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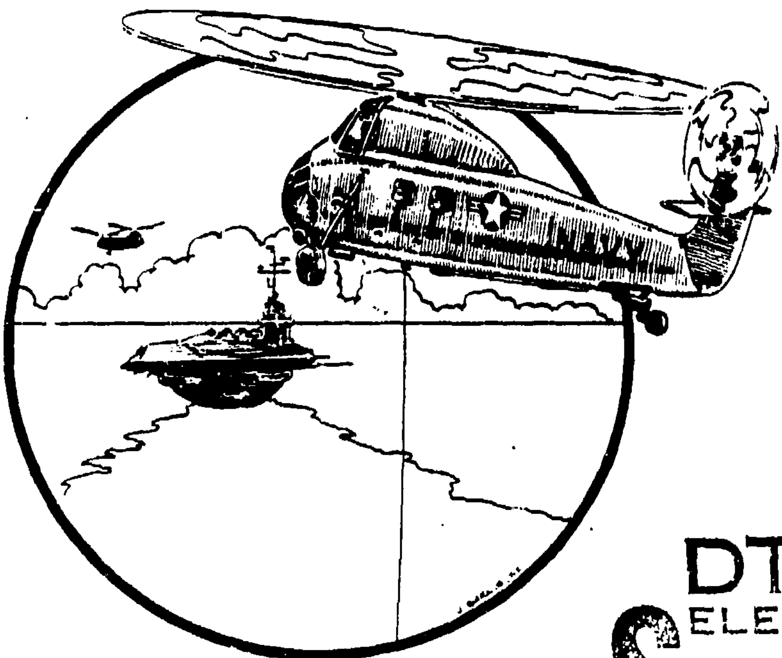
# FLIGHT TEST



# ROTARY WING MANUAL







## Naval Air Test Center
## Patuxent River, Md.

NAVAL AIR TEST CENTER

FLIGHT TEST DIVISION

# ROTARY WING MANUAL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June 1959

FLIGHT TEST DIVISION

ROTARY WING MANUAL

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## FOREWORD

The Rotary Wing Manual, Part I Organization, Part II Instrumentation, Part III Performance and Part IV Flying Qualities, is a compilation of test methods and procedures in use by the Flight Test Division, Naval Air Test Center, Patuxent River, Maryland when conducting tests in accordance with Board of Inspection and Survey Test Directives and/or Bureau of Aeronautics directives.

The general purpose of this manual is to provide adequate reference and text for incoming personnel, both military and civilian, employed by the Flight Test Division and may provide desirable information to other activities.

The test methods outlined herein are essentially a modification of methods presented by the Flight Research Division of the Langley Memorial Aeronautical Laboratory of the NASA.

The purpose of the Rotary Wing Branch of the Flight Test Division is to investigate and determine flight characteristics of helicopters procured by the Navy and to evaluate such equipment and developments which might enhance the value of the helicopters for Naval use. Specifically, the Branch determines whether or not the Navy is getting what it has paid for by checking contract performance guarantees and flight handling characteristics, conducting Navy Preliminary Evaluations to evaluate the Navy aircraft in the early stages of development in order that major deficiencies can be corrected, witnessing contractor's demonstrations to check design criteria, etc.

The manual does not attempt to present helicopter theory or develop the equations contained herein as this is covered quite adequately by many text books on the subject.

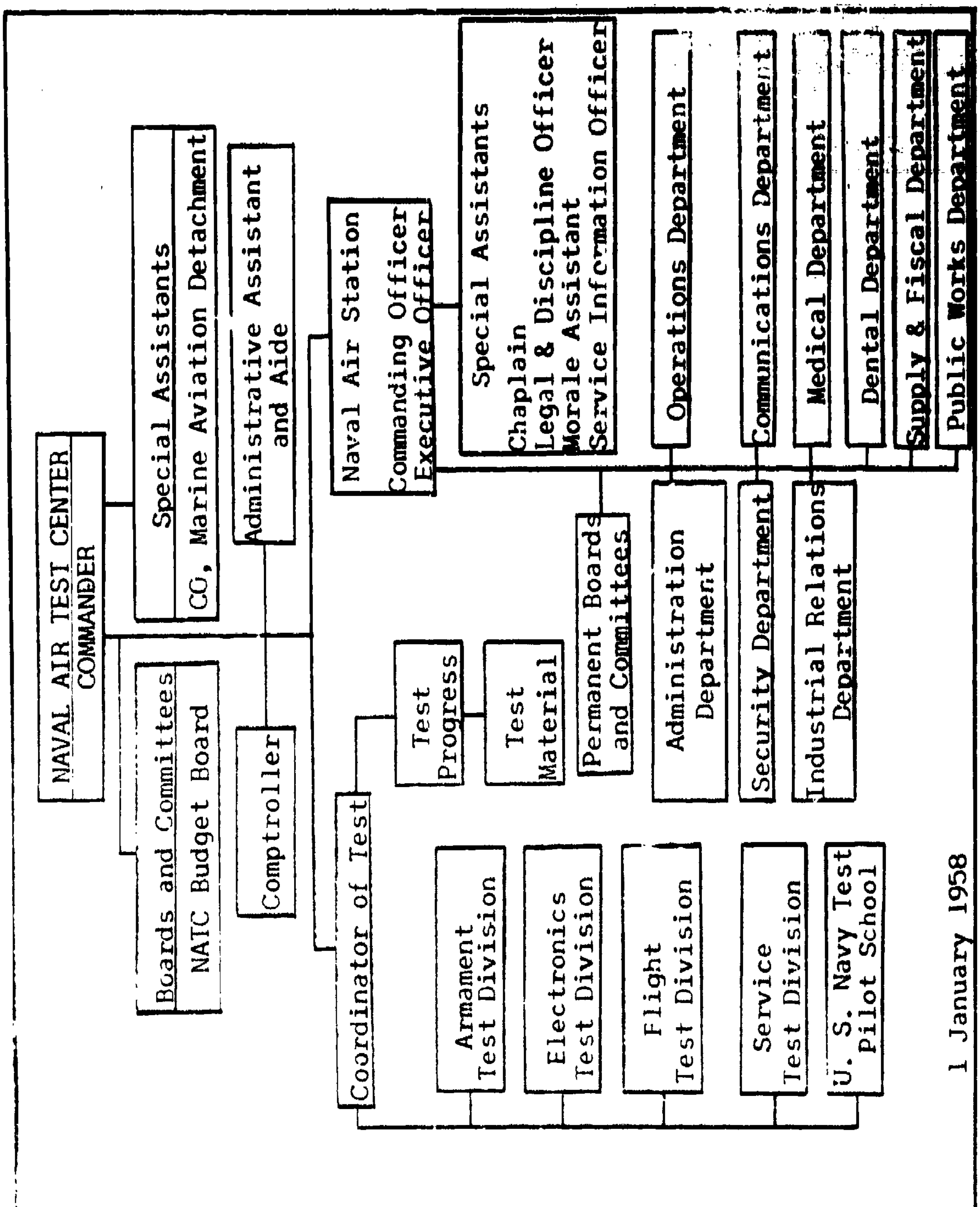
Although to some extent flight testing techniques and performance data analysis for helicopters require further refinement and development, this manual sets forth in writing the current testing techniques and data reduction procedures employed at NATC.

The performance data reduction methods include only piston type power plants. This activity to date has not performed to any great extent tests on turbine powered helicopters. Since future model helicopters will incorporate turbine engines like the T53, T58, etc., data reduction methods and flight techniques may have to be modified. However, the basic performance parameters will remain unchanged since they are based on rotor horsepower. That portion of the data reduction methods requiring modification as a result of turbine powerplant will be amended in the future as the need arises.

Although extreme care was taken in the formulation of this manual there is a possibility that errors are present. It would be appreciated that all errors discovered be forwarded to the following address:

Commander, Naval Air Test Center
Attn: Flight Test Division
Rotary Wing Branch
Patuxent River, Maryland

## 1.1.1 NATC ORGANIZATION



The Naval Air Test Center came into being when it was commissioned 16 June 1945 by the Secretary of the Navy. Prior to this time the various test sections were in existence but in other locations. Flight Test, the oldest test section, was established 1 Jan 1927 as a separate department of the Naval Air Station, Anacostia. Electronics Test was established as Aircraft Radio Test Section on 26 Jan 1929 as a subdivision of Flight Test at Anacostia. Armament Test was established as the Aircraft Armament Unit at Norfolk, Va. on 11 June 1941.

The limited facilities of Anacostia were soon outgrown. A new site was needed for the testing of aircraft and aircraft equipment. It was necessary that this site provide needed air space free from other air traffic, water facilities for amphibious aircraft tests and room for gunnery ranges and other armament tests. A board was established by BuAer to study available sites and make recommendations concerning the location of a proposed Navy Flight Test Center. The board reported that Cedar Point, Md. was the best choice of all sites investigated.

The Naval Air Station, Patuxent River was commissioned early in 1943 on the strength of the above recommendation. Flight Test moved to NAS Patuxent River on 29 June 1943; Radio Test Section (E.T.) completed its move to Patuxent River on 15 July 1943; Aircraft Armament Unit (A.T.) moved to Patuxent River on 15 July 1943; Service Test was established at Patuxent River 8 July 1944 by the Chief of Naval Operations.

The test units at Patuxent functioned as departments of the Naval Air Station before NATC was commissioned. The administrative organ of the test activities was headed by the Office of Director of Tests. On 16 June 1945 the Secretary of the Navy established NATC, as an activity of the Potomac River Naval Command and under the technical control of BuAer and responsible for the functions of the Aviation Test Activities formerly under the Naval Air Station.

The mission of the Naval Air Test Center is to conduct tests and evaluate aircraft and their components. In order to accomplish this mission the following specific tasks are assigned:

a. Evaluate aircraft weapons systems, components and equipment for service use in conformance to BuAer and BIS directives.

b. Perform research and development on aircraft and their components under service environmental conditions as necessary to recommend design changes required by test and evaluation results.

c. Conduct test pilot training.

d. Provide flight support of special projects for industrial and military agencies as assigned by the BuAer.

e. Provide technical advice and assistance to the Board of Inspection and Survey.

f. Conduct design development and test engineering of air navigational electronics traffic control, communications and identification systems under the cognizance of the Bureau of Ships.

g. Conduct Fleet Indoctrination Program for fleet personnel on new aircraft being introduced to the fleet.

Activities and elements of the NATC are:

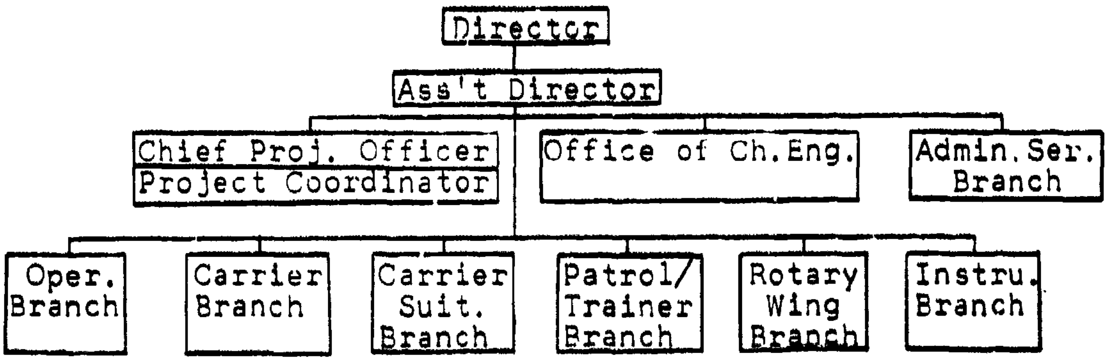
a. Naval Air Station, Patuxent River, Md.

b. Navy All Weather Testing Program, Climatic Hangar Detachment, Eglin Air Force Base, Fla.

c. Marine Aviation Detachment (Patuxent River)

d. NATC Test Divisions

### 1.1.2 FLIGHT TEST ORGANIZATION



The primary function of the Flight Test Division is to flight test and evaluate experimental and new production Naval aircraft through a scientific investigation of the characteristics exhibited on the ground, water or in-flight, making recommendations to correct deficiencies, improve operation and initiate further development. Projects are assigned by the Commander, Naval Air Test Center to provide:

a. Assistance to the Board of Inspection and Survey in the conduct of Service Acceptance Trials and Production Inspection Trials of experimental and new production Naval aircraft. Evaluation of the service suitability of the aircraft to determine whether or not specifications and contract guarantees have been met in the following:

1. Stability and control characteristics (aerodynamic and hydrodynamic) and flying qualities.

2. Aircraft and engine performance characteristics.

3. Carrier suitability characteristics.

b. Evaluation of aircraft and associated equipment for the Bureau of Aeronautics to:

1. Determine, through Navy Preliminary Evaluations of new aircraft models, the degree to which the aircraft will meet the operational requirement and/or design characteristics under which it was designed, to evaluate changes incorporated to correct deficiencies, and to determine when the aircraft is suitable for Board of Inspection and Survey Trials.

2. Determine, through contractor demonstrations of new aircraft models, the structural and aerodynamic adequacy of the aircraft for operation within its design and/or limit strength or maximum safe flight envelope. The Division witnesses and represents the authority (Commander, Naval Air Test Center) conducting the demonstrations.

3. Obtain information of significance to the advancement of aircraft design.

4. Obtain information for the use of operating forces.

Complementary functions include the following:

a. Conduct research in, develop and evaluate new techniques and concepts of operation of carrier-based aircraft to promote increased safety of launching and recovery.

b. Continuously develop new methods and procedures which reflect the current "state of the art" for evaluating:

1. Aircraft from an aerodynamic, hydrodynamic, aircraft and engine performance, and carrier suitability standpoint.

2. New and existing catapult and arresting equipment.

c. Continuously develop instruments and systems of instrumentation which will apply the latest developments in the aviation field for isolating, measuring and recording data in varied fields of aeronautical research and evaluation such as aircraft propulsion systems, aerodynamics in the subsonic, transonic and supersonic speed ranges, hydrodynamics, and take-off and landing loads.

d. Develop and review design specification criteria for Bureau of Aeronautics promulgation.

e. Act as a continuing consultant and advisor to the BuAer on aeronautical matters. Provide consultant services to other government agencies and aircraft and equipment contractors during the planning and development of new aircraft models and catapult and arresting equipment.

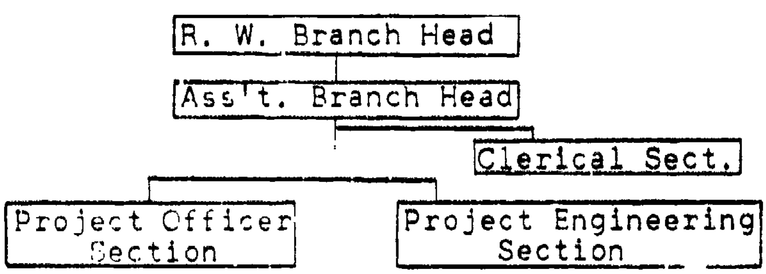
f. Represent the Naval Air Test Center at aircraft mock-up meetings, meetings of NASA sub-committees and during the evaluation of operational flight of Naval aircraft.

g. Maintain a BuAer pool of aircraft test instruments.

h. Provide test instrumentation services for all elements of the NATC.

i. Operate and maintain all catapult and arresting equipment at the NATC.

1.1.3 ROTARY WING BRANCH ORGANIZATION



The Rotary Wing Branch performs the following:

a. Conducts evaluations of the service suitability of experimental and production rotary wing aircraft to determine whether or not specifications and contract guarantees for stability and control (Flying Qualities, Hydrodynamics) and aircraft and engine performance have been met.

b. Conducts Navy Preliminary Evaluations of rotary wing aircraft to determine the degree to which such aircraft meet the operational requirements and/or design characteristics under which they were designed, to evaluate changes incorporated to correct deficiencies, and to determine when the aircraft are suitable for BIS trials.

c. Acts as the authority conducting contractor's demonstrations of rotary wing aircraft to determine the structural and aerodynamic adequacy of the aircraft for operation within their design and/or limit strength or maximum safe flight envelope.

d. Conducts tests of rotary wing aircraft to obtain information of significance to the advancement of aircraft design and to obtain information for the use of operating forces.

e. Develops and reviews design specification criteria for BuAer promulgation.

f. Represents the NATC at mock-up meetings for rotary wing aircraft.

g. Conducts Performance and Flying Quality Testing for the Army.

### 1.1.4 AIRCRAFT REQUIREMENTS AND PROCUREMENT

The procurement of a new helicopter is usually brought about by the requirements of the operating forces. These requirements are often brought to the attention of the Chief of Naval Operations by first hand information offered or sent in by the helicopter squadrons. When it becomes obvious that a new aircraft is needed to fill a mission or increase the operating efficiency of the operating forces, CNO directs BuAer to draw up specifications for such an aircraft.

BuAer, after thorough study of the requirements of the mission or missions for which the aircraft is to be designed, draws up a specification for the aircraft. These specifications are sent to the various contractors for study and design competition. The contractors who wish to enter the competition design an aircraft following the specifications and submit these designs to BuAer for study. A thorough study of all designs is conducted.

As a result of this study, BuAer awards a contract for a test article or maybe several articles. Occasionally a contractor will design and build an aircraft without a government contract in the hopes of later selling it as an "off the shelf aircraft" to the government, but this seldom happens - most all of the aircraft are built with either Navy, Army or Air Force money. In some cases aircraft have been built with company money for a civilian market and then later modified for service use - this is a rare case also.

1.1.5 BOARDS - MOCK-UP, COCKPIT, LIGHTING, etc.

After the contract has been awarded a mock-up of the aircraft is usually built. Representatives of BuAer, Fleet and occasionally representatives of NATC are invited to a mock-up board conference held at the contractor's plant. The purpose of this board is to uncover any deficiencies as to arrangement of components, shape or size of fuselage, etc. so that these shortcomings may be corrected while the aircraft is still in the planning stage. Other boards are sometimes held, such as cockpit, lighting etc. These are usually held after a test article is built.

1.1.6 STATIC TESTS

As the aircraft is being assembled, the various major components undergo static tests. Actual flight conditions are simulated as near as possible for the component concerned and each part is checked for stress, strain and fatigue. MIL-S-8698 covers these tests.

1.1.7 CONTRACTORS PRELIMINARY FLIGHT TESTS (BUILD-UP TESTS MIL-D-8708 (AER))

These flights cover the initial flights of the first flight article. These flights determine whether or not the aircraft is aerodynamically, structurally, and functionally safe for the tests to be performed by Navy Pilots during th Navy Preliminary Evaluation. The tests performed during th phase establish the flight envelope to be utilized during t NPE. The early demonstrations also enable the Navy to obta arly basic information regarding the military potential of new aircraft. Certain demonstration tests when properly performed and documented, eliminate the need for related trials by the Board of Inspection and Survey and thus, shorten periods otherwise required for trials. The Preliminary flights are conducted at the contractor's plant and are usually witnessed by the cognizant BAR.

Although Military Specification MIL-D-8708(Aer) applies specifically to airplanes the general provisions (location of tests, aircraft configuration for demo tests, test authorities, qualification and approval of contractor's pilots, aircraft operating limits, operating limits for contractor's pilots, operating limits for Navy pilots, etc.) also apply to helicopters. The actual contractor demo requirements for Flying Qualities and Aerodynamic, Structural and Power-plant Requirements are set forth in MIL Spec MIL-H-8501 and NAVAER SR-189 respectively.

### 1.1.8 MAINTENANCE PRODUCTION & RESEARCH & DEVELOPMENT CONFERENCE

This conference is usually held while the contractor's preliminary flight tests are being run. Representatives from BuAer (structures, engine, controls, airframe and electrical), NATC, various operating squadrons' representatives and others attend this conference. The aircraft is completely and thoroughly inspected for any violation of MIL Specs or any part of the pertinent aircraft specification. Other discrepancies are also checked such as arrangement of cockpit, arrangement of instruments, lighting, etc. Discrepancies that violate the detail specification or MIL Specs are corrected prior to delivery of the aircraft to the fleet; others are put in various categories for correction or study.

### 1.1.9 BIS AND PTR TESTS

Test work performed by F. T. Rotary Wing is either for the Board of Inspection and Survey or Bureau of Aeronautics. Board of Inspection and Survey projects are designated TED BIS trials and Bureau of Aeronautics tests are designated TED PTR (Patuxent River).

Board of Inspection and Survey Aircraft Directives prescribe the procedures applicable to the conduct of service acceptance trials or production inspections of new models of piloted naval aircraft.

The Bureau of Aeronautics Manual of Test Directives specifies in general terms instructions for the conduct of PTR tests of a recurring nature. Directives NOs 3-3,3-4, 4-5 and 4-6 specifically apply to rotary wing aircraft.

1.1.10 PROJECT NUMBER ASSIGNMENT

The Bureau of Aeronautics assigns all projects for the Naval Air Test Center. Each project is assigned an individual identifying number code. The code numbers for TED PTR Projects are standard for certain phases of the tests i.e. all Pilot Handbook Projects are designated PTR AC-1312. The name (HR2S-1) designates the particular aircraft concerned. In other projects the first two numbers of the TED PTR-AC-number designates the cognizant BuAer desk concerned; while the point number designates the project test phase. The following project numbers on the HR2S-1 helicopter are used as an example:

a. TED PTR-AC-4024.1 - The cognizant BuAer class desk office is AC-4. The designated project number for the HR2S-1 is 4024, the point number (.1) shows that this is phase 1 of the NPE.

b. TED PTR-AC-4024.1.1 - The point number (.1.1) indicates that this is the second NPE project on the HR2S-1.

c. TED PTR-AC-4024.2 - Contractor's Part II Demo., Performance Phase

d. TED PTR-AC-4024.3 - Contractor's Part II Demo., Stability and Control Phase

e. TED PTR-AC-4024.4 - Electronics Installation Phase

f. TED PTR-AC-4024.5 - Tactical Suitability and Operating Capabilities Phase

g. TED PTR-AC-1312(HR2S-1) - Handbook Review. BuAer also designates the project numbers for BIS trials but the number of the project for a specific aircraft undergoing these tests is the same for all phases of the BIS trials.

