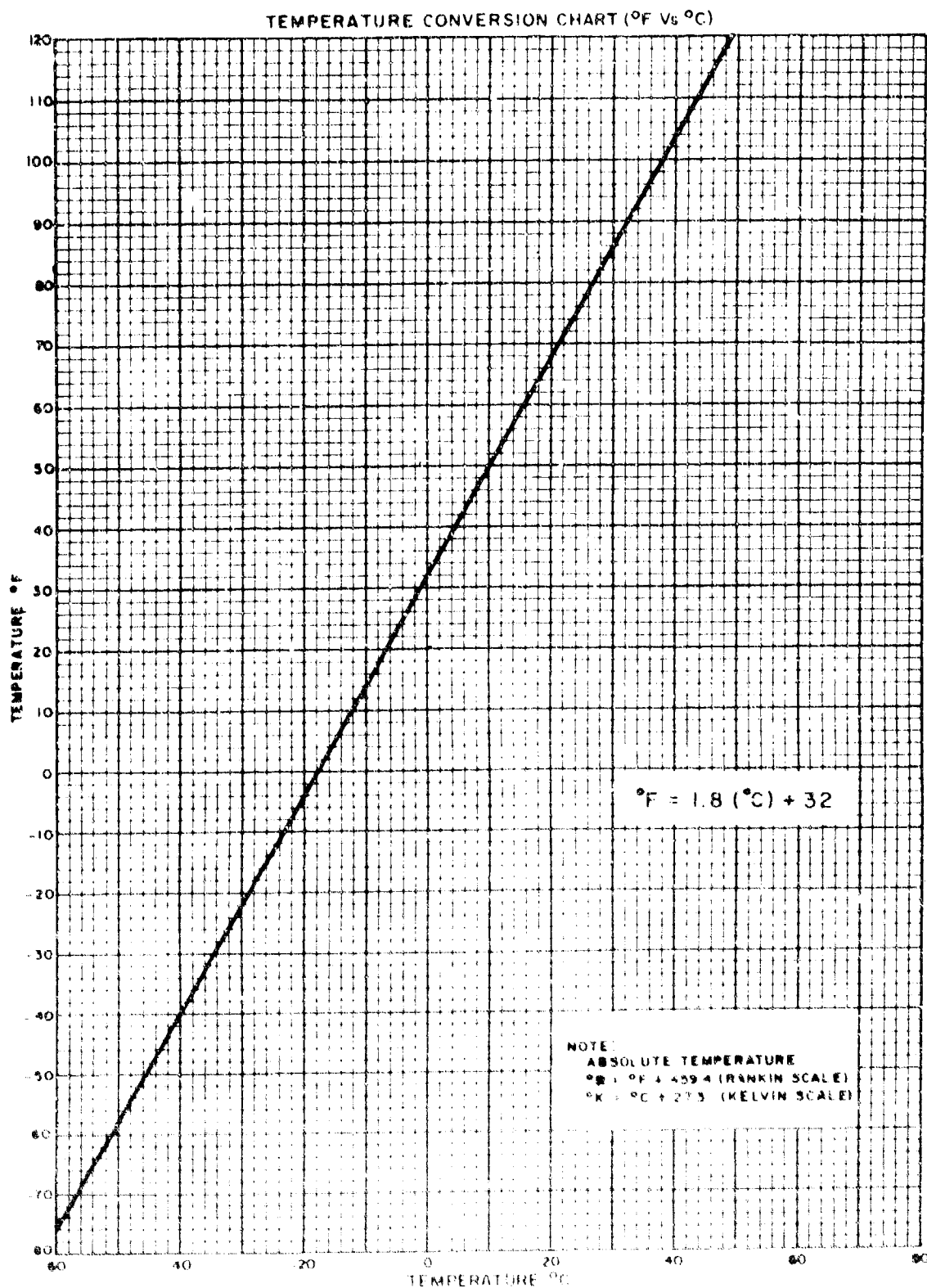
# AIRSPEED - ALTITUDE - MACH NUMBER

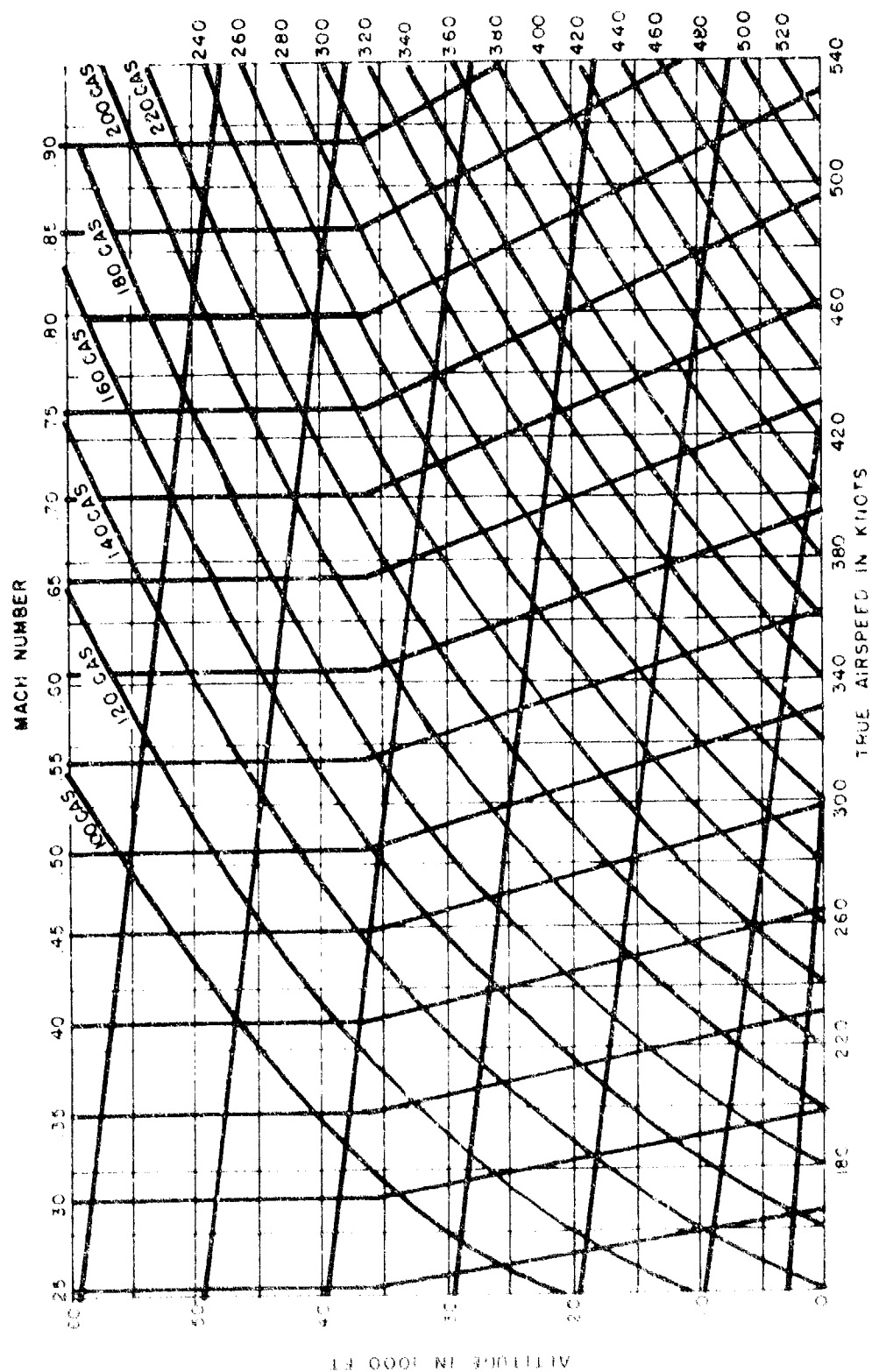




ATMOSPHERIC AIR TEMPERATURE DETERMINATION

# TEMPERATURE CONVERSION CHART (°F Vs °C)





AIRSPEED RELATIONS CALIBRATED AIRSPEED  
(ICAO STANDARD ATMOSPHERE)