1.1.11 NAVY PRELIMINARY EVALUATION (NPE)

a. The Navy Preliminary Evaluation (NPE) is authorized and directed by BuAer as are all tests. It is known as a Patuxent River TED (PTR) test. The purpose of the NPE is to evaluate the general performance, flying qualities and tactical suitability characteristics of the aircraft at the earliest possible stage in its development. The ability of the aircraft to perform the design mission is also evaluated. In this way deficiencies are pointed out and corrected in the early development phase of the aircraft and, this results in a better article being delivered to the fleet with a minimum time delay. On the strength of the NPE a recommendation is made as to when the aircraft will be suitable for BIS trials.

The NPE may be in one or more phases and may take one or more weeks to perform depending on the weather, availability of aircraft and extent of the tests.

F. T. Inst. 3960.1 and NATC Inst. 5200.1 give instructions for conducting the NPE. A detailed and comprehensive description of the responsibilities of both project pilot and engineer are given in the Rotary Wing Pilot and Engineer Info Folder.

### 1.1.12 CONTRACTORS PART II DEMONSTRATION

The purpose of the Part II Demo is to fulfill the requirements of the contractor's demonstration addendum. The Part II Demo is usually completed before the start of the BIS trials. This is not always so, however; at times the BIS Preliminary Phase will be in progress at the same time as the Contractors Part II Demo. The Structural and Aerodynamic Demo is conducted at the Naval Air Test Center and is witnessed by the representative of the Commander NATC. NAVAER SR-189 is the BuAer Specification for the aerodynamic structural, and power-plant requirements for helicopters. MIL-H-8501 delineates the flying qualities requirements for U. S. military helicopters. The demonstration addendum reflects these requirements to be demonstrated by the Contractor during the Part II demonstration.

### 1.1.13 BOARD OF INSPECTION AND SURVEY (BIS)

A field office of the Board of Inspection and Survey was established at Patuxent River in July 1943 for the purpose of expediting the conduct of tests under the cognizance of the Board of Inspection and Survey. BIS normally deals with aircraft for which new contracts have been let or are contemplated. This means that BIS trials usually involve new type aircraft, or perhaps comparatively severe modifications of existing types. The major responsibility of the BIS in the procurement of aircraft consists of determining that the government is actually getting an item equal to or better than that specified by contract and detail specification. Flight Test acts as a technical assistant to the Board of Inspection and Survey and conducts the aircraft and engine performance and stability and control trials which are part of the overall BIS trials.

1.1.14 BIS INITIAL TRIALS PHASE (ITP)

The Board of Inspection and Survey Initial Trials Phase is a program wherein the BIS carries out an accelerated evaluation, normally during a period of sixty days following the delivery of one aircraft to each of the test activities participating in the BIS, Initial Trials Phase for the specified model. It is a brief qualitative evaluation of service readiness (including support equipment) to establish that the aircraft and its basic components of power plant, armament, and electronic items are ready for fleet use and capable of fulfilling the basic mission of the aircraft.

1.1.15 BIS INITIAL TRIALS PHASE CONFERENCE (Patuxent River)

After the test divisions concerned have completed the ITP (Initial Trials Phase) of the BIS trials, BIS calls a conference which is held in the BIS Office at Patuxent River. Representatives of each unit, usually the branch head, project pilot and project engineer, meet with BIS and discuss the respective reports on the subject aircraft. The purpose of this meeting is to prepare for the BuAer Aircraft Trials Committee Conference which is held at BuAer. Each deficiency reported by the test units is discussed and an agreement or decision is made as to which category the deficiency belongs; that is, deficiencies to be corrected before delivery of the aircraft to the fleet, deficiencies to be corrected on a high priority basis to improve service use, deficiencies to be corrected on a lower priority basis. Based on the decisions and recommendations of this conference BIS submits a report to the President of the Board of Inspection and Survey. The preliminary reports of the various test units becomes a part of this report as enclosures.

1.1.16 AIRCRAFT TRIALS COMMITTEE (BuAer)

This conference may also be known as BIS Initial Trials Phase Conference. The conference is held at BuAer a short time after the ITP conference at Patuxent River for the purpose of reporting the results of the evaluation of the aircraft and support equipment, and the determination of action to be taken. This conference is attended by representatives of CNO, Chief of Naval Air Training, Fleets, BuAer, NATC, BIS, Aviation Support Office, Naval Aviation Safety Center, and the contractors. The configuration of the aircraft for fleet delivery is established at the conference. A most important decision to be arrived at is the readiness of the aircraft and its associated equipment to commence the Fleet Introduction Program (FIP), the configuration of the model for the FIP and the starting date thereof.

1.1.17 FLEET INTRODUCTION PROGRAM (FIP)

The Fleet Introduction Program is an accelerated flight program which lasts approximately eight weeks, or 100 flight hr per aircraft, normally conducted at the Naval Air Test Center, Patuxent River, for the purpose of introducing a new model to fleet for indoctrination in the operation and maintenance of the aircraft. A secondary purpose is to provide a further check on fleet readiness. This program is conducted under the management control of BuAer with fleet aircraft and fleet personnel. It is separate and distinct from the test programs of BuAer which establish the contractual requirements and the limitations of the model. The decision of CNO governs whether or not a FIP is necessary or desirable. The FIP program is conducted and monitored by the Service Test Division.

1.1.18 FLEET INTRODUCTION PROGRAM CONFERENCE

A Fleet Introduction Program Conference is held either at the Naval Air Test Center or BuAer at the conclusion of the FIP. At this conference the results of the FIP are presented and any changes which are found necessary in the configuration of the aircraft for fleet delivery will be determined. At this conference the adequacy of spare parts, ground handling, special tools, and test equipment will be examined as regards both quality and quantity. The conference is attended by representatives of CNO, Chief of Naval Air Training, Fleets, BuAer, NATC, Aviation Supply Office, Naval Aviation Safety Center and contractors.

1.1.19 BIS TRIALS

After the Initial Trials Phase of the BIS Trials is completed, the aircraft is normally fully instrumented (provided this wasn't done prior to the ITP). This part of the tests consists of two separate phases - Aircraft and Engine Performance Trials and Stability and Control Trials (Flying Qualities Phase). The data obtained during the Navy

Preliminary Evaluation, the Contractor's Part II Demo and the Initial Trials Phase may be utilized to fulfill the requirements of these phases. However, the Contractor's data must be properly witnessed and substantiated before it can be used.

At least one letter report is submitted on each phase of the project. It is practical to include more than one phase in a report if it will not delay the reporting of either phase. This phase of the tests gives complete results of aircraft performance and stability and control, vibration analysis and other factors that must be known to determine if the contractual guarantees have been met. Data gathered on NPE and ITP phases may be largely qualitative while the data on this last phase of the BIS trials is mostly quantitative.

1.1.20 TED PTR PROJECTS

TED PTR projects usually involve performance and flying quality evaluations of new aircraft not undergoing BIS trials. As mentioned before, Navy Preliminary Evaluations and the Contractor's Part II Demo are PTR projects, as is the Handbook Review.

1.1.21 FLIGHT TEST INTERNAL PROJECTS

Flight Test Division Internal Projects (FTDIP) are technical investigations not directly associated with BIS/PTR projects. The internal project system establishes these investigations on a formal basis in order to complete the work in a more efficient manner, establish priorities, authorize expenditure of funds and to assure proper publication of the results. This manual is an example of an internal project.

1.1.22 HANDBOOK REVIEW

A Flight Handbook Review project (PTR) is conducted on all new aircraft handbooks. This project is authorized by BuAer and is usually established during the early phases of the BIS Tests on the pertinent aircraft. Corrections or additions to the existing flight handbook are submitted on the standard Flight Handbook Revisions form. The flight handbook is studied very closely during the tests on the aircraft and any revisions needed should be submitted as soon as the discrepancies are discovered. Some of the most common revisions concern mistakes made in printing, discovery of improved operating techniques, etc. In order to get improved operating techniques, helicopter deficiencies, and maintenance items to the Fleet at the earliest the above items are published under project MA-3387.

### 1.1.23 YELLOW SHEETS

A "yellow sheet" is an established form for rapid reporting of deficiencies discovered during the BIS trials in design, materials, workmanship or inspection resulting in malfunctioning, failures or unsatisfactory performance of the aircraft, its structure components, or equipment. This includes failure to meet the guarantees incorporated in the contract. Repeated failures or unsatisfactory operation of structures, components, or equipment should be individually reported as they occur, since frequency of occurrence often has a definite bearing on corrective action taken.

### 1.1.24 TEST PROGRESS DATA SHEETS

A test progress data sheet ("pink sheet") is a standard form for reporting deficiencies discovered during PTR tests in design, material, workmanship or inspection resulting in malfunctioning, failures, or unsatisfactory performance of the aircraft, its structure components or equipment. Each failure or discrepancy should be reported individually as it occurs. Frequency of occurrence often has a definite bearing on corrective action taken.

### 1.1.25 DESIGNATION OF "Y" AND "Y/"(slant) AIRCRAFT

BuAer R & D designates certain aircraft as "Y" Project in accordance with BuAer ltr Aer 3740/229 of 15 May 1958. These particular aircraft are authorized to carry the prefix "Y", and become part of the non-program vice R & D program inventory. In order to separate the "Y Project" aircraft models from the BIS models in the various OPNAV publications and IBM listings (both categories have been authorized the "Y" prefix), the "Y Project" aircraft will carry a slant after the "Y" prefix for identification purposes only as follows:

| Aircraft Model | Non-Program Category |
|---|---|
| Y/A3D-2 | Project Development |
| Y A3D-2 | BIS |

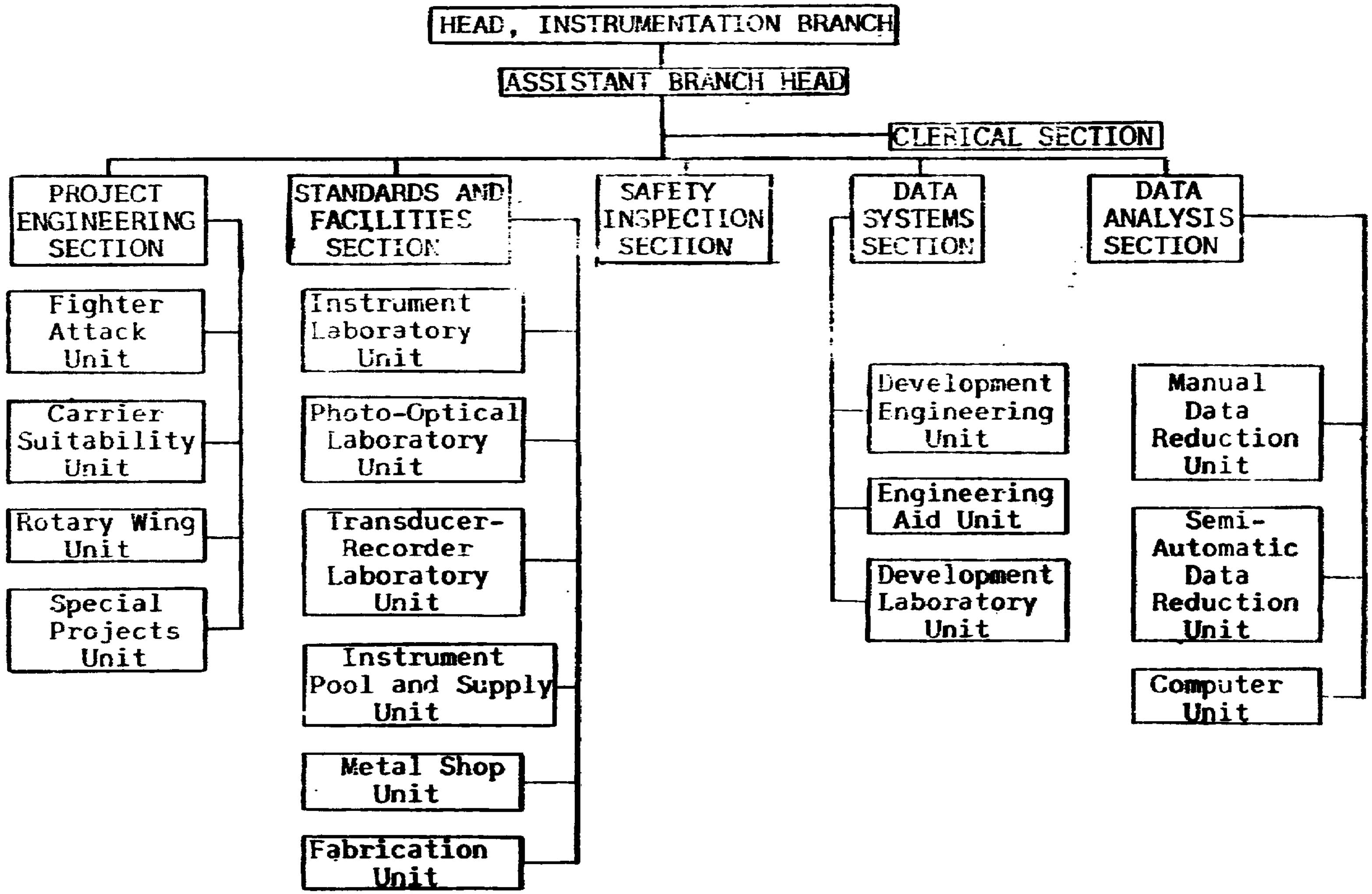
"Y/" aircraft are destined to remain in R & D custody during service life. "Y" BIS aircraft retains the "Y" prefix only while the aircraft is undergoing BIS trials.

INTRODUCTION

Section 2.1

## 2.1.1 DUTIES AND ORGANIZATION

The function of the Instrumentation Branch of Flight Test is to install, calibrate and maintain the instrumentation for various designated aircraft; check contractor instrumentation installations and calibrations; supply and maintain the external measuring devices necessary for flight testing; and support contractor instrumentation personnel where possible. The branch is divided into the following sections: clerical, data systems, data analysis, safety inspection, standards and facilities, and project engineering. The project engineering section is divided into the following units: fighter and attack, carrier suitability, rotary wing and special projects. An organization chart is shown in figure 2.1:1. The Rotary Wing Unit is responsible for all instrumentation used in conjunction with rotary wing aircraft and acts as the rotary wing coordinator between the other sections of Instrumentation.




INSTRUMENTATION ORGANIZATION CHART
Figure 2.1:1

INSTRUMENTATION CONFERENCE

**Section 2.2**

## 2.2.1 INSTRUMENTATION PROCEDURES

When a helicopter is to be instrumented completely or partially by the contractor, a conference should be held early in the program (eight months before the aircraft is expected to arrive or prior to the time the helicopter instrumentation commences) with the contractor to settle on the instrumentation that will be installed in the demonstration and/or BIS test helicopter. The obligation to confer is specified by MIL-D-8708 (Aer) for demonstration aircraft. The specification requires that the conference be held at the Naval Air Test Center. To prepare properly for the conference, the cognizant test personnel should review the detail specifications of the helicopter and test directives.

It is recommended that the following personnel attend the conference:

Rotary Wing Branch:

- Project Officer
- Project Engineer
- Engineering Section Head

Instrumentation Branch:

- Rotary Wing Unit Project Engineer
- Project Engineering Section Head
- Data Systems Section Head

Contractor:

- Structural Engineer
- Instrumentation Group Representative (s)
- Project Personnel (as contractor wishes)
- Senior Patuxent Field Service Representative (not essential but frequently helpful in settling questions of support for helicopter during trials)

The agenda of the conference should include as a minimum a determination of the following:

a. Specific items to be recorded

b. Type and number of recording devices

c. Location of instrumentation recording devices in the helicopter

d. How the recording device or devices will be mounted

e. Who will supply the instrumentation

f. Who will install the instrumentation

g. Extent of calibration required for each component

h. Accuracy required

i. Range of loads to be recorded

j. Frequency response

k. Who will maintain the instrumentation during the trials

l. Who will reduce instrumentation records subsequent to the trials

m. Who will retain the original records subsequent to the trials

Subsequent to the instrumentation conference with the contractor, the following action is required of the participating groups:

a. Instrumentation Branch prepares the Instrumentation Specification along with the wiring diagrams and installation instructions. These are prepared in accordance with Flight Test Division Instruction No. 4121.1 of 19 September 1955, titled "Instrumentation Specifications for BIS Aircraft, Contractor's Instrumentation Proposals for Demonstration Airplanes and Cost Proposals; procedures for handling of." A sample Instrumentation Specification page is shown in figure 2.2:2.

b. Rotary Wing Branch reviews the Instrumentation Specifications with particular emphasis on the items to be recorded and the range and accuracies required. A letter is then prepared by the Rotary Wing Branch forwarding the instrumentation specifications. The letter should include the branch

INSTRUMENTATION SPECIFICATION

AIRPLANE HSS-2 TYPE OF TEST BIS Stability and Control and Performance

| ITEM | MEASUREMENT | TYPE OF INSTRUMENT | CALIBRATED ACCURACY | RANGE | * | REMARKS | ** |
|---|---|---|---|---|---|---|---|
| 1 | Photopanel | NATC Type 26 hole | --- | --- | - | see note 1 | N |
| 2-3 | Free Turbine RPM | Percent Tachometer | ± 1% | 102% | P | --- | C |
| 4-5 | Gas Turbine RPM | Percent Tachometer | ± 1% | 102% | P | --- | C |
| 6-7 | Star Load | Ammeter Hickok Mod.46 | ± 2.5% | as req'd for expected loads | P | see note 2 | N |
| 8-9 | Airspeed | Kolls. Ind. mn586-BK-10-0155 | ± 1Kt | 0 to 170 Kts | P | see note 3 | N |
| 10-11 | Altitude | Kolls. Ind. R-88-A-350 | ± 20Ft | 50,000 Ft | P | see note 3 | N |
| 12 | Time | Clock R-88-C-590 | --- | 8 da.civil date | P | --- | N |
| 13 | Camera Burst Count | Veeder Root x62801-5 | --- | four digit | P | | N |

* RECORDING SYSTEM: T-Telemeter, P-Photo Observer; M-Magnetic Tape; O-Oscillograph

** "N" INDICATES NATC FURNISHED AND "C" INDICATES CONTRACTOR FURNISHED

Figure 2.2:2

Page 1 of 6,

requirements for the contractor assistance agreed upon at the conference and any other requirements that will have to be paid for by the Bureau of Aeronautics.

c. The contractor, upon receipt of the specification and upon determination of his instrumentation installation, submits a report on the calibration and installation to the Bureau of Aeronautics via the Commander, Naval Air Test Center. This arrangement offers the Rotary Wing Branch an opportunity to review the contractor's instrumentation and recommend changes as desired, thus minimizing disagreements over instrumentation data during tests.

## 2.3.1 NORMAL TEST EQUIPMENT

Test instrumentation in a helicopter usually includes a photopanel with a 35 mm camera and an oscillograph. Both are actuated by the pilot triggering a switch usually contained on the cyclic stick.

The calibrated instruments usually contained in the photopanel are as follows:

a. Record counter

b. Clock

c. Stop watch

d. Sensitive altimeter (service system)

e. Sensitive altimeter (trailing bomb)

f. Sensitive airspeed indicator (service system)

g. Sensitive airspeed indicator (trailing bomb)

h. Sensitive engine tachometer

i. Sensitive rotor tachometer

j. Outside air temperature gage

k. Carburetor air temperature gage

l. Sensitive carburetor deck pressure gage

m. Sensitive manifold pressure gage

n. Fuel flowmeter

The oscillograph usually records the following information:

a. Longitudinal cyclic stick position

b. Lateral cyclic stick position

c. Collective stick position

d. Rudder pedal position

e. Throttle position

f. Mixture position

g. Relative heading

h. Sideslip angle

i. Roll angle

j. Pitch angle

k. Rate of roll

l. Rate of pitch

m. Rate of yaw

n. Linear acceleration

o. Angular acceleration

p. Rotor shaft torque

q. Engine shaft torque

r. Tail rotor shaft torque

s. Vibrations (frequency and amplitude)

t. Airspeed

u. Altitude

v. Pressure

w. Structural loads

In addition to the photopanel and oscillograph recording instruments, the following special instruments are installed in the pilot's compartment:

a. Sensitive manifold pressure gage (replacing service instrument)

b. Sensitive airspeed indicator (replacing service instrument)

c. Outside air temperature gage

d. Carburetor air temperature gage

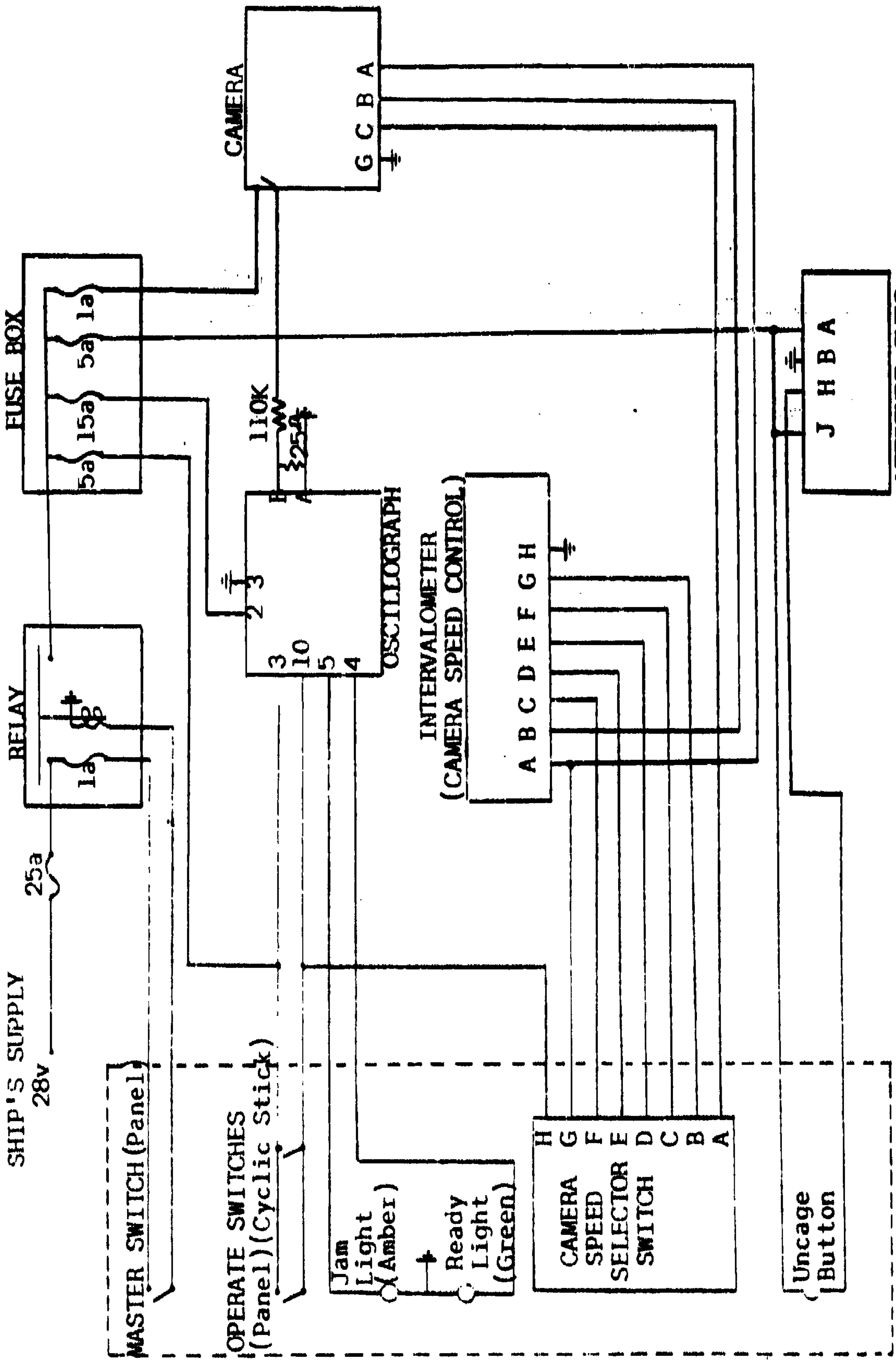
e. Fuel flowmet - indicator.

Prior to test flights, all instruments in the photopanel and oscillograph, special instruments in the pilot's compartment and the service engine-rotor tachometer and service altimeter are calibrated by the Instrumentation Branch of Flight Test. The Rotary Wing project engineer makes an official request for these calibrations using the standard work order form (AVO) PRNC-NATC-702 (Rev 4-53). The results of the calibrations will be reported to the Rotary Wing project engineer concerned in the form of calibration cards of photopanel and pilot compartment instruments and in the form of calibration curves for oscillograph instruments. The Rotary Wing project engineer will plot correction curves for photopanel and pilot compartment instruments where applicable. Copies of correction curves and calibration curves are sent to the Data Analysis Section of Instrumentation.

The project pilot must be familiar with the instrumentation operational control switches and lights located in the cockpit. Figure 2.3:1 shows a typical power circuit diagram of an instrumentation installation and includes cockpit instrumentation operational controls and lights.

Instrumentation pre-flights and post-flights are to be made before and after every instrumented test flight. The Rotary Wing project engineer will call the Rotary Wing Unit head of Instrumentation and give notification of these events. It is desired that Instrumentation be notified one-half hour before take-off time and immediately after landing. Prior to flight, Instrumentation personnel are to assure that all necessary switches and circuit breakers are positioned correctly and that all recording clocks are wound.

After oscillograph and photopanel film are removed from the helicopter by Instrumentation personnel, the film is deposited in the Instrumentation Photo-Optical Section where it is processed and developed. The film is then visually checked for discrepancies by the Instrumentation Rotary Wing




INSTRUMENTATION POWER CIRCUIT

**Figure 2.3:1**

Unit and forwarded to the Data Analysis Section. The Data Analysis Section contacts the Rotary Wing project engineer to determine what information is desired. The Data Analysis Section then proceeds to read, reduce and plot the desired data and forwards the results to the Rotary Wing project engineer.

### 2.3.2 OSCILLOGRAPH

An oscillograph is an instrument that is capable of permanently recording data with the use of D'Arsonval galvanometers and photosensitive paper. The D'Arsonval galvanometer has the capability of recording stabilized and also, due to its low inertia, high frequency phenomena up to 3000 cycles per sec. Correct damping, to prevent overswing of the galvanometer and to linearize frequency response over the expected range of frequency to be encountered, is attained by combining the proper interchangeable galvanometer and the proper circuiting.

The galvanometers, one for each channel, are grouped side-by-side in a common magnetic block. Each galvanometer consists of a small mirror attached to a small wire coil mounted on a rotatable vertical axis and suspended in a housing. When current flows through the coil, the magnetic field set up around the coil due to current flow reacts with the field of the magnetic block causing the coil to deflect through an angle proportional to the current flowing through the coil. The attached mirror deflects a fine beam of light laterally on a moving strip of photosensitive paper. This photographs the trace of the channel with time base provided by the moving paper. For high timing accuracy, timing lines appear across the full record width at intervals of 1/100 sec, with every tenth line being slightly heavier to simplify interpretation. The lines are generated by a light beam sweeping the paper at a precisely controlled rate. The paper speed may be varied to accommodate the test conditions; for example, high frequency information should be recorded with a high paper speed in order to expand the time scale for better readability. The oscillograph type most often used at NATC has 18 channels and a paper width of 7 in. Other models have from 6 to 36 channels and paper widths from 2 to 12 in.

### 2.3.3 POTENTIOMETERS

A potentiometer consisting of a circular wire winding and a contact arm that is connected to a shaft fitted with a helically grooved spring return pulley may be used to measure the travel of movable members of the helicopter. The unit is rigidly mounted on a stationary member in the vicinity of the movable member. A small cable is fastened to the pulley at one end and attached to the movable member at the other end with such geometry that travel of the movable member causes rotation of the potentiometer shaft thereby changing the effective resistance of the potentiometer an amount proportional to the amount of travel of the movable member.

Other potentiometers are integral parts of attitude and rates of change of attitude gyros. An attitude gyro measures angles of roll and pitch. It consists of an electrically driven gyroscope mounted in a pair of gimbals. The outer gimbal has a longitudinal axis of rotation and the inner gimbal a lateral axis of rotation. A circular wire winding of a potentiometer is mounted inside the unit's casing around a longitudinal axis. The contact arm of this potentiometer is mounted on the outer gimbal. When a helicopter rolls, the casing of the attitude gyro rolls with the helicopter around the gyro and both gimbals causing the wire winding to move in relation to the contact arm resulting in a change in effective resistance of the potentiometer proportional to the angle of roll. Another circular wire winding is mounted inside the outer gimbal around a lateral axis. The contact arm of this potentiometer is mounted on the inner gimbal. When a helicopter pitches, the casing and outer gimbal pitch with the helicopter around the inner gimbal causing the wire winding to move in relation to the contact arm resulting in a change of effective resistance of the potentiometer proportional to the angle of pitch.

A rate gyro is an electrically driven gyroscope mounted in a single gimbal. The gimbal has the same axis of rotation as the precession axis of the gyroscope. As the gyro rotates about an axis perpendicular to the spin axis, a gyroscopic couple is produced causing the gyro and gimbal to precess. The precession is opposed by a cantilever spring restraining the gimbal causing the angle to which the gyro and gimbal precess to vary proportionately to the gyroscopic couple imposed by the rotation of the gyro. As the gyro and gimbal precess, a sliding contact arm, attached to the gimbal, moves over the wire winding of a potentiometer, attached to the stationary gimbal mounting, resulting in a change in effective resistance of the potentiometer proportional to the rate of rotation.

A potentiometer may also be used to indicate sideslip angle. The sideslip transducer, mounted on a nose boom, consists of a small horizontal weather vaning probe that maintains a heading into the relative wind and is mounted on a vertical potentiometer shaft. Inducing a sideslip angle causes the potentiometer shaft to rotate a contact arm over the circular wire winding of the potentiometer thereby changing the effective resistance of the potentiometer an amount proportional to the amount of sideslip.

### 2.3.4 STRAIN GAGES

A wire wound strain gage consists of a few inches of small diameter high resistance wire wound in the form of a grid and bonded to a small piece of paper. The two ends of the wire are attached to two strips of larger diameter low resistance wire which are the terminals.

If the strain gage is glued to a piece of metal and the metal is stretched, the strain gage wire will be stretched and its resistance will increase. If the metal is compressed, the strain gage wire will also be compressed and its resistance will decrease. To obtain maximum change in resistance, the strain gage should be mounted parallel to the strain in the metal.

The versatility of strain gages is such that they are capable of measuring bending, axial and torque loads, depending on their geometrical placement; linear and angular accelerations; vibration frequencies and amplitudes; pressures; altitudes; and airspeeds. Accelerations and vibrations are measured by bonding strain gages to cantilever beams that are stressed due to deflections caused by acceleration and vibration forces. Pressure, altitude, and airspeed are measured by bonding strain gages to diaphragms that are stressed due to differential pressures induced into the transducer being used.

Strain gages are also capable of being used in place of potentiometers to measure control travel or angular velocities. To measure control travel, strain gages are bonded to a cam-deflected cantilever beam. The cam rotates with the movement of the control to which it is linked, increasing or decreasing the deflection of the beam thereby increasing or decreasing the stress in the beam.

To measure angular velocities, a rate gyro is modified by placing strain gages on the cantilever spring that restrains the precession of the gyro and its gimbal. Precession causes the cantilever spring to deflect inducing stress in the spring an amount proportional to the angular velocity

### 2.3.5 GALVANOMETERS

Oscillograph sensing devices usually consist of potentiometers or strain gages incorporated in a Wheatstone bridge. The change of resistance of a potentiometer or a strain gage incorporated in a Wheatstone bridge will result in an electrically unbalanced bridge and cause current flow through a galvanometer connecting opposite arms of the bridge. In this manner, the galvanometer, placed in an oscillograph, is capable of recording desired information.

Galvanometers may also be used to pick up signals other than those resulting from an electrical unbalance of a Wheatstone bridge. Signals may be generated by induction from a strain gage bonded to a cantilever beam vibrating through field of a permament magnet or induced by moving a magnetic substance through an electrical field. The transducer used to measure amplitude and frequency of helicopter vibrations is an example of the former, and the pickup used to measure RPM of a shaft is an example of the latter. A signal may also be produced by means of a micro-switch periodically completing an electrical circuit between a voltage source and the galvanometer. An example of this is the camera correlation signal that is transmitted for every camera frame taken.

### 2.3.6 TORQUEMETERS

It is necessary to determine engine horsepower for reduction of performance data. With each helicopter being tested, the engine manufacturer submits an engine power chart. This chart is the result of a test stand calibration of the bare engine without aircraft induction and exhaust system and is not necessarily correct in all regimes of flight; therefore, a torquemeter installation is desirable in all helicopters undergoing performance tests. The torquemeter may be mounted on the engine or rotor shaft, whichever is more convenient. A torquemeter may also be mounted on a tail rotor shaft when measurement of tail rotor horsepower is desired.

A torquemeter consists of four strain gages bonded to the shaft at angles of 45° to the centerline of the shaft and at intervals of 90° around its circumference with each strain gage at a 90° angle to its adjacent strain gage. These strain gages are wired such as to compose a four arm Wheatstone bridge. When torque is applied to the shaft, two opposite arms of the bridge are activated to cause a current flow through a D'Arsonval galvanometer connecting alternate junctions of the bridge. All axial and bending loads are cancelled out due to the geometrical location of the strain gages.

2.3.7 CALIBRATIONS

A calibration of a torquemeter is made by applying known amounts of torque to the torquemeter shaft and recording the resulting oscillograph trace deflections. A calibration curve of torque versus trace deflection is drawn. Using this calibration curve would result in a true indication of torque applied to the calibrated shaft if the deflection sensitivity of the trace were to remain the same as it was during the calibration; however, this is very unlikely. Therefore, a resistor of very high resistance, called a calibration resistor, and a series switch, called an Rcal switch, are wired in parallel with one of the arms of the Wheatstone bridge. Closing the Rcal switch causes the calibration resistor to be in parallel with the resistor of the arm of the Wheatstone bridge thereby decreasing the effective resistance of that arm of the bridge. This causes an unbalance in the bridge resulting in a trace deflection. The resulting trace deflection at the time of calibration represents an equivalent torque as determined by plotting the Rcal trace deflection on the calibration curve. This value of equivalent torque will remain the same regardless of sensitivity change. All other Wheatstone bridge calibrations are made in a similar manner.

All oscillograph calibration graphs received from instrumentation contain the following information: helicopter type and bureau number, instrument being calibrated, oscillograph channel number, direction of trace deflection when installed in the helicopter, resistance of Rcal resistor, and equivalent Rcal value. All this information is necessary before reading oscillograph records.

2.3.8 OSCILLOGRAPH RECORD REDUCTION

Every oscillograph has a trace interrupter to aid in identifying channels. The order of trace interruption may be obtained from the Rotary Wing Unit Head of Instrumentation.

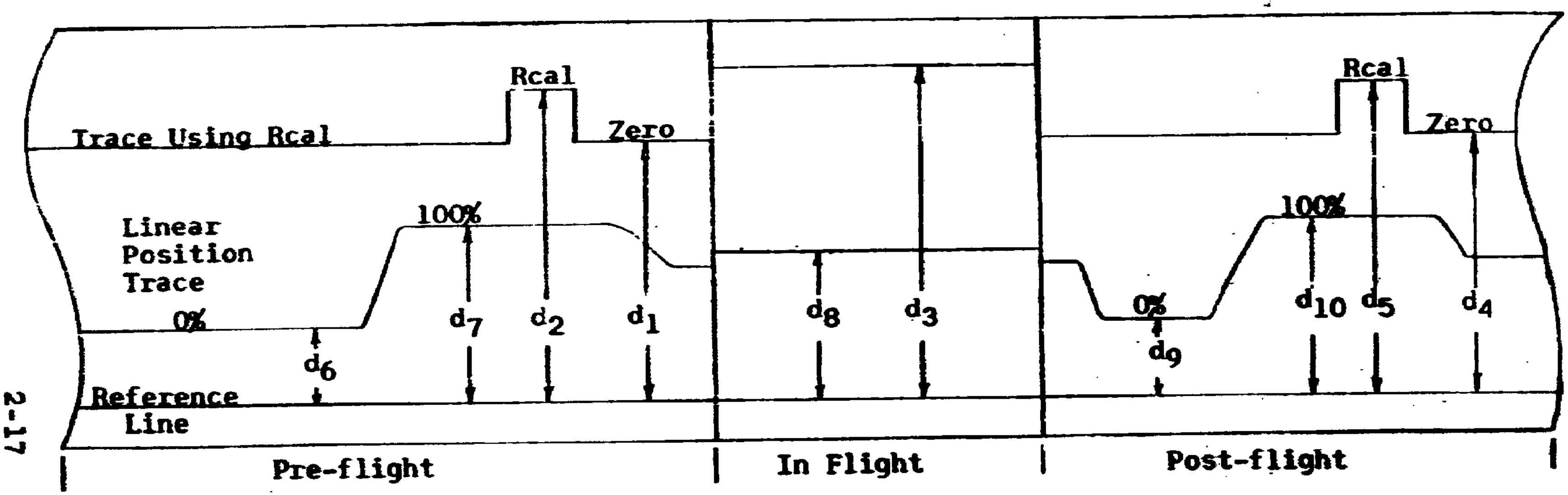
Every oscillograph flight record should contain pre-flight and post-flight records that consist of traces of zero load conditions and Rcal deflections or full throw deflections for each channel. When a position calibration curve is linear, a record of the full travel of the trace recording the position is taken rather than recording its Rcal deflection.

Every oscillograph record has one or more stationary traces that are used as reference lines. All trace readings are measured from the same reference line for all records of the same flight. A glass reader may be used to measure, to the nearest .01 in, trace distances from the reference line.

To determine flight sensitivity of a channel trace having a linear calibration, the pre-flight and post-flight zero load distances from the reference line are averaged ($\frac{d_1 + d_4}{2}$, fig 2.3:2) and subtracted from the pre and post-flight average of the Rcal deflections ($\frac{d_2 + d_5}{2}$, fig 2.3:2). The resulting average delta Rcal deflection is divided into the calibration Rcal equivalent load to determine the sensitivity in load units per inch of trace deflection.

To obtain loads from flight records, the average zero load deflection obtained from the pre and post-flight ($\frac{d_1 + d_4}{2}$, fig 2.3:2) is subtracted from the load deflection ($d_3$, fig 2.3:2) measured from the reference line to obtain the delta load deflection. The product of the delta load deflection and the sensitivity equals the load.

For linear position indicators, a full throw is recorded in place of an Rcal deflection and is considered as an Rcal with an equivalent load equal to 100% of full travel of the position being measured. Either of the extremes of the trace travel may be considered as the zero trace deflection. Sensitivity and flight load determination is similar to that using Rcals (i.e., Sensitivity = Calibration Equivalent ÷ $\left(\frac{d_7 + d_{10}}{2} - \frac{d_6 + d_9}{2}\right)$, fig 2.3:2; Flight Load = Sensitivity x $d_8 - \frac{(d_6 + d_9)}{2}$, fig 2.3:2).




**SAMPLE OSCILLOGRAPH REDUCTION RECORD**

**Figure 2.3:2**

If a channel has a non-linear calibration, the trace sensitivity will not be a constant. To determine flight loads, the delta load deflection ($d_3 - \frac{(d_1 + d_4)}{2}$, fig 2.3:2) is multiplied by the ratio of the calibration delta Rcal (from calibration curve) over the average pre and post-flight delta Rcal deflection ($\frac{d_2 - d_1 + d_5 - d_4}{2}$ fig 2.3:2). Enter the calibration curve using this new delta load deflection to read the load.

Pre and post-flight zero load deflections and delta Rcal deflections or full throws may differ by a percentage greater than that which is desired. If such is the case, using pre and post-flight averages may result in inaccurate data reduction. Many instrumentation installations include an automatic Rcal box that automatically Rcal's each channel at the end of each record. If the difference between pre and post-flight Rcal delta deflections or full throws is too great, these intermediate Rcals may be used to determine the sensitivity of the trace at the time of recording. They will not, however, account for zero load trace shifts. If the difference between the pre and post-flight zero load deflections is too great, it may be assumed that the shift varied directly as a function of time.

In the event that there is no automatic Rcal box installed as an integral part of the instrumentation and pre and post-flight Rcal delta deflections or full throws differ greatly, the difference may be assumed to vary directly as a function of time. When conditions such as the above exist, it is best to repeat the type of flight being performed in order to substantiate the resulting data.

Provided the calibration curve is linear, the process of reading time histories or a large number of records is simplified by using a strip or stick of cardboard marked in increments of desired units. To obtain the correct increments to be marked on the cardboard stick, a mark is placed on the stick and is used as a reference to align the stick with the reference line of the oscillograph record. After alignment of the stick reference and the oscillograph record reference, two marks are placed on the stick, one at a distance from the reference equal to the average pre and post-flight zero load trace deflection and the other at a distance from the reference equal to the average pre and post-flight Rcal trace deflection.

The former mark is given a value of zero, the latter a value equal to the calibration Rcal equivalent load. With these values and the aid of a Gerber variable scale, a scale may be easily constructed. Flight loads may be read directly by aligning the stick and oscillograph record references. Similar application of cardboard sticks may be used for traces using full throws in place of Rcals. To use this method of trace reading the pre and post-flight differences must be within the desired accuracy.

Oscillograph traces may contain vibrations that are not a true indication of actual loads. These may be due to numerous causes. A potentiometer may have dirt on its windings causing erratic trace oscillations, or a strain gage may be oscillating due to external exciting frequencies. When a trace having a potentiometer as a transducer sporadically oscillates erratically through small portions of its range, a line should be drawn through the oscillating section connecting the non-oscillating sections. Load readings are read from this line. The engineering head of the Rotary Wing Unit of Instrumentation should be notified about such oscillations and the feasibility of eliminating them should be discussed.

When a trace having a strain gage as a transducer vibrates, an average value of the vibration should be read to determine the load. If the vibration amplitude is excessive, the head of the Rotary Wing Unit of Instrumentation should be notified and steps be taken to dampen the vibrations by means of circuitry changes.

There are traces in which a vibration is a true measurement of the load. These include vibration transducer traces and star load traces. Both the frequency and amplitude of vibration transducer traces should be read. For star load traces, the average load and the vibratory load range should be read.

### 2.3.9 PHOTOPANEL

The automatic photo-observer (photopanel) usually consists of a recording camera mounted in such a manner as to photograph a lighted instrument panel. The photo-observer is selected or designed to accommodate the required number of instruments, to provide maximum practicable image size, and to fit the space available. The electrically driven data recording camera is usually a commercial product designed for the purpose. The camera uses 35 mm film which is loaded in magazines of 100, 200 or 400 ft capacity. Frame speeds of ½, 1, 2, 4, 8 or 16 frames per sec may be selected. The camera has a standard lens mount so that the proper lens may be selected for the photo-observer dimensions. Focusing and exposure adjustments are also provided.

In addition to standard photopanel instruments, micro-ammeters may be installed to record items in the same manner as an oscillograph galvanometer.

### 2.3.10 READING PHOTOPANEL FILM

Photopanel records are read with a commercial type of film viewer that projects an enlarged image of a frame on a ground glass screen. The enlargement is from 19x to 30x depending on the type of viewer used.

It is not necessary to read every frame when reading photopanel film records unless a time history is desired. For steady state conditions, a visual average of each indicator for each burst should be recorded. When reading a climb record, indicators on only one frame should be read and recorded for each burst. The photopanel camera records a number for each burst. The beginning and end of bursts are easily delineated by an exceptionally light or dark frame.