# STATIC PRESSURE - PRESSURE ALTITUDE (psf)

PRESSURE ALTITUDE Feet	0	10	20	30	40	50	60	70	80	90
-1000	2195.8	2194.6	2195.4	2196.2	2197.0	2197.8	2198.5	2199.3	2200.1	2200.9
-900	2186.0	2186.7	2187.5	2188.3	2189.1	2189.9	2190.7	2191.5	2192.2	2193.0
-800	2178.1	2178.9	2179.7	2180.5	2181.2	2182.0	2182.8	2183.6	2184.4	2185.2
-700	2170.3	2171.1	2171.9	2172.6	2173.4	2174.2	2175.0	2175.8	2176.6	2177.3
-600	2162.5	2163.3	2164.1	2164.8	2165.6	2166.4	2167.2	2168.0	2168.7	2169.5
-500	2154.7	2155.5	2156.3	2157.1	2157.8	2158.6	2159.4	2160.2	2160.9	2161.7
-400	2147.0	2147.8	2148.5	2149.3	2150.1	2150.9	2151.6	2152.4	2153.2	2154.0
-300	2139.3	2140.0	2140.8	2141.6	2142.3	2143.1	2143.9	2144.7	2145.4	2146.2
-200	2131.6	2132.3	2133.1	2133.9	2134.6	2135.4	2136.2	2136.9	2137.7	2138.5
-100	2123.9	2124.6	2125.4	2126.2	2126.9	2127.7	2128.5	2129.2	2130.0	2130.8
0	2116.2	2117.0	2117.7	2118.5	2119.3	2120.0	2120.8	2121.6	2122.3	2123.1
100	2108.5	2109.3	2110.0	2110.8	2111.6	2112.4	2113.2	2114.0	2114.8	2115.6
200	2100.8	2101.6	2102.4	2103.2	2104.0	2104.8	2105.5	2106.3	2107.1	2107.9
300	2093.1	2093.9	2094.7	2095.5	2096.3	2097.1	2097.9	2098.7	2099.5	2100.3
400	2085.4	2086.2	2087.0	2087.8	2088.6	2089.4	2090.2	2091.0	2091.8	2092.6
500	2077.7	2078.5	2079.3	2080.1	2080.9	2081.7	2082.5	2083.3	2084.1	2084.9
600	2070.0	2070.8	2071.6	2072.4	2073.2	2074.0	2074.8	2075.6	2076.4	2077.2
700	2062.3	2063.1	2063.9	2064.7	2065.5	2066.3	2067.1	2067.9	2068.7	2069.5
800	2054.6	2055.4	2056.2	2057.0	2057.8	2058.6	2059.4	2060.2	2061.0	2061.8
900	2046.9	2047.7	2048.5	2049.3	2050.1	2050.9	2051.7	2052.5	2053.3	2054.1
1000	2039.2	2040.0	2040.8	2041.6	2042.4	2043.2	2044.0	2044.8	2045.6	2046.4
1100	2031.5	2032.3	2033.1	2033.9	2034.7	2035.5	2036.3	2037.1	2037.9	2038.7
1200	2023.8	2024.6	2025.4	2026.2	2027.0	2027.8	2028.6	2029.4	2030.2	2031.0
1300	2016.1	2016.9	2017.7	2018.5	2019.3	2020.1	2020.9	2021.7	2022.5	2023.3
1400	2008.4	2009.2	2010.0	2010.8	2011.6	2012.4	2013.2	2014.0	2014.8	2015.6