### 2.3.11 RADIO TELEMETRY AND TAPE RECORDING

Radio telemetry is a special technique for obtaining data at a position remote from the data source. As a complete instrumentation system, it includes transducers, similar to those used for oscillograph and photopanel recordings; modulators; radio link; demodulators; recorders; and data processing equipment. The phenomena to be measured are first converted to electrical signals by transducers and then changed to forms that are suitable for transmission over a low fidelity radio link to remote receiving stations. There, demodulation produces output voltages which are proportional to the input data, and these voltages can be either recorded by oscillographs, pen recorders, or television monitoring screens; processed into tabulated form; or any combination of these.

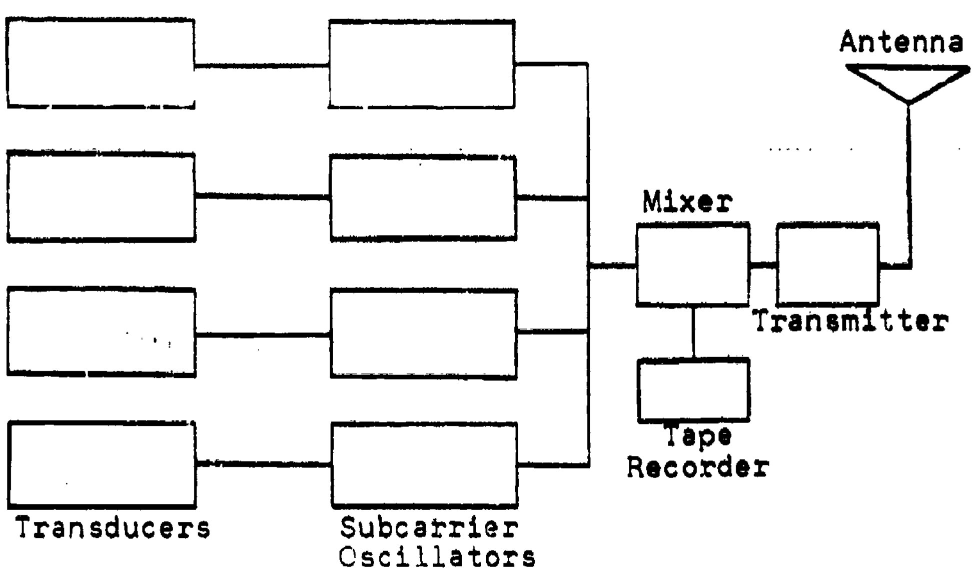
The fundamental processes involved in the transmission and reception of telemetry signals are essentially the same as those for ordinary radio communication with the intelligence to be transmitted being superimposed on a radio frequency carrier by one of several modulation methods. In ordinary radio transmission, the intelligence, which might be audio frequency sound, is modulated on the radio frequency carrier. This type of radio transmission is limited to one intelligence channel. An increase in intelligence channels can be obtained by the use of frequency division or time division multiplexing that makes it possible to transmit much more intelligence over a given portion of the radio frequency spectrum, in spite of increased radio frequency band width requirements for each individual transmitter.

In order to allow measurements to be recorded at the receiving stations with a high degree of accuracy, it is necessary to convert the transducer signals to a form that is unaffected by the low fidelity of the radio link. Therefore, transducer signals are converted to proportional frequency, phase, or time variations early in the system. When this is done the radio link serves merely as a carrier for the converted intelligence; and normal radio frequency transmission difficulties, such as atmospheric noise and signal fading, cause little difficulty.

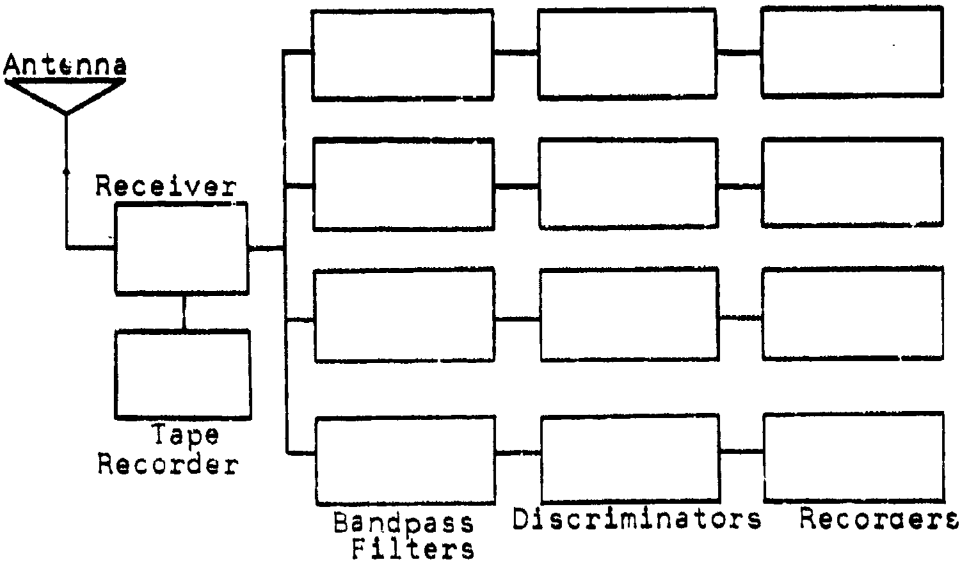
## 2.3.12 FREQUENCY DIVISION MULTIPLEXING

In frequency division multiplexing (fig 2.3:3), the direct current output of each transducer is used to modulate a subcarrier oscillator. Each subcarrier oscillator must differ sufficiently in frequency from all the other subcarrier oscillators to permit its separation at the receiving station. The output signals from the subcarrier are combined, and the composite signal thus obtained is used to modulate a single radio frequency carrier. The radio frequency carrier is detected at the receiving station, with the output of the receiver being a composite subcarrier signal similar to that at the input of the transmitter. The composite subcarrier signal is then separated into the individual subcarrier signals by means of a number of bandpass filters, each tuned to one of the subcarrier frequencies. Subcarrier discriminators demodulate the individual subcarrier signals and produce a direct current output which is proportional to the direct current output of the transducer at the data source.

Frequency division telemetry systems can be sub-divided into many individual systems classified by the type of modulation used on the subcarrier oscillators and on the radio frequency carrier. Several of these classifications are AM/FM (subcarrier oscillators: amplitude modulated; radio frequency carrier: frequency modulated), FM/AM (subcarrier




TRANSMITTING




RECEIVING

FREQUENCY - DIVISION MULTIPLEXING

Figure 2.3:3

oscillators: frequency modulated; radio frequency carrier: amplitude modulated), FM/FM (subcarrier oscillators: frequency modulated; radio frequency carrier: frequency modulated), PM/FM (subcarrier oscillators: phase modulated; radio frequency carrier: frequency modulated), etc.

### 2.3.13 TIME DIVISION MULTIPLEXING (COMMUTATION)

Time division multiplexing (fig 2.3:4) differs from frequency division in that the transducer outputs are sampled in a recurring sequence rather than simultaneously. There are a number of transducers connected to a multi-pole commutation switch (electronic or mechanical) which samples each of the transducers in turn. The output of the commutation switch is a train of pulses whose amplitudes represent the intelligence impressed upon the corresponding transducers. This train of amplitude modulated pulses is sent either directly to the radio transmitter, as is done in PAM (pulse amplitude modulation) systems, or to a converter for conversion to a form less easily disturbed by transmission difficulties. The converter changes each variable amplitude pulse into one of the following:

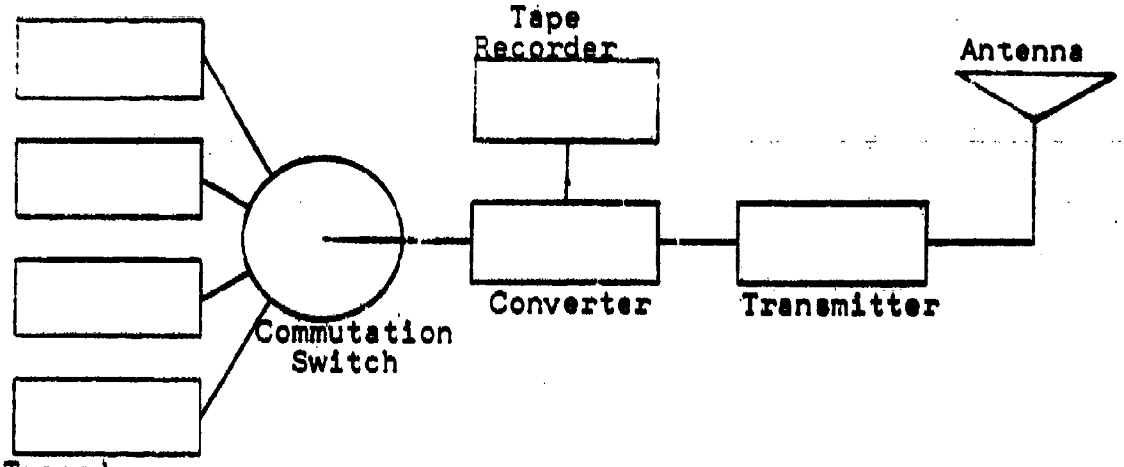
(1) A pulse whose duration is proportional to the amplitude of the input pulse (PDM or PWM: pulse duration modulation or pulse width modulation)

(2) A group of pulses forming a code representation of the amplitude of the corresponding input pulse (PCM: pulse code modulation).

The output of the converter is used to modulate the radio frequency carrier by any of the standard radio frequency modulation methods. The radio frequency signal is detected at a receiving station, with the output of the receiver being a train of pulses identical in nature to the output of the converter in the airborne equipment. These pulses can be separated by an electronic decommutation gating circuit, with the pulses representing individual data channels being placed on separate outputs. The pulses can then be converted to yield continuous voltages or digital codes representing the intelligence being transmitted.

### 2.3.14 TIME DIVISION SUB-MULTIPLEXING

Frequency division multiplexing systems can handle a relatively few data channels per radio frequency carrier, but have the ability to handle high frequency phenomena.

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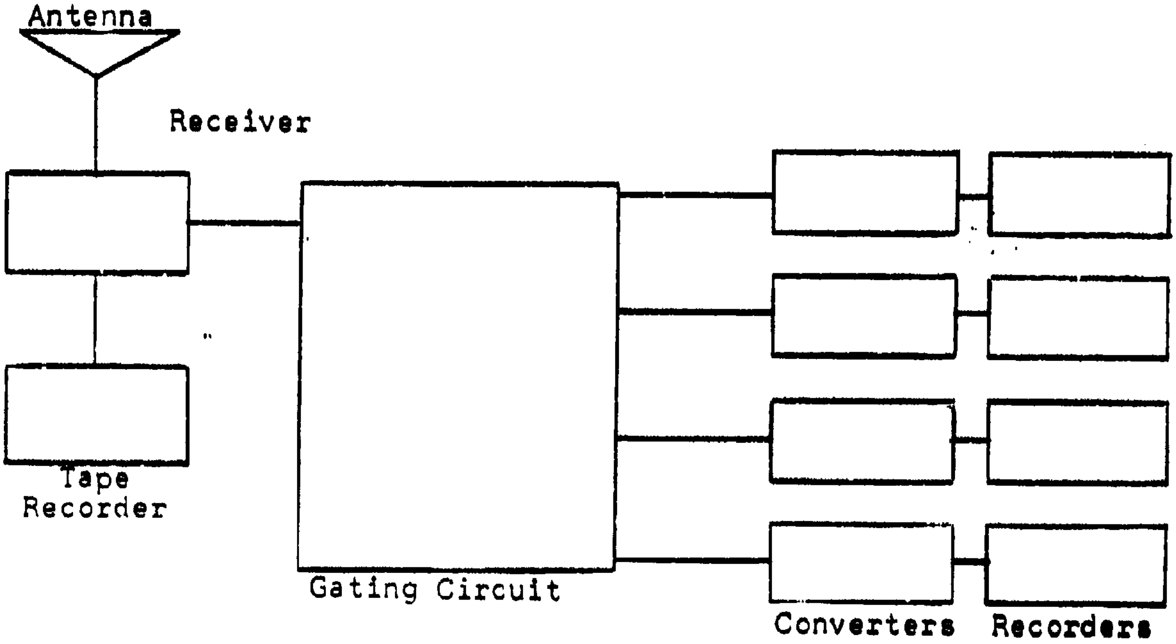



RECEIVING

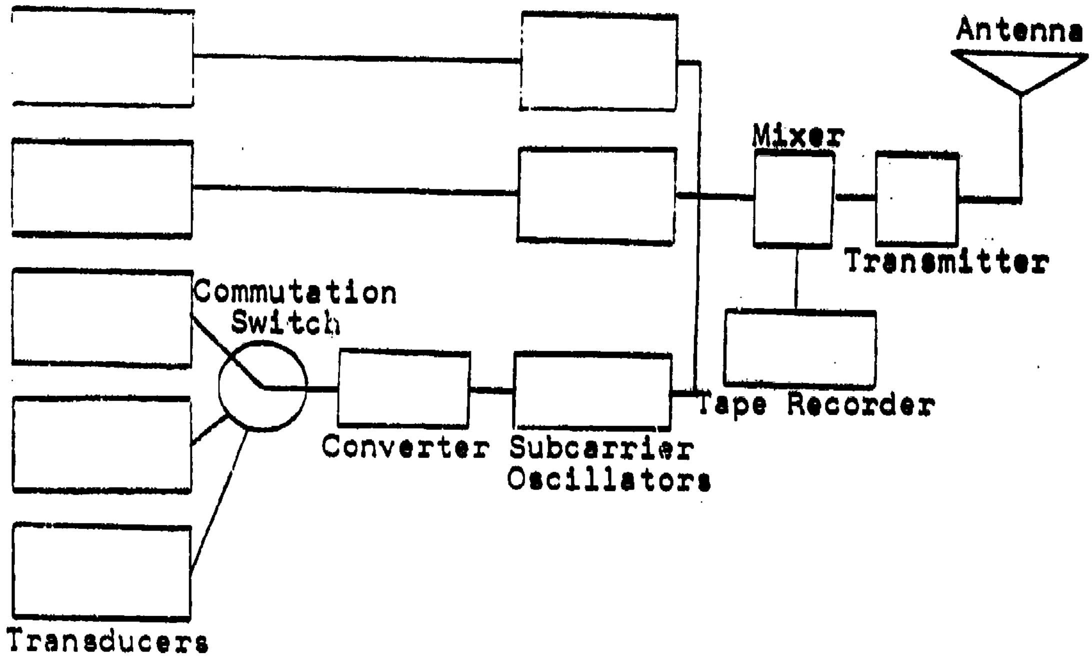
TIME - DIVISION MULTIPLEXING

Figure 2.3:4

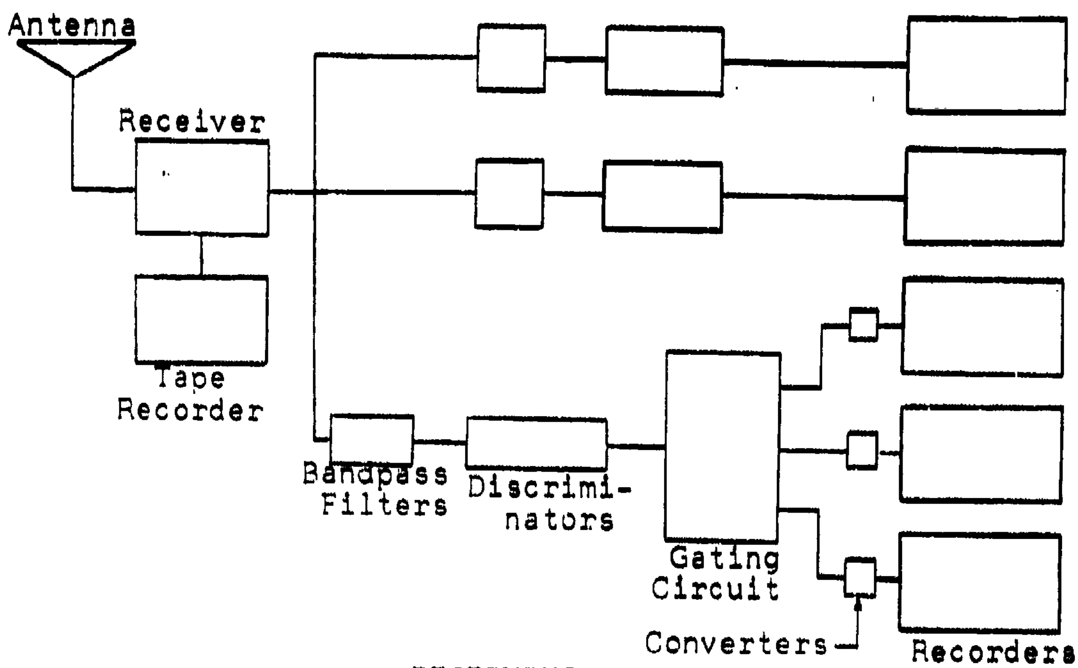
Time division multiplexing systems have the capability of handling a large number of data channels, but with a relatively low frequency response. To obtain the desired amount of data with desired frequency responses, a combination of the two systems can be used by converting a subcarrier band of a frequency division system into a time division subsystem (fig 2.3:5). High frequency phenomena can be recorded on the standard frequency division subcarrier bands. All other data can be recorded on the time division sub-system. Sub-systems are similar to standard time division systems with the exception that the converted signals modulate a subcarrier oscillator. The signal from this subcarrier oscillator and the signals from the other subcarrier oscillators are combined to modulate the radio frequency carrier.

## 2.3.15 MAGNETIC TAPE RECORDING

Magnetic tape recordings are used in conjunction with radio telemetry. The same information being transmitted from aircraft to ground station can also be recorded on airborne magnetic tape. The same signal that modulates the radio frequency carrier is recorded on the magnetic tape (fig 2.3:3, 2.3:4, and 2.3:5) for a permanent record. After the aircraft has landed, the tape can be removed and played back through a ground station similar to a ground receiving station. Also, tape recordings of transmitted signals can be made at the ground receiving station. The output signal of the receiver is recorded on the magnetic tape (fig 2.3:3, 2.3:4, and 2.3:5) for a permanent record.




TRANSMITTING




RECEIVING

TIME - DIVISION SUB-MULTIPLEXING

Figure 2.3:5

## 2.4.1 INTRODUCTION

In addition to instrumentation used within the aircraft, there are several external instrumentation devices. The following may be used in conjunction with helicopter flight tests:

Theodolite Camera
Fairchild Photographic Flight Analyzer
Mitchell Camera
Anemometer
TRODI

The theodolite camera, Fairchild photographic flight analyzer, and Mitchell camera may be used to determine fuselage attitude, airspeed, and rates of descent and climb. The anemometer is used to measure wind velocity and the TRODI is used to measure touch-down rates of descent.

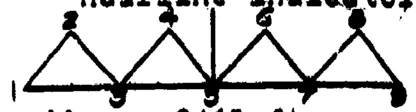
## 2.4.2 THEODOLITE CAMERA

The theodolite camera is a phototheodolite instrument that measures the azimuth, time and elevation of a point and records a picture of that point and the measurements of its azimuth and elevation on photographic film. The instrument consists of a telescope and a 35 mm motion picture camera that will rotate in both a horizontal and vertical plane. The camera is mounted so the center of the photographs continually coincides with the center of the field of view of the telescope.

Each frame records time, azimuth and elevation. Time is recorded on a clock. Azimuth and elevation are recorded on mechanical counters that include traveling hairline indicators. The counters and traveling hairline indicators record in mils. A mil is the angle subtended by an arc of 1 unit on a radius of 1000 units. Therefore, a radius of 1000 ft results in a mil equal to 1 ft, a radius of 500 ft results in a mil equal to ½ ft, etc. The theodolite camera is usually set up at a distance of 1000 ft from the flight path of the object being tracked.

The azimuth and elevation scales record to the nearest mil with the scales arranged as illustrated below:

| Azimuth or Elevation Counter | | | 1 mil traveling |
|---|---|---|---|
| 1000 digit | 100 digit | 10 digit | hairline indicator |
| 2 | 4 | 6 | |



This illustration would be read as 2465 mils or 2465 ft for a 1000 ft radius.

Each frame is notched on both sides and top and bottom. A vertical line and a horizontal line, connecting opposite notches, would intersect at the center of the frame. This point of intersection is called the notch center. When reading theodolite film, a point on the helicopter, usually the rotor head, is used as a reference point. If the notch center is not directly over the reference point, a correction to the azimuth and elevation readings must be made by measuring the horizontal and vertical distances of the reference point from the notch center and multiplying these distances by a scale factor determined from the ratio of a known measurement of the helicopter to its corresponding image size. These horizontal and vertical corrections are added or subtracted to the azimuth and elevation readings, respectively.

The corrected readings on the first frame of a continuous theodolite record are taken as zero reference points. The first frame corrected readings are subtracted from all other frame corrected readings to determine differential azimuth, elevation and time readings. Curves of differential azimuth and/or differential elevation versus differential time are plotted to permit calculations of horizontal, vertical and/or flight path velocities.

### 2.4.3 FAIRCHILD PHOTOGRAPHIC FLIGHT ANALYZER

The Fairchild photographic flight analyzer is a photogrammetric instrument which freezes the space-time record of an aircraft, moving in a path parallel to the focal plane of the analyzer, as a series of photographs of the aircraft on a single plate. In addition to recording vertical and horizontal movement, the analyzer records the time at each instant of exposure by photographing on the same plate the face of a timer capable of measuring to 1/1000 of a sec. The recording of time increments permits the calculation of velocity and acceleration characteristics.

A test flight recorded by the analyzer appears as 58 strip exposures on a single 8" x 10" plate, each strip containing a single image of the target. The analyzer records the targets position each time it has moved a fixed distance and not in accordance with predetermined time intervals. This results in equal spacing of points along the length of the plate, as opposed to a grouping of points at the slow speed portions of a run and a scarcity of points along the high speed portions, as would be the case if the analyzer were time controlled.

The flight analyzer tracks a moving object without camera rotation; however, the use of a wide-angle lens makes it possible to track a moving object and cover a large range. Since the camera does not rotate, the parallel relationship between the photographic plate and the flight course is unchanged; therefore, there is no image distortion or change in image size due to the changing distance between the target and the camera as the target passes the camera.

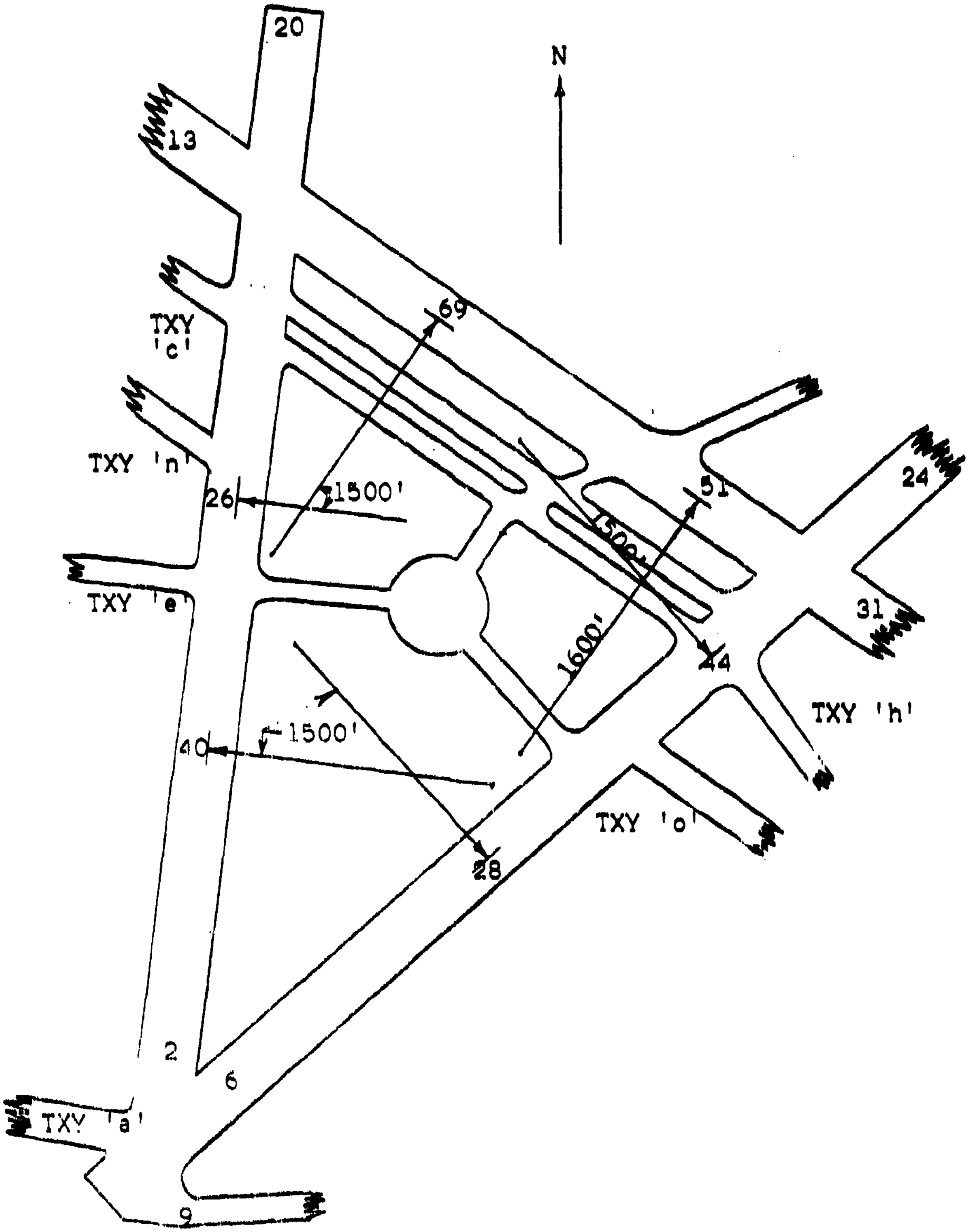
When the analyzer plate is developed, contact paper reproductions or enlarge paper reproductions are made and are used for data reduction.

When recorded flights are made, the flight analyzer is positioned on a spot 1500 ft from the centerline of the runway being used as a flight course. The analyzer set-up locations have been accurately determined with the use of surveying instruments and are indicated on the diagram shown in figure 2.4:1. The blacktop on the field is marked with paint at these locations.

With a distance of 1500 ft between the analyzer and the flight course, each frame is equal to 37.5 ft. Time is registered on counters and traveling reeds. The counters register sec, tenths of a sec, and hundredths of a sec from bottom to top. The traveling reed, similar to the traveling hairline indicator of a theodolite camera, registers thousandths of a sec. Knowing the distance scale and the time at each exposure, one can calculate the horizontal, vertical and flight path velocities."

### 2.4.4 MITCHELL CAMERA

The Mitchell camera is an electrically driven 35 mm motion picture camera which, in addition to photographing the helicopter, also photographs the dial of a precision timer on each frame. The camera is mounted in a fixed




FAIRCHILD ANALYZER STATIONS

Figure 2.4:1

position on a line perpendicular to the line of flight to the helicopter.

To read the film from a Mitchell camera, a scale factor is determined by comparing a known measurement of the helicopter to its corresponding image distance. This image distance is measured when the helicopter passes directly in front of the camera placing the helicopter in the center of the frame. Horizontal and vertical distances are measured from vertical and horizontal stationary reference lines, respectively, and are multiplied by the scale factor. Knowing differences in distance and differences in time from one frame to another, one can calculate vertical, horizontal and flight path velocities.

### 2.4.5 ANEMOMETER

The hand-held anemometer is used to measure wind velocities during hover tests and to pace aircraft from the ground in a moving vehicle for test flights flown at low airspeeds. The instrument is read directly in kt. Two scales are used, one ranging from 0 to 60 kt; another actuated by depressing the red button on the handle, ranging from 0 to 15 kt.

### 2.4.6 TRODI

The TRODI is an electronic optical instrument used to measure the vertical rate of descent of an aircraft just prior to touchdown on a landing surface. The major components consist of a detector, an indicator and two trihedral prisms. The detector emits a fan of light, receives reflected light from the trihedral prisms that are mounted externally on the aircraft being tested and transfers the optical reflection into an electronic signal. The indicator receives the electronic signal and transforms it into a meter reading indicating the vertical rate of descent of the aircraft at touchdown. The rate of descent is immediately available at the time of touchdown. The TRODI is used to measure vertical touchdown rates of descent when tests involving hard landings are made.

# AIRSPEED POSITION ERROR CALIBRATION

## Section 2.5

### 2.5.1 INTRODUCTION

One or more flights are made to determine an airspeed position calibration. This is done by comparing, by means of a graph, indicated airspeed versus calibrated airspeed; or in the case of an oscillograph, trace deflection versus calibrated airspeed. Calibrated airspeed may be determined using one or more of the following methods:

- Theodolite Camera
- Fairchild Photographic Flight Analyzer
- Mitchell Camera
- Trailing Bomb
- Fishing Point Altimeter Depression
- Aircraft Pace
- Anemometer Pace
- Measured Distance vs Time

Calibrated airspeed using the theodolite camera, Fairchild photographic flight analyzer, or Mitchell camera is determined using the methods previously described and taking into account the component of wind along the flight path.

### 2.5.2 TRAILING BOMB METHOD

A trailing bomb consists of a bomb-shaped airspeed sensitive differential pressure pickup and a cable approximately 150 ft long connecting the pickup and an airspeed indicator usually located in a photopanel. The trailing bomb is hung from the helicopter away from the turbulent air flow caused by the rotating blades and the airframe. Records are taken at various intervals of predetermined airspeeds; the readings of the trailing bomb indicator are compared graphically to the readings of the system being calibrated. Caution is of prime importance during take-off and landing to allow adequate ground clearance when using the trailing bomb method of airspeed position error calibration.

### 2.5.3 FISHING POINT ALTIMETER DEPRESSION METHOD

The altimeter depression method of determining calibrated airspeed compares the true pressure altitude, as determined from the true elevation of the helicopter as it passes over a fixed point, with the pressure altitude as indicated by an altimeter in the helicopter. The difference between the two pressure altitudes is the position error in terms of an altitude increment which can be expressed as an airspeed increment. The airspeed increment is then added algebraically to the indicated airspeed to obtain the calibrated airspeed.

Altimeter depression calibrations are made by flying the helicopter between two markers and past a sentry tower located on Fishing Point course, shown in figure 2.5:1. Prior to the fly-bys, the pilot taxis the helicopter to the red spot located in the northeast corner of the Flight Test apron, sets the pressure altimeter to a pressure reading of 29.92 in of Hg, faces the helicopter into the wind and records time and indicated altitude. The helicopter is flown by the sentry tower on the prescribed course at different airspeeds as required for a complete calibration. During each fly-by, the pilot attempts to hold airspeed and altitude constant. When the helicopter reaches a position directly in front of the sentry tower, the helicopter's indicated altitude and airspeed are either noted by the pilot or recorded on the photopanel. After the completion of the runs, the test helicopter is again taxied to the red spot on the Flight Test apron where time and indicated altitude are again recorded.

Two ground observers are required for an altimeter depression calibration, one in the sentry tower and one at the Fishing Point marker. A time history of pressure altitude is taken by the sentry tower observer during the tower runs. A sensitive altimeter can be obtained from the Instrumentation Branch for this purpose. The instrument should be tapped lightly prior to each reading.

When the helicopter reaches a position in front of the sentry tower, the tower observer sights on the helicopter and records the sighting stand reading (distance N in figure 2.5:2) and the time of the run. A sample data card is shown in figure 2.5:3. To insure correct alignment of the sighting stand, the tower observer should check that it is level and that a white house across the river is seen when sighting through it.

The observer at the Fishing Point marker aligns himself with the flight path of the helicopter and estimates the lateral displacement of the helicopter from the marker

FISHING POINT COURSE
NATC
PATUXENT RIVER, MD

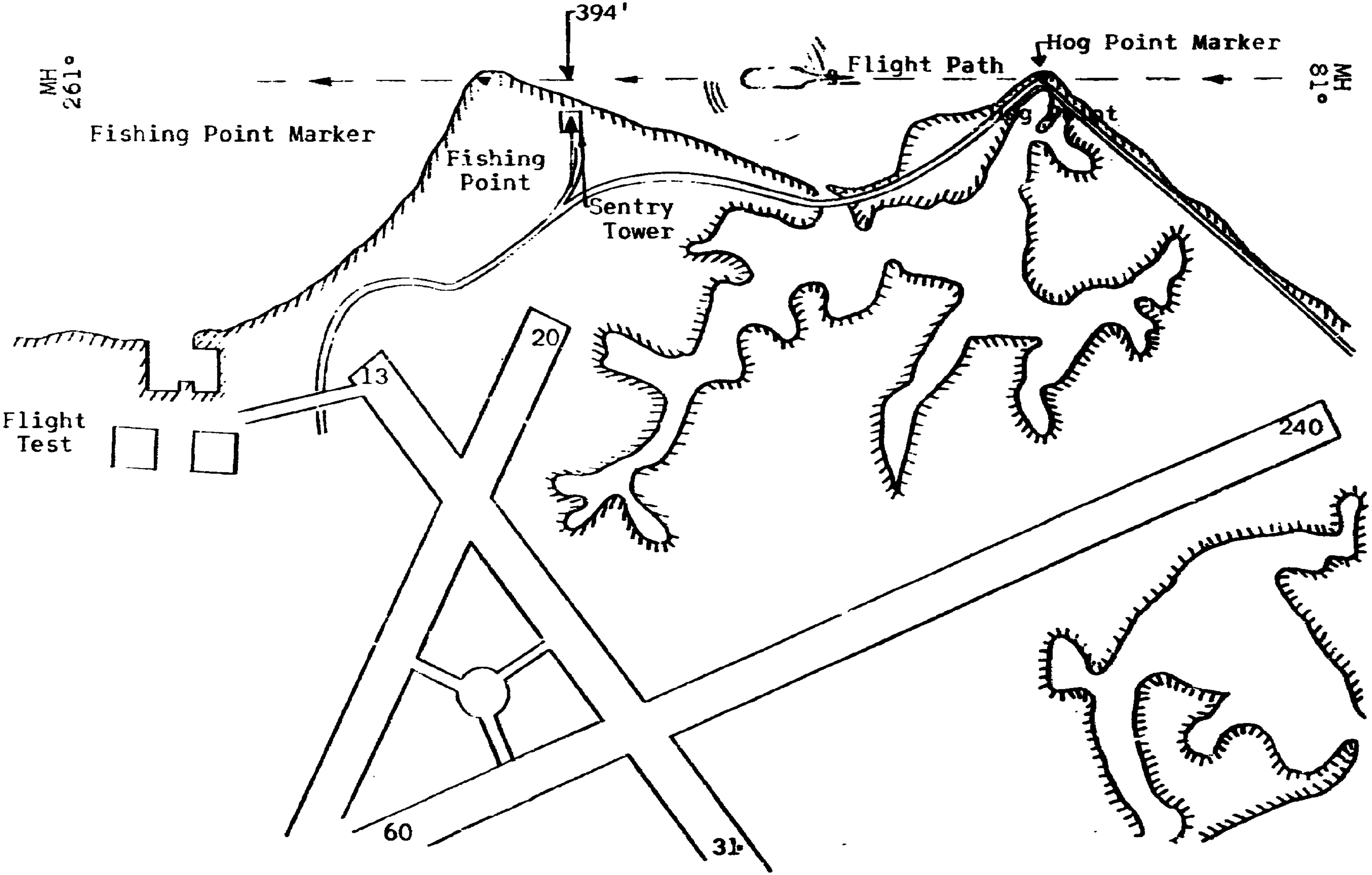
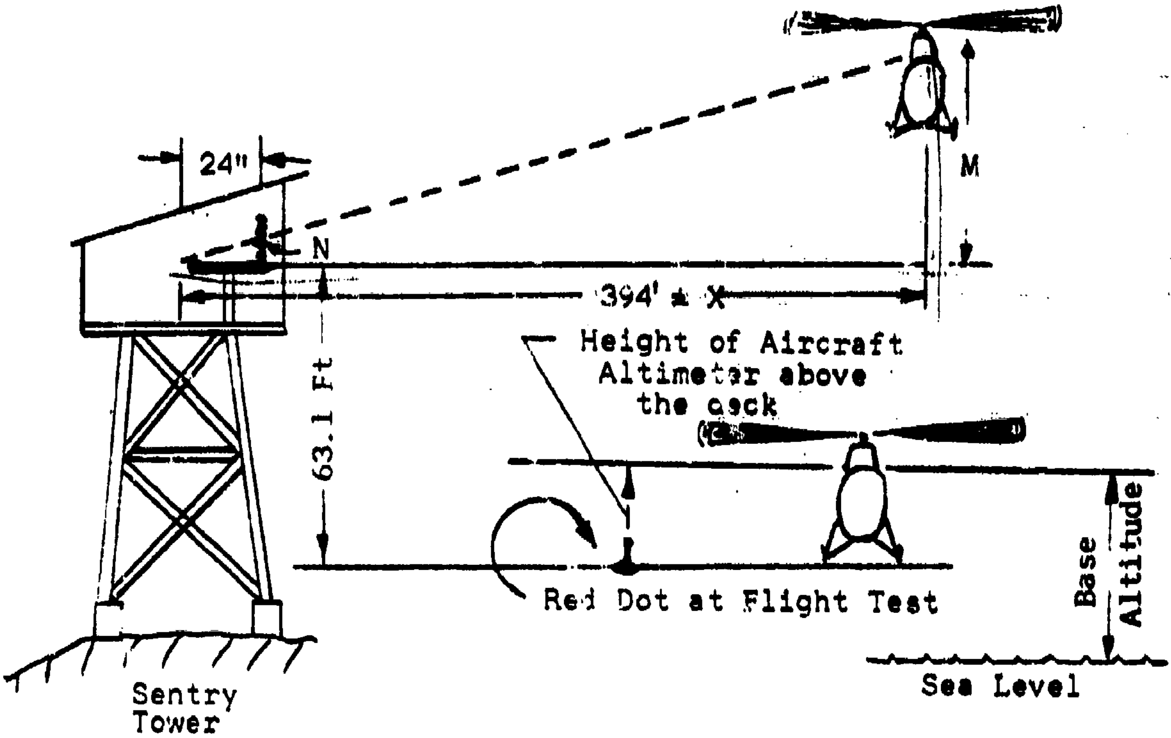


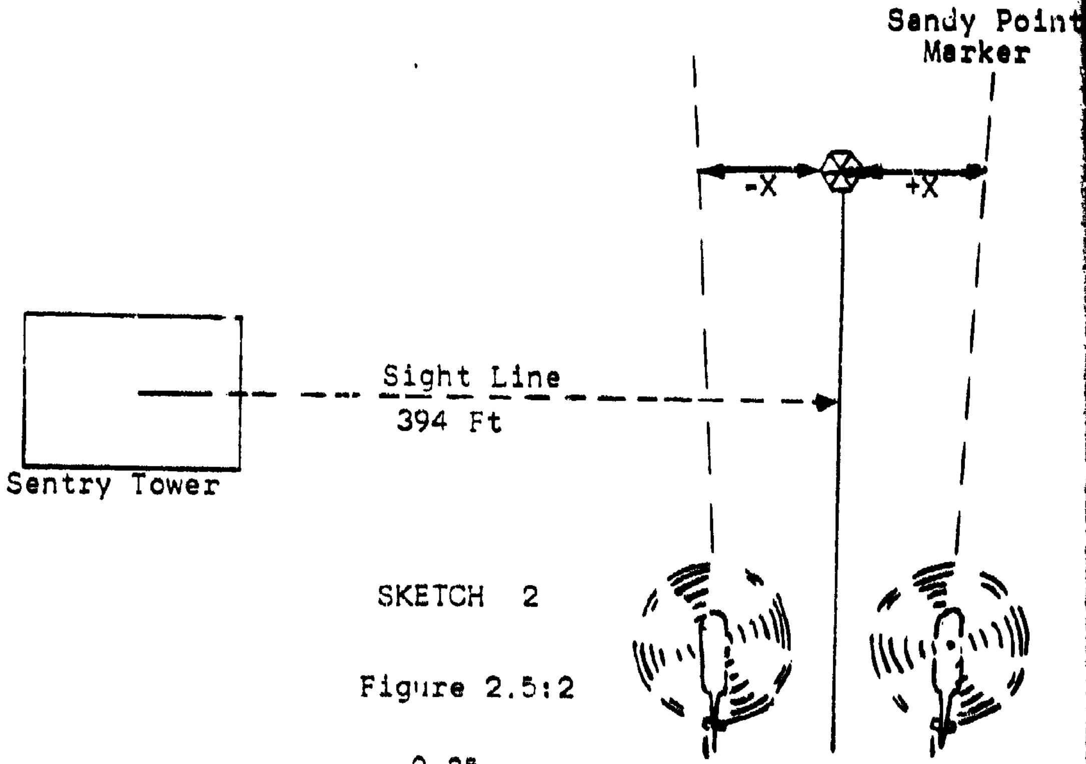

Figure 2.5:1

SKETCHES OF DISTANCES

RECORDED AT FISHING POINT TOWER




SKETCH 1




SKETCH 2

Figure 2.5:2

CARD NO 1b

TOWER OBSERVERS CARD

AIRPLANE TYPE BU NO

CONFIG:

CONDITION OBSERVER

| Run No | Time | ALT ind | N | |
|---|---|---|---|---|
| | | | | |
| | | | | |
| | | | | |
| | | | | |
| | | | | |
| | | | | |
| | | | | |
| | | | | |
| | | | | |

CARD NO 1c

FISHING POINT OBSERVERS CARD

AIRPLANE TYPE BU NO

CONFIG:

CONDITION OBSERVER

| Run No | X (+OR-) | | | |
|---|---|---|---|---|
| | | | | |
| | | | | |
| | | | | |
| | | | | |
| | | | | |
| | | | | |
| - X | MEASURED | TOWARD | TOWER | |
| + X | MEASURED | TOWARD | RIVER | |
| | | | | |

Figure 2.5:3

(distance x in figure 2.5:2). Sample data cards for ground observers are shown in fig 2.5:3.

An altimeter depression sample data reduction sheet is shown in figure 2.5:4. Columns 1 and 13, the indicated airspeed and indicated altitude, respectively, are values which are either pilot or photo-recorded. Columns 2 and 3 change indicated airspeed in MPH to indicated airspeed in kt corrected for instrument error. The base altitude of the sentry tower, recorded in column 5, is the altitude of the initial helicopter ground run plus or minus an incremental pressure. The incremental pressure is the tower pressure altitude at each fly-by minus the tower pressure altitude at the time of the helicopter's initial ground run. The base altitude values are determined in the following manner:

a. Plot a time history of the pressure altitude (tabulated in column 4) recorded at the sentry tower.

b. Plot the initial and final ground run altimeter readings on the same graph.

c. Draw time lines at the time of each calibration run. As shown in figure 2.5:5, the base altitude for run 1 is the first ground run reading plus $P_1$. The base altitude for run 2 is the first ground reading plus $P_2$, etc.