# STATIC PRESSURE - PRESSURE ALTITUDE (psf)

PRESSURE ALTITUDE ft., Feet	00	10	20	30	40	50	60	70	80	90
1500	2004.0	2003.5	2002.5	2001.8	2001.1	2000.3	1999.6	1998.9	1998.1	1997.4
1600	1996.7	1996.0	1995.2	1994.5	1993.6	1993.0	1992.5	1991.6	1990.9	1990.1
1700	1989.4	1988.7	1987.9	1987.2	1986.5	1985.8	1985.0	1984.3	1983.6	1982.9
1800	1982.1	1981.4	1980.7	1980.0	1979.2	1978.5	1977.8	1977.1	1976.3	1975.6
1900	1974.9	1974.2	1973.5	1972.7	1972.0	1971.3	1970.6	1969.8	1969.1	1968.4
2000	1967.7	1967.0	1966.2	1965.5	1964.8	1964.1	1963.4	1962.6	1961.9	1961.2
2100	1960.5	1959.8	1959.0	1958.3	1957.6	1956.9	1956.2	1955.5	1954.7	1954.0
2200	1953.3	1952.6	1951.9	1951.1	1950.4	1949.7	1949.0	1948.3	1947.6	1946.9
2300	1946.1	1945.4	1944.7	1944.0	1943.3	1942.6	1941.9	1941.1	1940.4	1939.7
2400	1939.0	1938.3	1937.6	1936.9	1936.2	1935.4	1934.7	1934.0	1933.3	1932.6
2500	1931.9	1931.2	1930.5	1929.8	1929.1	1928.3	1927.6	1926.9	1926.2	1925.5
2600	1924.8	1924.1	1923.4	1922.7	1922.0	1921.3	1920.6	1919.8	1919.1	1918.4
2700	1917.7	1917.0	1916.3	1915.6	1914.9	1914.2	1913.5	1912.8	1912.1	1911.4
2800	1910.7	1910.0	1909.3	1908.6	1907.9	1907.2	1906.5	1905.8	1905.1	1904.3
2900	1903.6	1902.9	1902.2	1901.5	1900.8	1900.1	1899.4	1898.7	1898.0	1897.3
3000	1896.6	1895.9	1895.2	1894.5	1893.8	1893.1	1892.4	1891.7	1891.0	1890.3
3100	1889.5	1889.0	1888.3	1887.6	1886.9	1886.2	1885.5	1884.8	1884.1	1883.4
3200	1882.5	1881.9	1881.3	1880.6	1879.9	1879.2	1878.5	1877.8	1877.1	1876.4
3300	1875.5	1875.0	1874.4	1873.7	1873.0	1872.3	1871.6	1870.9	1870.2	1869.5
3400	1868.8	1868.1	1867.4	1866.7	1866.0	1865.4	1864.7	1864.0	1863.3	1862.6
3500	1861.9	1861.2	1860.5	1859.8	1859.2	1858.5	1857.8	1857.1	1856.4	1855.7
3600	1855.0	1854.3	1853.6	1853.0	1852.3	1851.6	1850.9	1850.2	1849.5	1848.8
3700	1848.2	1847.5	1846.8	1846.1	1845.4	1844.7	1844.1	1843.4	1842.7	1842.0
3800	1841.3	1840.6	1840.0	1839.3	1838.6	1837.9	1837.2	1836.5	1835.9	1835.2
3900	1834.7	1834.0	1833.1	1832.5	1831.8	1831.1	1830.4	1829.7	1829.1	1828.4