The geometrical height of the sentry tower instrumentation above the red dot on the Flight Test apron is a constant value of 63.1 ft and is recorded in column 6. The height of the helicopter altimeter above the deck when the helicopter is on the deck is a constant that varies with each different helicopter and is recorded in column 7. The distances, x, recorded by the observer stationed at Fishing Point marker are listed in column 8. Column 9 is 394 ft plus or minus x, depending on whether the helicopter is outboard or inboard, respectively, of Fishing Point marker. The sighting stand readings, N, are recorded in column 10. In column 11, the distances, M, that the helicopter was above the tower are recorded and are calculated by a simple proportion:

$$\frac{M}{N} = \frac{394 \pm x}{24} \qquad\qquad M = \frac{394 \pm x}{24} \cdot N$$

The true altitudes of the helicopter are recorded in column 12 and are found by subtracting the height of the helicopter altimeter above the deck, column 7, from the sum of the base altitude, column 5, the geometrical height of the tower, column 6, and the height of the helicopter above the tower,

## SAMPLE DATA REDUCTION SHEET FOR FISHING POINT POSITION ERROR DETERMINATIONS

| | | |
|---|---|---|
| 1 | Vi mph | READ FROM PHOTOPANEL OR PILOT RECORDED |
| 2 | Vi* mph | CORRECT (1) FOR INSTRUMENT ERROR |
| 3 | Vi* Kt | CHANGE (2) TO Kt |
| 4 | Tower Altimeter Read. | RECORDED BY TOWER OBSERVER |
| 5 | Base Alt. | DETERMINED FROM TIME HISTORY OF PRESSURE ALTITUDE PLOT |
| 6 | Height of Tower | CONSTANT VALUE OF 63.1 Ft |
| 7 | A/C Altimeter Height | HEIGHT OF AIRCRAFT ALTIMETER ABOVE THE DECK |
| 8 | X Ft | DISTANCE FROM AIRCRAFT FLIGHT PATH TO FISHING POINT MARKER |
| 9 | 394 ± X Ft | IF TEST AIRCRAFT IS INBOARD OF COURSE FLIGHT PATH, SUBTRACT X<br>IF TEST AIRCRAFT IS OUTBOARD OF COURSE FLIGHT PATH, ADD X |
| 10 | N Ft | SIGHTING STAND READING RECORDED BY TOWER OBSERVER |
| 11 | M Ft | HEIGHT OF AIRCRAFT ABOVE SIGHTING STAND |
| 12 | True Alt. Ft | COLUMNS (5) + (6) + (11) - (7) |
| 13 | Ind. Alt. Ft | READ FROM PHOTOPANEL OR PILOT RECORDED |
| 14 | $\Delta H$ Ft | COLUMN (12) - (13) |
| 15 | $\Delta V$ Kt | FROM $\Delta H$ - $\Delta V$ CHARTS |

Figure 2.5:4

SAMPLE PLOT TO DETERMINE

BASE ALTITUDE

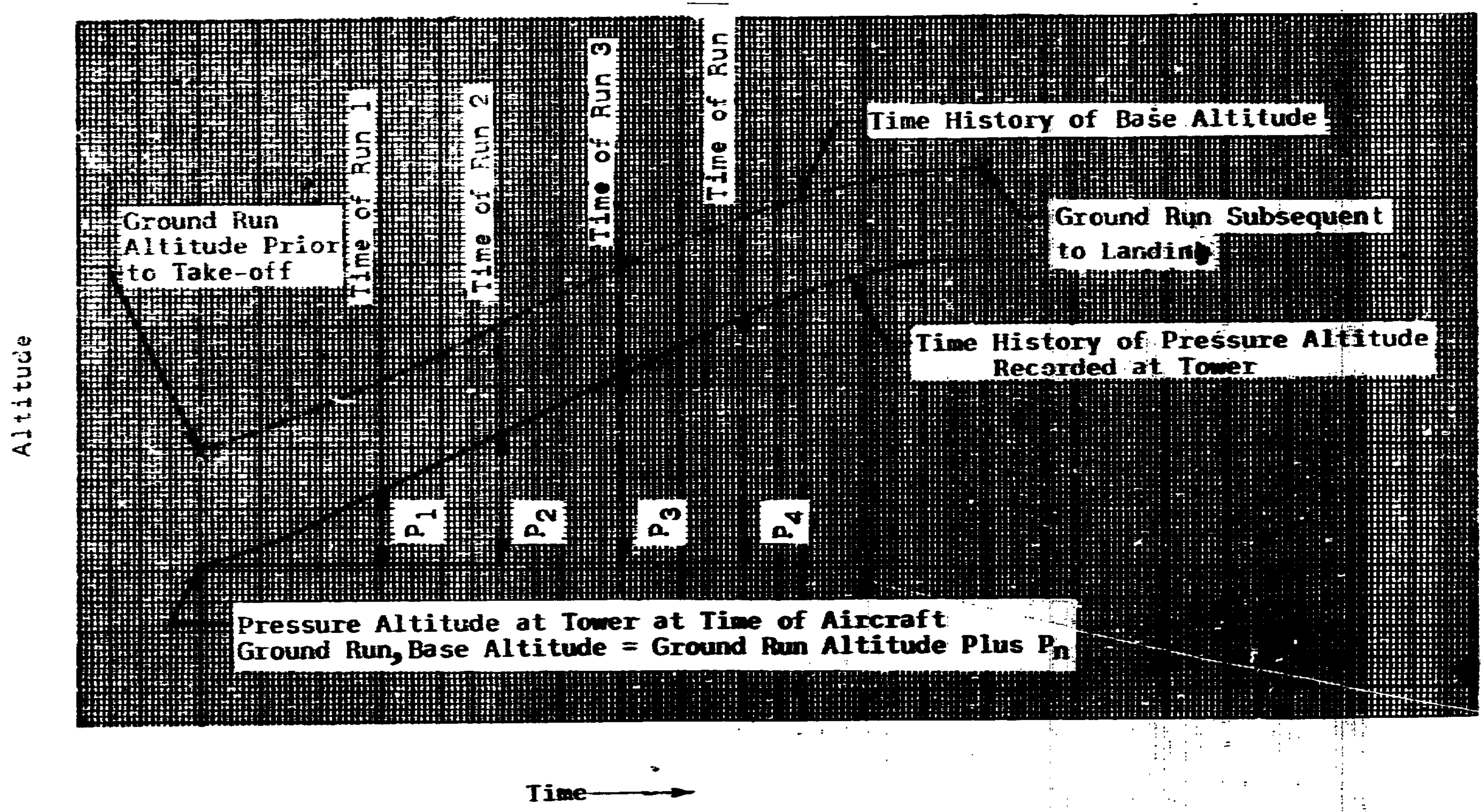



Figure 2.5.5

column 11 (column 12 = columns 5 + 6 + 11 - 7). The differences between true altitude and indicated altitude, column 12 minus column 13, are recorded in column 14. Using the appropriate chart (depending on the sign of $\Delta H$) in figure 2.5:6 or 2.5:7, $\Delta V$ in kt is determined using values from column 3, $V_{ic}$ kt, and column 14, $\Delta H$ in ft. Enter the right hand side of the appropriate chart and read up the $\Delta H$ line to the point where it intersects the sea level altitude line. Read across horizontally to the appropriate indicated airspeed line on the left hand side of the graph. Drop vertically to the $\Delta V$ scale and record $\Delta V$ in column 15. The airspeed increment, $\Delta V$, is added algebraically to the indicated airspeed, corrected for instrument error, to obtain the calibrated airspeed.

### 2.5.4 PACE METHOD

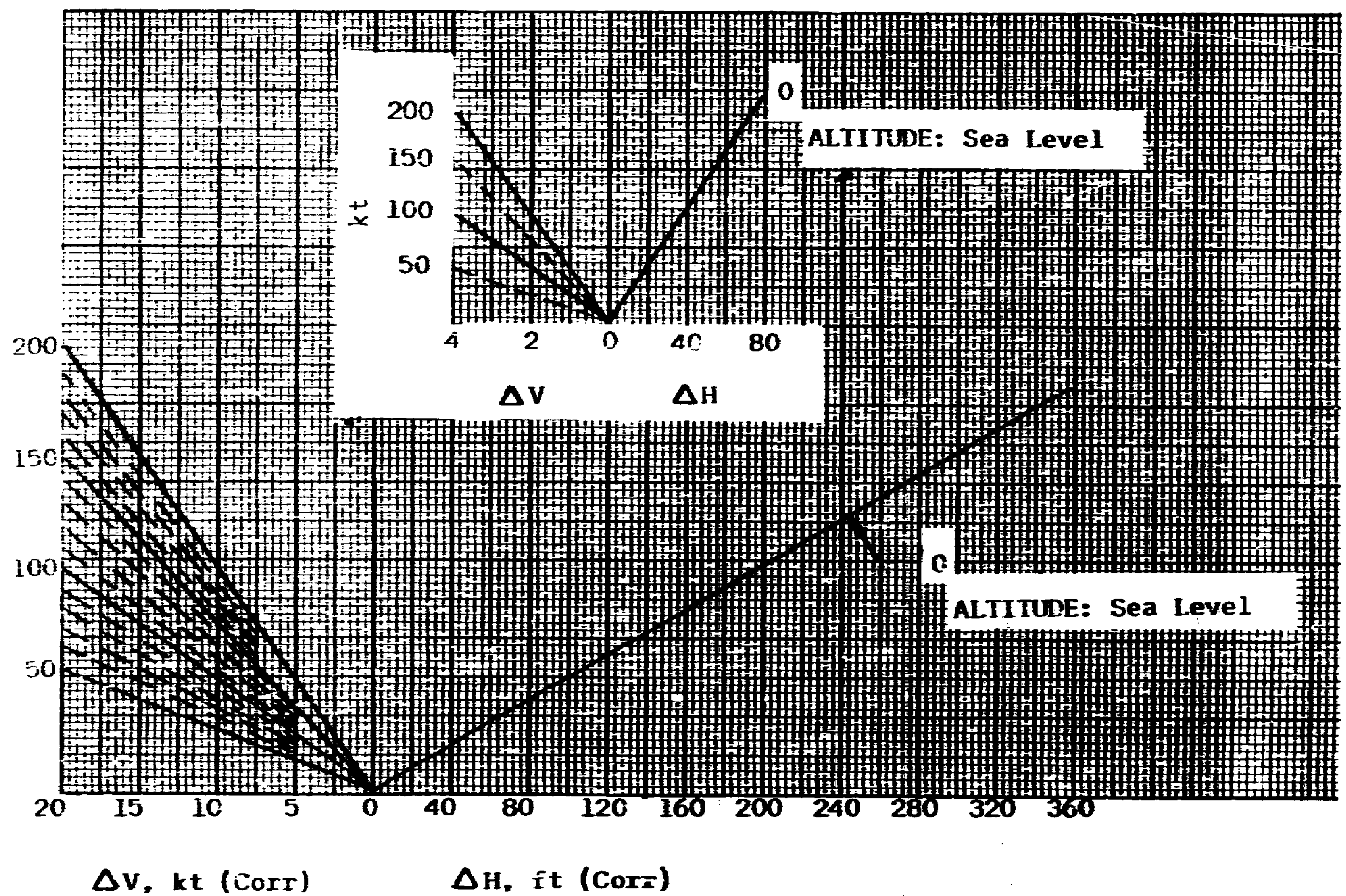
Calibrated airspeed may be determined by pacing the aircraft with another aircraft that has an airspeed indicator already calibrated for position error. An aircraft may also be paced by a ground vehicle using an anemometer. Pacing with another aircraft is usually done in conjunction with large high speed helicopters; pacing with a ground vehicle is usually done in conjunction with small low speed helicopters and all size helicopters in rearward and sideward flight.

### 2.5.5 MEASURED DISTANCE VS TIME METHOD

Measuring the time a helicopter takes to fly over a known distance is another way to obtain calibrated airspeed. The NATC runways are marked every hundred ft and present an accurate measurement of distance. Knowing the distance, time and component of wind along the flight path; the calibrated airspeed can be determined. Preferably, this type of calibration should be conducted in light winds (under 5 kt).

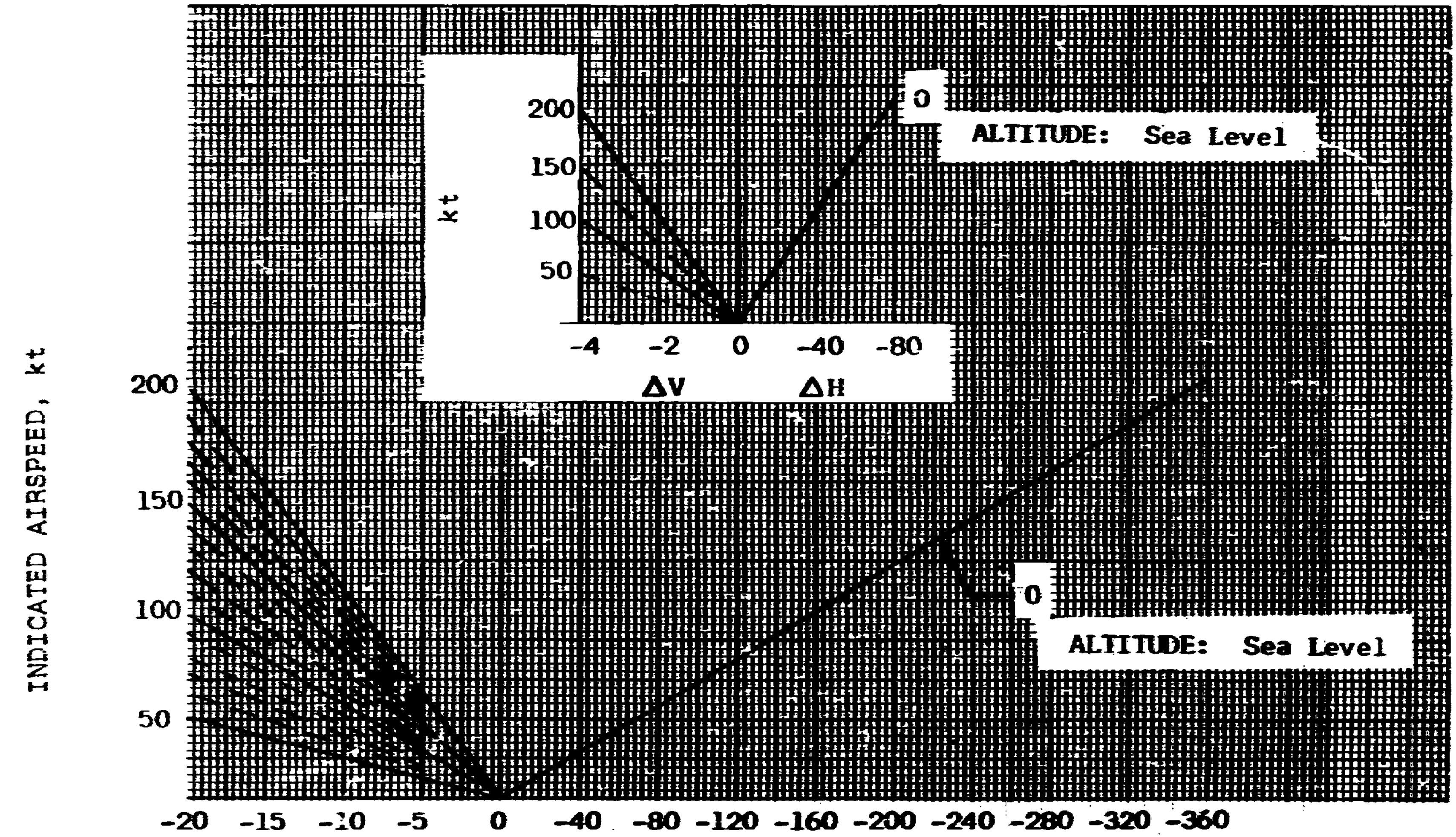
CORRECTION CHART FOR STATIC SOURCE ERROR IN AIRSPEED INDICATOR & ALTIMETER

(Positive Altitude and Airspeed Correction-Aircraft Altimeter Reading Less than True Pressure Altitude)



CORRECTION CHART FOR STATIC SOURCE ERROR IN AIRSPEED INDICATOR & ALTIMETER

(Negative Altitude and Airspeed Correction - Aircraft Altimeter Reading Greater Than True Pressure Altitude)




ΔV, kt (Corr)

ΔH, ft (Corr)

Figure 2.5:7

DATA ANALYSIS SECTION

Section 2.6

## 2.6.1 GENERAL

When project personnel have data reduction that is extensive enough for computer reduction, the Rotary Wing project engineer informs the Data Analysis Section of the information that is desired. With this information and with the necessary calibration curves, the Data Analysis Section sets up a prepared program on paper tape. From the raw data, the Data Analysis Section prepares another paper tape to be fed into the computer along with the prepared program. The computer used is the Datatron located in the Armament Test Division. The Datatron produces a tabulation of reduced data and/or a paper tape containing reduced data.

In addition to the computer services available in the Data Analysis Section, there are also Oscars, Boscars, and Electroplotters, all manufactured by the Benson-Lehner Corporation. The Oscar and/or the Boscar can be used to read raw data. The output of the Oscar or Boscar can be used to record raw data in tabulated form, prepare a paper tape to be fed into the Datatron, activate the Electroplotter to plot a graph of the raw data, or any combination of these. Also, paper tape containing reduced data from the Datatron can be used to activate the Electroplotter to plot a graph of reduced data.

## HOVER

## Section 3.1

### 3.1.1 INTRODUCTION AND PURPOSE

Hover flight is unique to the helicopter permitting a variety of missions utilizing this characteristic such as plane guard, ASW, rescue and etc. For the purpose of this manual the word hover applies to a helicopter that is airborne at a given altitude over a fixed point on the ground regardless of wind. "Pure hover" is when the wind velocity is zero. In order to determine the hover characteristics of a helicopter, flight tests are conducted both in ground effect (IGE) and out of ground effect (OGE) under a variety of winds usually 0-20 kt and maximum and minimum gross weights. To determine the exact dividing line between IGE and OGE, hover tests are conducted using a weighted rope of various lengths and individual tests are performed at each altitude. As the helicopter ascends from one altitude to the next the horsepower required to hover will increase (discounting weight change as a result of using fuel) until the OGE point is reached. At this point the power required to hover will remain constant providing wind velocity, RPM, and weight are not varied. Usually the OGE point is approximately one rotor diameter above the ground under zero wind conditions. As the wind velocity increases the OGE point will be lowered or, in other words the "ground cushion" effect is being blown out from under the helicopter.

Hover data are usually presented in dimensionless parameter form ($C_T$ vs $C_P^{2/3}$) at various Z/R values or IGE and OGE. The method of obtaining the data and reduction of data is presented below together with sample charts. At present, lack of accurate instrumentation prevents the helicopter from attaining a true hover at altitude, thus a true hover ceiling is impractical to attain or determine. To circumvent this, OGE hover data are utilized to determine values of power required at various altitudes, RPM's and gross weights. The intersection between the power required curves and the power available curves (determined from service ceiling climbs) are the hover ceilings. Substantiation of employing the sea level OGE hover data ($C_T$ vs $C_P^{2/3}$) has been accomplished by performing hover tests at various altitudes using stations on a mountain for ground reference.

Hover data obtained in various wind velocities up to 20 kt are used to complete the level flight power required charts, since present airspeed systems are inaccurate and stabilized flight is impractical in these regimes of flight.

### 3.1.2 METHOD AND TECHNIQUE

Flight testing is a coordinated effort of pilot, engineer, instrumentation and maintenance personnel. Prior to performing hover tests the engineer and pilot together fill out the flight card. A typical flight card is shown in figure 3.1:1.

The pilot should be briefed on engine limits with respect to MAP, RPM and BMEP. Proper hand signals should be discussed in order for the pilot to know whether he is high or low when attempting to obtain a test point. The exact weight and balance information should be listed on the reverse side of the flight data card, as well as the increments of weight to be removed and the location of such weight. Also, the starting and ending counter number shown on the oscillograph and photopanel should be clearly listed on the card by the engineer. There is a place at the top of the flight card for recording humidity information. This data should be obtained from Aerology after the flight at the closest time to the flight or the information should be determined by the engineer using a sling psychrometer for each particular test point.

In performing the hover tests at any particular altitude as previously mentioned a weighted rope is usually attached inside the helicopter so that in case of emergency it can be released by the pilot or observer. Once the line is secured it can be extended to its full length perpendicular to the helicopter. This is a safety measure to prevent the line from blowing up into the rotor system. It also allows the pilot to keep the weight in view until it is almost off the deck. The engineer or observer should be stationed slightly to one side (pilots) of the aircraft and give the standard LSO signals (hi, low, Roger) in order to convey to the pilot the exact hover height. A variation of 1-2 ft of the weight on and off the deck should be allowed by the LSO or else he will create considerable pilot effort trying to hold such a close tolerance on hover. The hover should be done into the wind (except for a tandem where cross wind hover may be desirable) with the maximum of landing area in front of the helicopter. In case of emergency this will allow maximum area in which to land the helicopter and since

WEATHER Wet Bulb
Dry Bulb
Baro
Time

CARD NO. 12

PTR DIS AC-4037

| AIRPLANE TYPE | BU. NO. | TIME T.O. / TIME LAND | DATE |
|---|---|---|---|
| H-23D | 57-2987 | | 3/15/58 |

PILOT: Major Gill

T.O.C.G.

GEAR DOWN % UP %

CONDITION: Skid Conf.

T.O. GROSS WEIGHT: 2700

EXTERNAL CONFIGURATION

Hover (zero wind) IGE OGE

Red Line

Mixture automatic MAP 27.3 @ 3200 RPM

| CTR | RPM | MAP | OAT | ALT | Fuel | | Ballast |
|---|---|---|---|---|---|---|---|
| | 3200 | | | | | | |
| | 3100 | | | Skid Clear | | | |
| | 3000 | | | | | | |
| | 2900 | | | | | | |
| | | | | | | | |
| | 3200 | | | | | | |
| | 3100 | | | 10 ft Skid Clear | | | |
| | 3000 | | | | | | |
| | 2900 | | | | | | |
| | | | | | | | |
| | 3200 | | | | | | |
| | 3100 | | | 20 ft Skid Clear | | | |
| | 3000 | | | | | | |
| | 2900 | | | | | | |
| | | | | | | | |
| | 3200 | | | | | | |
| | 3100 | | | 60 ft Skid Clear | | | |
| | 3000 | | | | | | |
| | 2900 | | | | | | |

FLIGHT DATA-PRNC-NATC-512 (Rev. 6-51)

TYPICAL DATA CARD

Figure 3.1:1

the engineer is slightly to one side he will also have a better chance of getting out of the way. This maneuver is performed in the "dead man's curve area" and if a co-pilot is along a pre-arranged course of action on his part in event of engine failure will prevent confusion in an emergency.