# IMPACT PRESSURE ( $q_c$ ) - CALIBRATED AIRSPEED

CALIBRATED  
AIRSPEED  
 $V_C$ , Knots

	0	1	2	3	4	5	6	7	8	9
50	8.42	8.82	9.17	9.53	9.89	10.26	10.64	11.02	11.41	11.80
60	12.21	12.62	13.04	13.47	13.90	14.34	14.78	15.24	15.70	16.16
70	16.64	17.12	17.60	18.10	18.60	19.10	19.62	20.14	20.67	21.20
80	21.75	22.30	22.85	23.42	23.98	24.56	25.15	25.74	26.33	26.94
90	27.55	28.17	28.79	29.43	30.07	30.71	31.37	32.03	32.69	33.37
100	34.05	34.74	35.43	36.14	36.85	37.56	38.29	39.02	39.75	40.50
110	41.25	42.01	42.77	43.55	44.33	45.11	45.91	46.71	47.52	48.33
120	49.15	49.98	50.82	51.66	52.52	53.37	54.24	55.11	55.99	56.88
130	57.77	58.67	59.58	60.50	61.42	62.35	63.28	64.23	65.18	66.14
140	67.10	68.08	69.06	70.04	71.04	72.04	73.05	74.67	75.09	76.12
150	77.16	78.21	79.26	80.32	81.39	82.46	83.54	84.63	85.73	86.83
160	87.95	89.07	90.19	91.33	92.47	93.61	94.77	95.93	97.11	98.28
170	99.47	100.66	101.86	103.07	104.29	105.51	106.74	107.98	109.22	110.49
180	111.74	113.01	114.28	115.57	116.86	118.16	119.46	120.77	122.10	123.42
190	124.76	126.11	127.46	128.82	130.18	131.56	132.94	134.33	135.73	137.13
200	136.55	137.97	141.40	142.83	144.28	145.73	147.19	148.65	150.13	151.61
210	153.10	154.60	156.11	157.62	159.14	160.67	162.21	163.76	165.31	166.87
220	168.44	170.02	171.61	173.20	174.80	176.41	178.03	179.65	181.29	182.93
230	184.58	186.23	187.90	189.57	191.25	192.94	194.64	196.35	198.06	199.78
240	201.51	203.25	205.00	206.75	208.51	210.28	212.06	213.85	215.65	217.45
250	219.62	221.08	222.91	224.75	226.59	228.45	230.31	232.18	234.06	235.95
260	237.84	239.75	241.66	243.58	245.51	247.44	249.39	251.34	253.31	255.28
270	257.26	259.25	261.25	263.25	265.27	267.29	269.32	271.36	273.41	275.47
280	277.53	279.61	281.69	283.78	285.88	287.99	290.11	292.24	294.37	296.52
290	298.67	300.85	303.01	305.19	307.37	309.57	311.78	313.99	316.22	318.45

# IMPACT PRESSURE ( $q_c$ ) - CALIBRATED AIRSPEED

CALIBRATED  
AIRSPEED  
V<sub>a</sub>, knots

	0	1	2	3	4	5	6	7	8	9
500	320.69	322.94	325.20	327.47	329.75	332.04	334.34	336.64	338.96	341.28
510	343.61	345.94	348.30	350.66	353.03	355.41	357.80	360.20	362.60	365.02
520	367.44	369.88	372.32	374.76	377.23	379.71	382.19	384.68	387.18	389.68
530	392.20	394.73	397.27	399.81	402.37	404.94	407.51	410.10	412.69	415.29
540	417.91	420.53	423.17	425.81	428.46	431.12	433.79	436.48	439.17	441.87
550	444.58	447.30	450.03	452.77	455.52	458.28	461.05	463.83	466.62	469.42
560	472.23	475.05	477.88	480.72	483.57	486.43	489.30	492.18	495.07	497.97
570	500.88	503.80	506.73	509.67	512.63	515.59	518.56	521.54	524.54	527.54
580	530.55	533.58	536.61	539.66	542.71	545.78	548.85	551.94	555.04	558.15
590	561.26	564.39	567.53	570.68	573.84	577.85	580.20	583.39	586.60	589.81
600	593.04	596.27	599.52	602.77	606.05	609.33	612.62	615.92	619.24	622.56
610	625.80	629.24	632.60	635.97	639.35	642.74	646.14	649.55	652.98	656.41
620	659.86	663.32	666.79	670.27	673.76	677.26	680.78	684.30	687.84	691.39
630	694.95	698.51	702.10	705.70	709.30	712.92	716.55	720.20	723.85	727.52
640	731.19	734.88	738.58	742.29	746.02	749.75	753.50	757.26	761.03	764.82
650	768.64	772.42	776.24	780.07	783.92	787.77	791.64	795.52	799.41	803.32
660	807.24	811.17	815.11	819.06	823.03	827.01	831.00	835.00	839.02	843.05
670	847.09	851.14	855.21	859.29	863.38	867.48	871.60	875.73	879.87	884.03
680	888.20	892.38	896.57	900.78	905.00	909.23	913.48	917.74	922.01	926.29
690	930.59	934.90	939.23	943.56	947.91	952.28	956.66	961.05	965.45	969.87
700	974.30	978.74	983.20	987.67	992.15	996.65	1001.16	1005.69	1010.23	1014.78
710	1019.35	1023.93	1028.52	1033.13	1037.75	1042.38	1047.03	1051.69	1056.37	1061.06
720	1065.77	1070.49	1075.22	1079.97	1084.73	1089.50	1094.29	1099.10	1103.91	1108.75
730	1113.53	1118.44	1123.35	1128.22	1133.12	1138.04	1142.98	1147.92	1152.89	1157.87
740	1162.86	1167.86	1172.88	1177.92	1182.97	1188.04	1193.12	1198.21	1203.32	1208.45