If pure hover is being performed, the maximum allowable wind for this condition is 3 kt. Vertical climb tests are usually done concurrent with the pure hover tests as this atmospheric condition is not normal and may occur only a couple times a month. Vertical climb performance is discussed in section 3.2.

In attaining any particular data point the pilot will be consistently adjusting the collective stick and throttle until he is completely stable. He should attempt to remain at this point for approximately 30 sec (pure hover) prior to taking a 3 sec burst on the instrumentation. The stabilization time will reduce with increase in wind velocity as a result of gusts. When the pilot is taking a record an external light will be triggered simultaneously informing ground personnel that a record is being taken. The engineer taking the surface wind with a hand held anemometer will record the wind upon seeing the light.

### 3.1.3 DATA REDUCTION PROCEDURE

The efficiency of a helicopter rotor may be expressed as the minimum power required to hover to the actual power required to hover or $M = \frac{Tv}{P}$ (1) where M is the rotor figure of merit. Since from helicopter theory the induced power $v = \sqrt{\frac{T}{2\rho\pi R^2}}$ (2) then $M = \frac{1}{\sqrt{2}}\frac{T}{P}$ [illegible] $R^2$ (3)

To obtain non-dimensional parameters and using coefficients similar to propellor coefficients where

$$T = C_T \pi R^2 \rho (\Omega R)^2$$

$$Q = C_Q \pi R^2 \rho (\Omega R)^2 R \quad (4)$$

$$P = C_P \pi R^2 \rho (\Omega R)^3$$

and $C_Q = C_P$ by

$$C_Q = \frac{Q}{\pi R^2 \rho (\Omega R)^2 R} \times \frac{\Omega}{\Omega} = \frac{P}{\pi R^2 \rho (\Omega R)^3} = C_P$$

By substitution of equations (4) in equation (3) we obtain

$$M = 0.707 \frac{C_T^{3/2}}{C_Q = C_P}$$

A plot of $C_T$ vs $C_P^{2/3}$ is a straight line and when $M = 1$

$$.707\, C_T^{3/2} = C_P$$

$$\text{or } C_P^{2/3} = .794\, C_T$$

This presents a convenient way of presenting hover data and comparison between the ideal figure of merit.

The method of reducing hover flight test data are shown in figure 3.1:2. The three constants $K_{C_T}$, $K_{C_P}$ and $K_h$ are constant for any particular helicopter and are valid for all tests. The term $K_{C_T}$ is a thrust coefficient constant and is equal to $\frac{(\pi R^4)(.1047)^2 \rho_o}{(GR)^2}$ where (GR) is the gear ratio between engine and rotor. $K_{C_P}$ is the power coefficient constant and is equal to $\frac{K_{C_T}\, R(.1047)}{GR(550)}$. The humidity factor $K_h$ is obtained from figure 3.1:3. The data sheet shown in figure 3.1:2 is based on the use of the engine manufacturer's power chart. If torquemeters are used columns 11 and 16 are omitted and in column 17 will be placed the engine power based on torquemeters. The power losses to obtain rotor shaft power will be as shown unless an efficiency factor is used.

Hover ceiling data reduction procedure is shown in figure 3.1:4. The $C_T$ vs $C_P^{2/3}$ pure hover dimensionless curve is used to obtain the hover ceiling. By establishing the desired parameters of weight, RPM and density $C_T$ can be determined for any particular altitude. Entering the zero wind $C_T$ vs $C_P^{2/3}$ curve with the previously determined $C_T$, $C_P^{2/3}$ is established. Knowing $C_P^{2/3}$, engine horsepower is determined. The above procedure is repeated for different altitudes and gross weights.

Card ____________ $K_{CT}$ =
Observers Sheet $K_{CP}$ =
$K_h$ =

## HOVER

Date Test of

| | | | | | | | | | | | |
|---|---|---|---|---|---|---|---|---|---|---|---|
| CTR No. | | | | | | | | | | | |
| ① * $H_p$ | | | | | | | | | | | |
| ② * MAP | | | | | | | | | | | |
| ③ * CAT | | | | | | | | | | | |
| ④ * OAT | | | | | | | | | | | |
| ⑤ $\rho/\rho_0$ * | | | | | | | | | | | |
| ⑥ * RPM eng. | | | | | | | | | | | |
| ⑦ $RPM^2$ eng. | | | | | | | | | | | |
| ⑧ $RPM^3$ eng. | | | | | | | | | | | |
| ⑨ W | | | | | | | | | | | |
| ⑩ $C_T$ $\frac{⑨ \times ⑤}{K_{CT} \times ⑦}$ | | | | | | | | | | | |
| ⑪ $BHP_{ch}$ ①②⑥ | | | | | | | | | | | |
| ⑫ $T_{as}$ | | | | | | | | | | | |

* Corrected for instrument error where $C_T = \frac{W \rho/\rho_0}{(RPM_e)^2 (K_{CT})}$

$C_P = \frac{BHP_{rs} \rho/\rho_0}{K_{CP} (RPM_e)^3}$

HOVER PERFORMANCE DATA REDUCTION

Figure 3.1:2.1

Card __________ **H O V E R** (Cont')

Project Number Project Observer

| | | | | | | | | | | | |
|---|---|---|---|---|---|---|---|---|---|---|---|
| 13 $T_{co}$ | | | | | | | | | | | |
| 14 $\sqrt{\rho/\rho_o}$ | | | | | | | | | | | |
| 15 $\sqrt{T_{as}/T_{co}}$ | | | | | | | | | | | |
| 16 $BHP_o$ 11 x 15 | | | | | | | | | | | |
| 17 $BHP_{oh}$ 16 x $K_h$ | | | | | | | | | | | |
| 18 $HP_{Fan}$ | | | | | | | | | | | |
| 19 $HP_{Tail}$ | | | | | | | | | | | |
| 20 $HP_{Trans}$ | | | | | | | | | | | |
| 21 HP Total Loss | | | | | | | | | | | |
| $BHP_{rs}$ 17 - 21 | | | | | | | | | | | |
| $C_p$ | | | | | | | | | | | |
| $C_p^{2/3}$ | | | | | | | | | | | |
| Wind (Kt) | | | | | | | | | | | |

Figure 3.1:2.2

## CHART FOR DETERMINATION OF HUMIDITY FACTOR, $K_h$

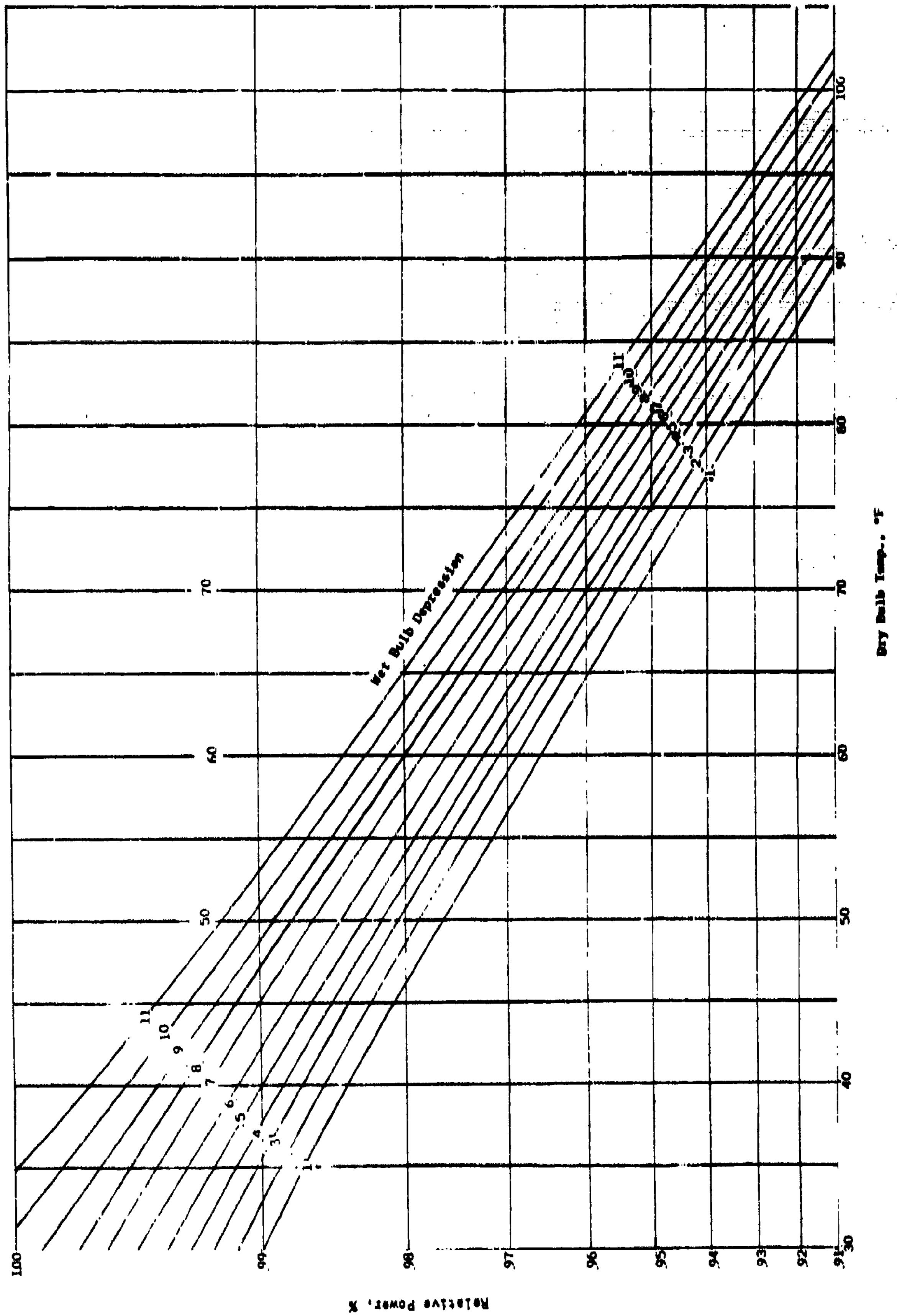



Figure 3.1:3

$$C_T = \frac{W \rho_0/\rho}{K_{CT}(RPM_e)^2}$$

$$C_P = \frac{BHP_{rs} \rho_0/\rho}{K_{CP}(RPM_e)^3}$$

HOVER CEILING

Gross Weight ______ lbs Aircraft ______

| $H_s$ | -2000 | -1000 | 0 | 1000 | 2000 | 3000 | 4000 | 5000 |
|---|---|---|---|---|---|---|---|---|
| RPM | | | | | | | | |
| $RPM^2 \times 10^3$ | | | | | | | | |
| $RPM^3 \times 10^7$ | | | | | | | | |
| WIND | | | | | | | | |
| W | | | | | | | | |
| $\rho_0/\rho$ | | | | | | | | |
| $C_T$ | | | | | | | | |
| $C_P$ | | | | | | | | |
| $C_P^{2/3}$ | | | | | | | | |
| $BHP_{rs}$ | | | | | | | | |
| $HP_{Fan}$ | | | | | | | | |
| $HP_{Tail}$ | | | | | | | | |
| $BHP_e$ | | | | | | | | |

HOVER CEILING DETERMINATION

Figure 3.1:4

### 3.1.4 TYPICAL RESULTS

A typical plot of $C_T$ vs $C_P^{2/3}$ is shown in figure 3.1:5. From this data with known values of gross weight, power available, RPM, wind and density the hover ceiling is determined. Data reduction for hover ceiling is shown on figure 3.1:4. A typical plot of the hover ceiling is shown in figure 3.1:6. The chart showing the effects of $Z/_R$ hover height above the ground is shown in figure 3.1:7.

Model HRS-1 Helicopter
BuNo 127783

HOVER PERFORMANCE
IN GROUND EFFECT

Thrust Coefficient
vs
Rotor-Shaft Power Coefficient

Z/R = 1
(Z=26.5 ft)

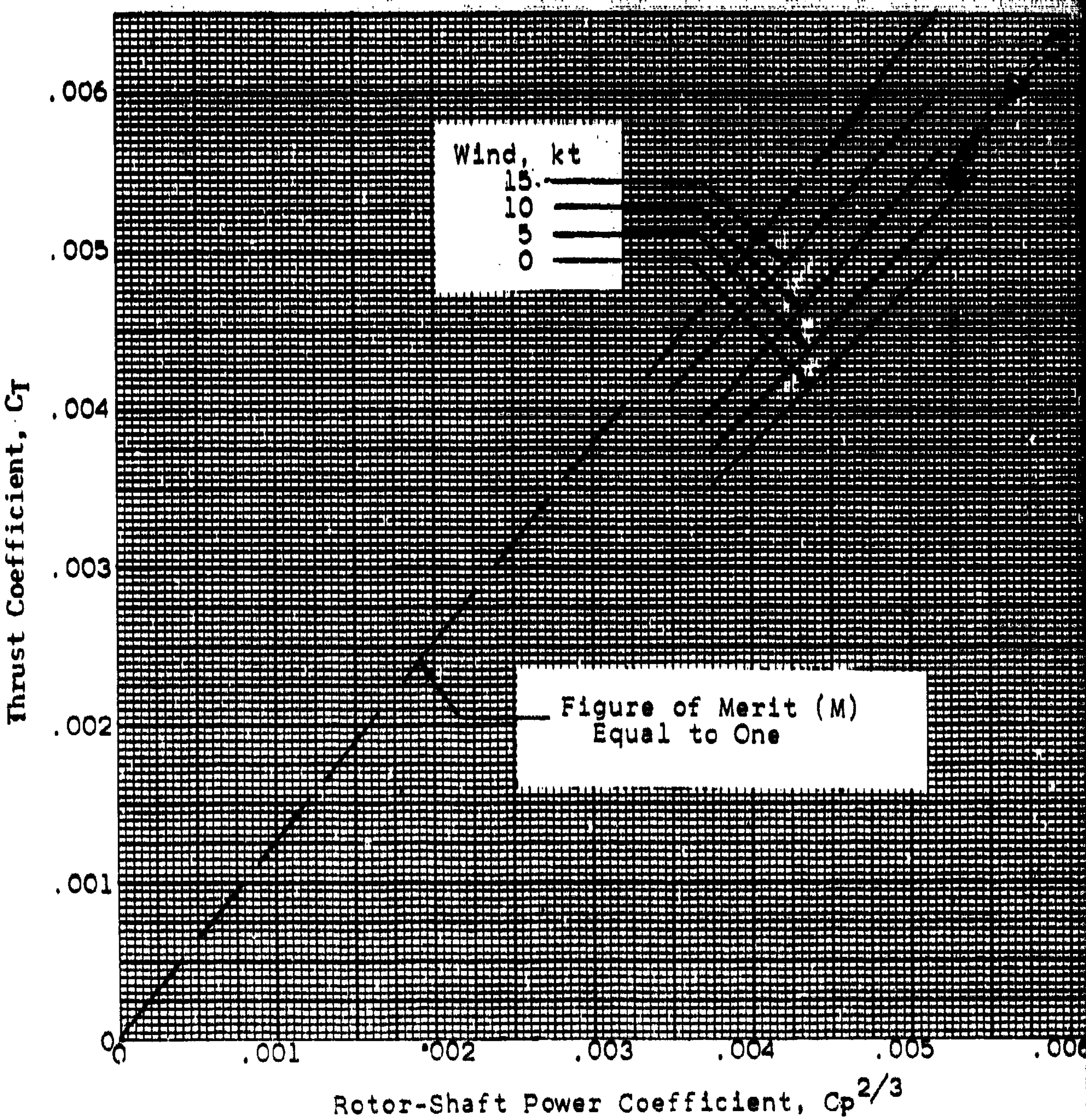
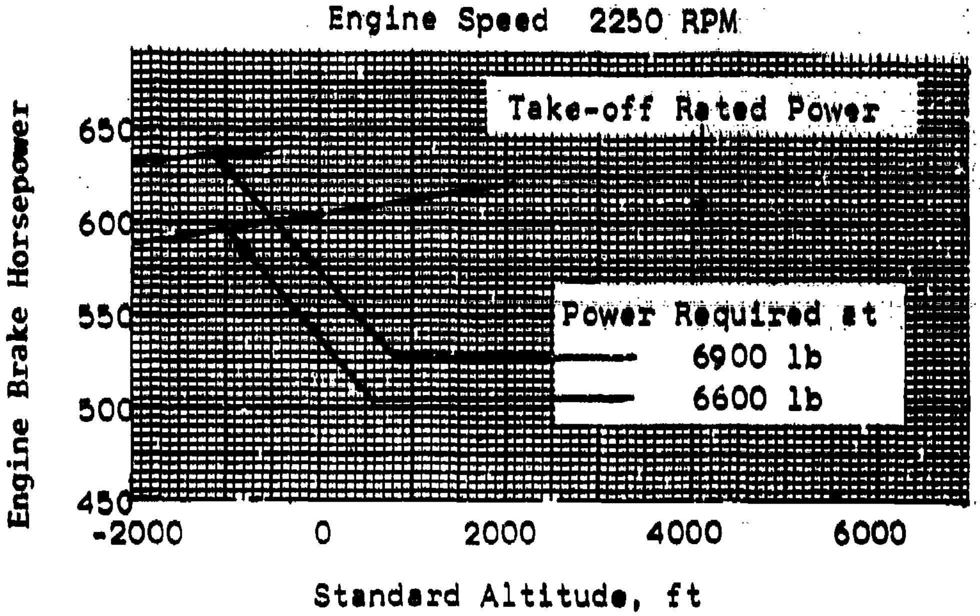



Figure 3.1:5

Model HRS-1 Helicopter
BuNo 127783

HOVER PERFORMANCE
OUT OF GROUND EFFECT



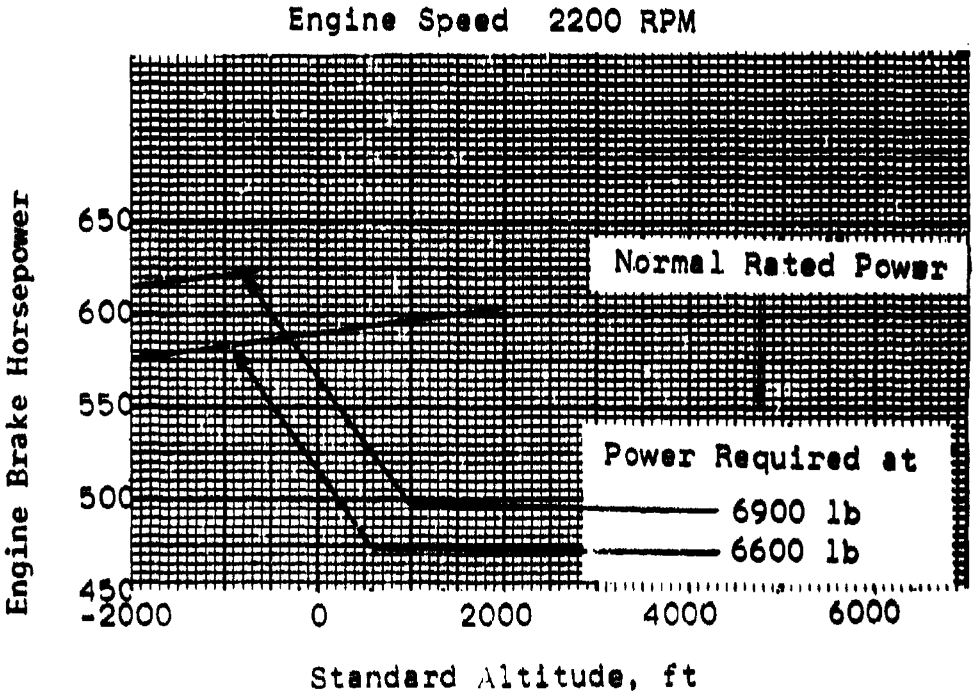




Figure 3.1:6

Model HRS-1 Helicopter
BuNo 127783

HOVER PERFORMANCE
IN GROUND EFFECT

Rotor Hub Height Above Ground
vs
Brake Horsepower Required

Altitude Sea Level
Gross Weight 7200 lb
Engine Speed 2200 RPM

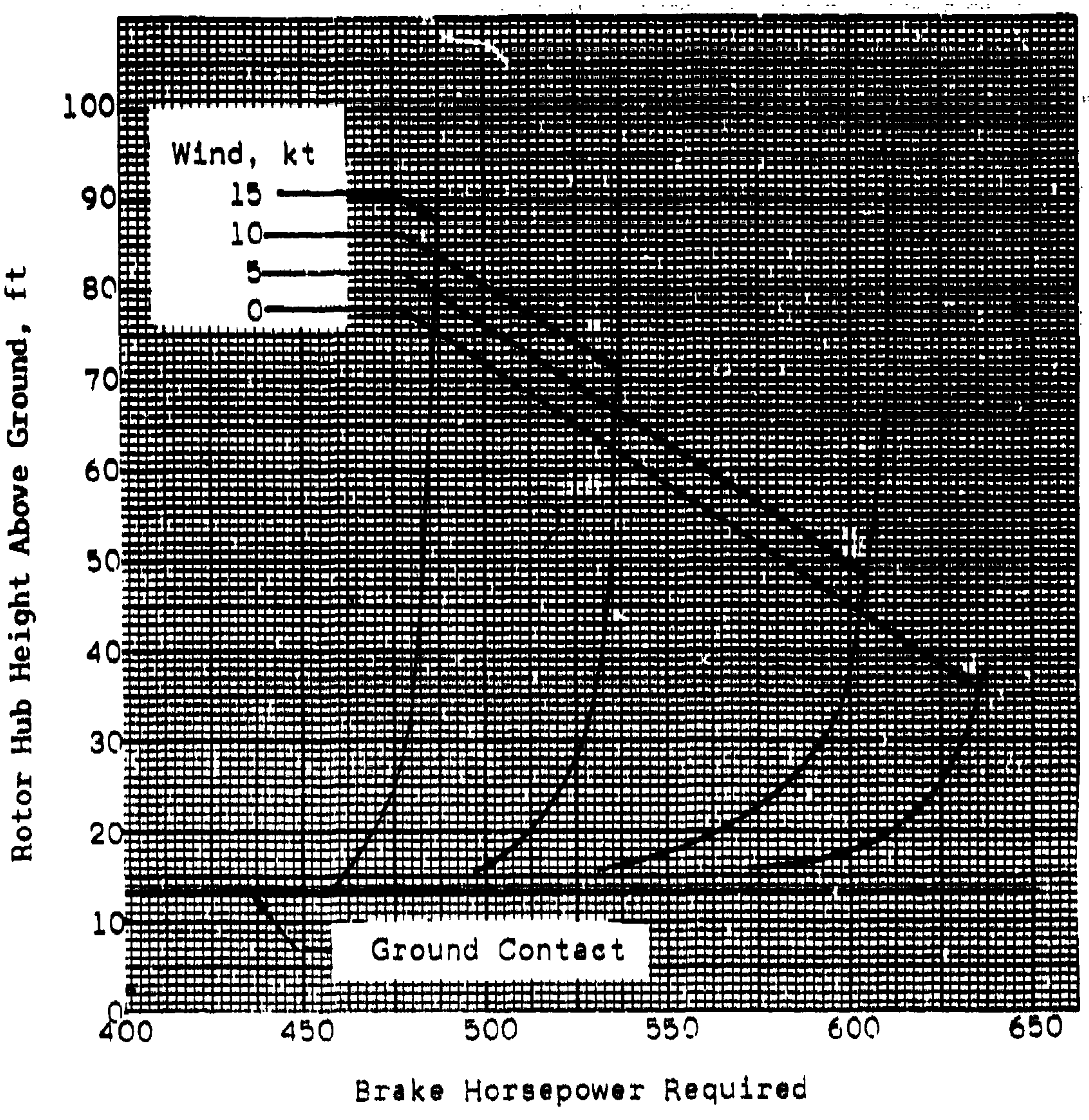



Figure 3.1:7

VERTICAL CLIMB

Section 3.2

## 3.2.1 INTRODUCTION AND PURPOSE

Many operational missions require an aircraft which has the capability of rising and descending vertically. The helicopter has this characteristic providing enough excess power is available. Many of the earlier helicopters were under powered, or the weight growth as a result of alterations in design and mission, preventing the aircraft from having any appreciable vertical climb capability. A vertical rate of climb of 300 ft/min is considered a minimum requirement for a helicopter, since for higher than standard temperature operation, climb capability will be considerably reduced as a result of decreased power available.