# IMPACT PRESSURE ( $q_c$ ) - CALIBRATED AIRSPEED

CALIBRATED AIRSPEED $V_C$ , Knots	0	1	2	3	4	5	6	7	8	9
550	1213.59	1218.75	1223.92	1229.10	1234.30	1239.52	1244.75	1250.00	1255.26	1260.54
560	1265.83	1271.14	1276.46	1281.80	1287.15	1292.52	1297.91	1303.31	1308.73	1314.16
570	1319.61	1325.07	1330.55	1336.05	1341.56	1347.08	1352.63	1358.19	1363.76	1369.35
580	1374.96	1380.58	1386.22	1391.88	1397.55	1403.24	1408.94	1414.67	1420.40	1426.16
590	1431.93	1437.71	1443.52	1449.34	1455.17	1461.03	1466.90	1472.78	1478.69	1484.61
600	1490.55	1496.50	1502.47	1508.47	1514.47	1520.49	1526.53	1532.58	1538.66	1544.75
610	1550.86	1556.98	1563.13	1569.29	1575.46	1581.66	1587.87	1594.10	1600.35	1606.62
620	1612.90	1619.20	1625.52	1631.86	1638.21	1644.58	1650.97	1657.38	1663.81	1670.25
630	1676.71	1683.19	1689.69	1696.21	1702.75	1709.30	1715.87	1722.46	1729.07	1735.70
640	1742.35	1749.01	1755.69	1762.40	1769.12	1775.86	1782.61	1789.39	1796.19	1803.00
650	1809.84	1816.69	1823.56	1830.45	1837.36	1844.29	1851.24	1858.21	1865.20	1872.21
660	1879.23	1886.28	1893.35	1900.43	1907.54	1914.66	1921.80	1928.97	1936.15	1943.35
670	1950.57	1957.81	1965.07	1972.35	1979.64	1986.96	1994.29	2001.65	2009.02	2016.41
680	2023.82	2031.24	2038.69	2046.15	2053.64	2061.14	2068.66	2076.20	2083.75	2091.33
690	2098.92	2106.53	2114.16	2121.81	2129.47	2137.15	2144.85	2152.57	2160.31	2168.06
700	2175.85	2183.62	2191.43	2199.25	2207.09	2214.95	2222.83	2230.73	2238.64	2246.57
710	2254.51	2262.48	2270.46	2278.45	2286.47	2294.50	2302.55	2310.62	2318.70	2326.80
720	2334.92	2343.06	2351.21	2359.38	2367.56	2375.76	2383.98	2392.22	2400.47	2408.74
730	2417.03	2425.33	2433.65	2441.98	2450.33	2458.70	2467.09	2475.49	2483.91	2492.34
740	2500.79	2509.26	2517.74	2526.24	2534.75	2543.29	2551.83	2560.40	2568.98	2577.57