## 3.2.2 METHOD AND TECHNIQUE

Vertical climbs are conducted usually concurrent with hover flights under zero wind conditions. Maximum winds tolerated for conducting vertical climbs are three kt and even then the data is questionable as a result of increased wind velocity with altitude. Usually climbs are started from a hover IGE and conducted from sea level to 1000 ft pressure altitude with 3 sec instrumentation bursts taken at each hundred feet. Once the desired RPM is obtained in a hover and the helicopter stabilized, the throttle is increased to obtain the desired MAP, utilizing collective pitch to maintain proper RPM. A positive coordinated action of the throttle and collective is necessary to obtain the required MAP and RPM as quickly as possible, thus obtaining a stabilized climb as quickly as possible. Constant power is held until the climb is terminated. If the helicopter has a very high rate of climb a continuous instrumentation record is taken. Climbs are usually conducted at several weights and powers to obtain an adequate spread in the data. Data are also recorded on a flight card by a co-pilot or passenger if available utilizing cockpit instrumentation as shown in figure 3.2:1. The pilot's flight card which is made out jointly by the pilot and engineer contains a MAP schedule for the pilot to fly. The engineer should note on the card the amount of weight to be taken out and from what loading station the weight will be removed after each vertical climb. After each climb it is also a good idea to

WEATHER Time
Wet Bulb
Dry Bulb
Baro

CARD NO. 2

PTR BIS

AIRPLANE TYPE HRS-1 | BU. NO. 127783 | TIME T.O. | TIME LAND | DATE

PILOT Major Anderson | T.O.C.G. | GEAR DOWN % UP %

CONDITION Photo CTR
Oscil CTR

T.O. GROSS WEIGHT Vary

EXTERNAL CONFIGURATION

Vertical Rate of Climb

Normal Mixture

| CTR | RPM | MAP | ALT | OAT | CAT | FUEL | Time |
|---|---|---|---|---|---|---|---|
| | 2250 | MAP varies with increased alt. to maintain constant MP | 0 | | | | WT 1 |
| | | | 100 | | | | |
| | | | 200 | | | | |
| | | | 300 | | | | |
| | | | 400 | | | | |
| | | | 500 | | | | |
| | | | 600 | | | | |
| | | | 700 | | | | |
| | | | 800 | | | | |
| | | | 900 | | | | |
| | | | 1000 | | | | |
| | | | | | | | |
| | 2200 | | 0 | | | | WT 2 |
| | | | 100 | | | | |
| | | | 200 | | | | |
| | | | 300 | | | | |
| | | | 400 | | | | |
| | | | 500 | | | | |
| | | | 600 | | | | |
| | | | 700 | | | | |
| | | | 800 | | | | |

FLIGHT DATA—PRNC-NATC-312 (Rev. 6-51)

PILOT'S FLIGHT CARD

Figure 3.2:1

spot check the instrumentation to see if it's functioning properly and if sufficient film and oscillograph footage is available for the next climb.

In the first few climbs performed it may be difficult to keep the helicopter climbing vertically. It is very easy for the pilot to drift forward or backward, either instance will give poor data. To assist in maintaining a vertical climb the pilot should pick a reference point at some distance in front of the aircraft and shift his gaze from this to some point that is perpendicular to the helicopter. In doing this the heading can be maintained and the reference will show if the aircraft is drifting fore or aft. As altitude is gained throttle adjustments will have to be made or at least until full throttle is attained. Sometimes in performing a vertical climb in the above manner the pilot will sense an increase in wind velocity as altitude is gained even though the surface wind is calm. Adjustments are then made by drifting with the wind. A Column of smoke if available will give an excellent guide as to the wind velocity from the deck to 1000 ft pressure altitude.

It is of interest to point out that the difference between power available and power required for steady level flight at any selected airspeed will not give a reasonably accurate vertical rate of climb value for the helicopter as in the case of fixed wing aircraft. That power difference available for climb is substantially increased when conducting vertical climbs. The explanation for this phenoenon lies in the fact that rotor efficiency increases with air mass flow through the disc. The relative change of air mass flow through the disc in climb as compared with flow in level flight becomes larger as airspeed is reduced to zero.

### 3.2.3 DATA REDUCTION PROCEDURE

The method of reducing vertical climb data are shown in figure 3.2:2. The determination of the constants $K_{C_T}$, $K_{C_P}$ and $K_h$ are discussed in paragraph 3.1.3. The zero curve of $\frac{V_V}{\Omega R}$ shown in figure 3.2:3, where $V_V$ is vertical velocity and $\Omega R$ is rotational velocity is obtained from the zero wind hover plot of $C_T$ vs $C_P^{2/3}$ of figure 3.1:3. Columns (18) through (21) are omitted if torquemeters are used and the torquemeter column added either for rotor shaft power or engine power or both dependent on the location of

# VERTICAL CLIMB

Observers Sheet

Dry Bulb
Wet Bulb
Bar.Press.
$K_h$

Date Test of

| CTR No. | | | | | | | | | | |
|---|---|---|---|---|---|---|---|---|---|---|
| 1 $H_p$ | | | | | | | | | | |
| 2 $\Delta$ TIME (Sec) | | | | | | | | | | |
| 3 $R/C_o$ 1 vs 2 | Determined from slope of 1 vs 2 | | | | | | | | | |
| 4* OAT | | | | | | | | | | |
| 5* CAT | | | | | | | | | | |
| 6* MAP | | | | | | | | | | |
| 7* RPM | | | | | | | | | | |
| 8 $RPM^2$ | | | | | | | | | | |
| 9 $RPM^3$ | | | | | | | | | | |
| 10 $T_{as}$ | | | | | | | | | | |
| 11 $T_{ao}$ | | | | | | | | | | |
| 12 $T_{co}$ | | | | | | | | | | |
| 13 $\frac{T_{ao}}{T_{as}}$ | | | | | | | | | | |
| 14 $R/C_T$ 3 x 13 | | | | | | | | | | |

Figure 3.2:2.1

VERTICAL CLIMB (Cont')

Project Number                Project                Observers

| | | | | | | | | | | |
|---|---|---|---|---|---|---|---|---|---|---|
| 15 * RPM Rotor | | | | | | | | | | |
| 16 $\Omega R$ $\frac{15}{9.55} \times R$ | | | | | | | | | | |
| 17 $V_c/\Omega R$ 14 ÷ 16 | | | | | | | | | | |
| 18 BHP Chart 1, 6, 7 | | | | | | | | | | |
| 19 $\sqrt{\frac{T_{as}}{T_{co}}}$ | | | | | | | | | | |
| 20 $BHP_o$ 18 x 19 | | | | | | | | | | |
| 21 $BHP_{oh}$ 20 x $K_h$ | | | | | | | | | | |
| 22 $\sigma/\sigma_o$ | | | | | | | | | | |
| 23 $L_o$ Fan Loss | | | | | | | | | | |
| 24 $L_c$ 23 x 22 | | | | | | | | | | |
| 25 HP Tail Loss | | | | | | | | | | |
| 26 HP Trans. Loss | | | | | | | | | | |
| 27 HP Total 24 + 25 + 26 | | | | | | | | | | |

Figure 3.2:2.2

Observers Sheet

Dry Bulb
Wet Bulb
Bar.Press.
$K_h$

Project Number Project Observers

| | | | | | | | | | | |
|---|---|---|---|---|---|---|---|---|---|---|
| 28 $BHP_{rs}$ 21 - 27 | | | | | | | | | | |
| 29 W | | | | | | | | | | |
| 30 $C_T$ $\frac{29 \times 22}{K_{CT} \times 8}$ | | | | | | | | | | |
| 31 $C_P$ $\frac{28 \times 22}{K_{CP} \times 9}$ | | | | | | | | | | |

Figure 3.2:2.3

Model HRS-1 Helicopter
BuNo 127783

VERTICAL CLIMB PERFORMANCE

**Thrust Coefficient vs**

**Main-Rotor-Shaft Power Coefficient**

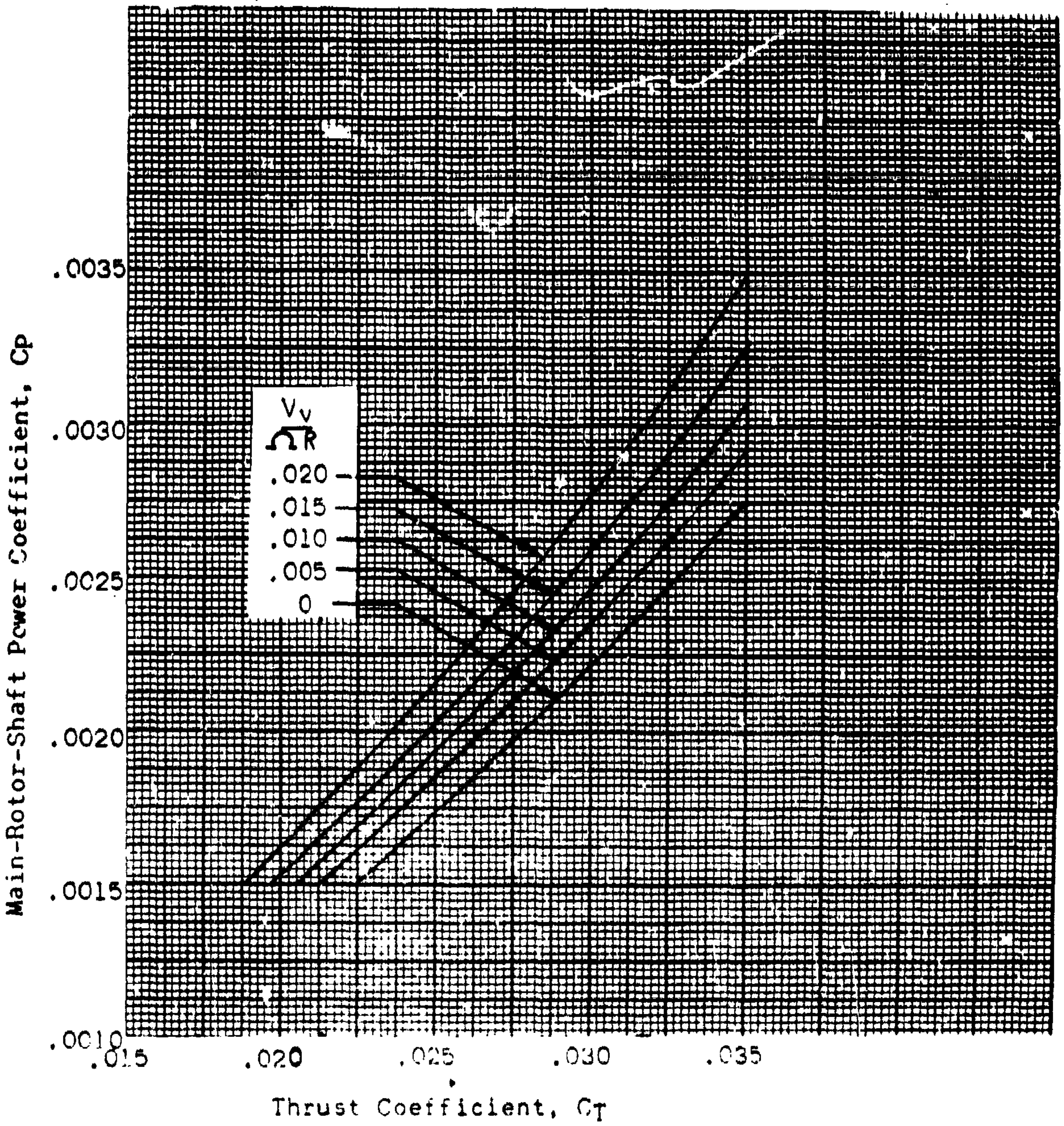



Figure 3.2:3

the torquemeters. Normally if torquemeters are used they will be positioned on the main and tail rotor drive shafts as it is impractical to locate a torquemeter on the engine shaft because of the fan location.

### 3.2.4 TYPICAL RESULTS

Once the curve of figure 3.2:3 is established from the data of figure 3.2:2 a cross plot for any desired gross weight and RPM condition is made in accordance with figure 3.2:4 and plotted as shown in figure 3.2:5.

## CROSS PLOT - VERTICAL CLIMB

Engine RPM ______ Gross Weight ______

(1) ALT 0 - 1000 ft in 100' increments

(2) - for (1) above

(3) $C_T$

(4) $BHP_e$ - Constant

(5) - Efficiency factor - (Determined from losses)

(6) (5) x (4) = $BHP_{rs}$

(7) $C_P$ - from (6) and (2) above

(8) $V_V/\Omega R$ - from graph

(9) $V_V$ ft/sec (knowing $\Omega R$ - k($RPM_e$); (8) x $\Omega R$

(10) $V_V$ ft/min

Plot (10) vs (1) for particular GW repeat above for other GW or horsepowers.

Figure 3.2:4

VERTICAL CLIMB PERFORMANCE
AT
TAKE-OFF RATED POWER SETTINGS

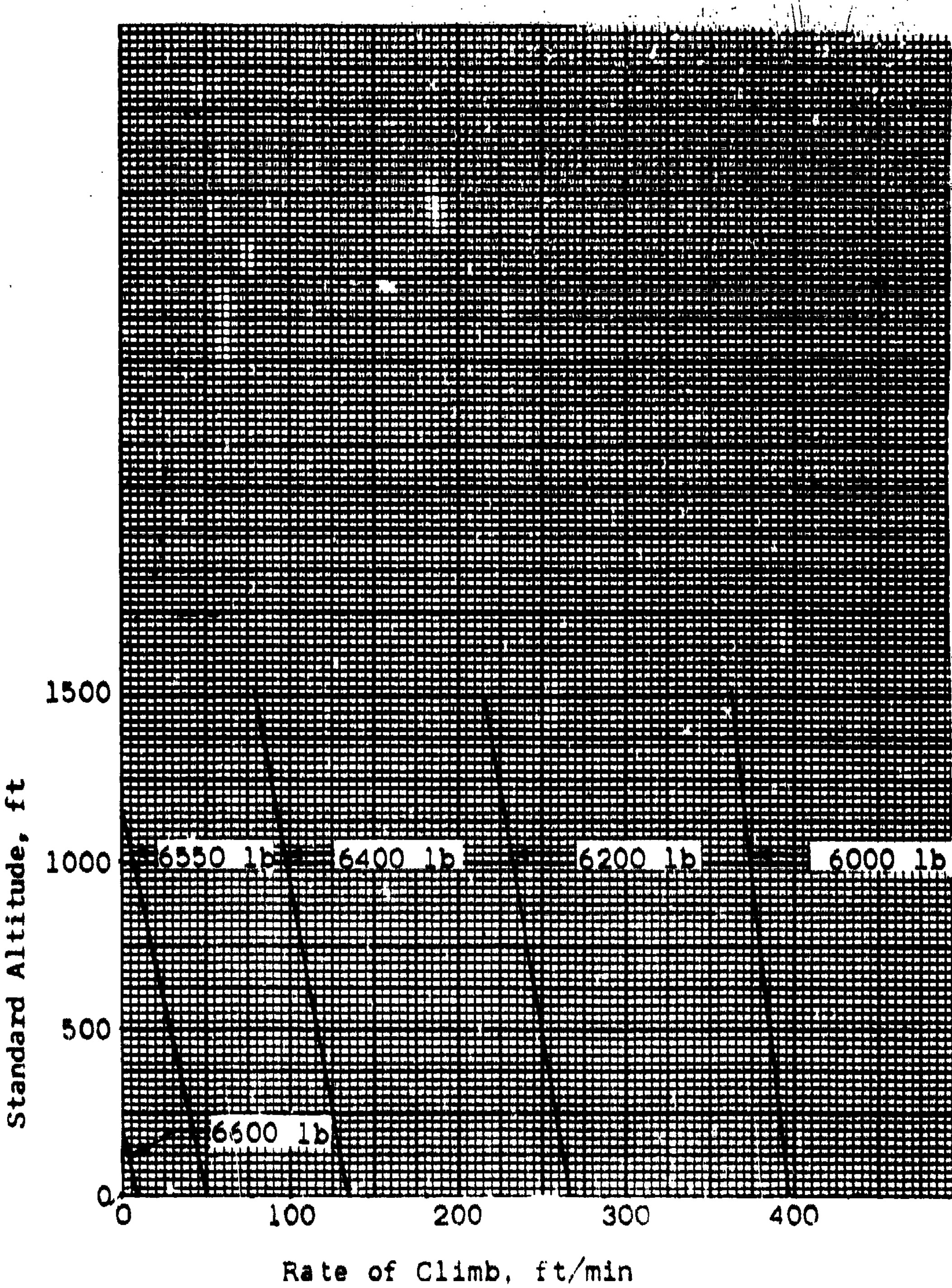



Figure 3.2:5

LEVEL FLIGHT

Section 3.3

## 3.3.1 INTRODUCTION AND PURPOSE

Level flight performance analysis of a helicopter like those applied to the airplane are based essentially on curves of power required against speed for various flight conditions. Dependent on the mission generally the highest percentage of flight time of the helicopter is in the level flight phase and involves short haul operation (approximately 2-3 hr flight duration). The power expended by the helicopter includes rotor induced and blade profile power losses, as well as parasite drag of the fuselage, rotor hub, tail rotor and potential and kinetic energy power changes. Considering stabilized level flight condition, potential and kinetic energy changes are zero and basicly the main rotor shaft power required involves rotor power losses and parasite power losses in propelling the helicopter at any particular airspeed. Rotor power losses may be reduced by selection of the best airfoil for the mission and having and maintaining a smooth rotor blade. Parasite power losses can be held to a minimum by fuselage and rotor hub design. The consideration of an airfoil section should include the type of mission and general anticipated operating speeds and gross weights or disc loading. Production and economical considerations in many instances dictate the type of blade used even though it may not be the best from a mission consideration. Design criteria for a rotor blade of desirable aerodynamic characteristics usually includes an airfoil that has high stall angle and high critical Mach number with a delayed rise in the drag divergence curve. Also the blade section should have zero pitching moment, low drag throughout the range of low and moderate lifts and moderate drag at high lifts. Further consideration of this matter is discussed in the book "Aerodynamics of the Helicopter" by Gessow & Myers and many other NASA publications on the subject.

Maximum speed restriction of a helicopter is usually a result of a limitation of power, blade stall or controllability. It is desirable from a pilot's standpoint that the limitation be on power available rather than controllability and blade stall. Blade stall is usually observed by the pilot as an increase in fuselage vibration or erratic motion of the cyclic stick. In some helicopters, normal aircraft vibration and powered control systems camouflages blade stall and optimum performance is not attained without a device to inform the pilot of the onset of blade stall. Blade stall occurs whenever the indicated airspeed of the tip of the retreating blade falls below a predetermined value. This value in the HRS-1 helicopter for example is 215 kt at a gross weight of 7500 lb and a load factor of unity. Since the velocity at the tip of the rotor is 348 kt at 2400 RPM engine speed, stall would not occur until a forward speed of 133 kt is attained. However the forward speed at which stall occurs decreases approximately 15 kt for each 100 RPM decrease in engine speed or each 1000 lb increase in gross weight and/or load factor. A temperature increase of 15°F(8°C) will have the same affect as an increase of 1000 ft of altitude. Blade stall generally does not present any undue hazard as the vibratory warning is felt either in the control system or fuselage. However, during maneuvering flight such as pull-ups or high angle of bank turns and during mountainous terrain operation severe blade stall may be encountered without ample warning. In such cases, generally, recovery can be accomplished by reduction of collective pitch and/or increasing rotor speed.

### 3.3.2 METHOD AND TECHNIQUE

Level flight power required is usually performed at various weights, RPM's, altitudes and at approximate mid CG position. The pilot's flight card shown in figure 3.3:1 contains the desired speed range to investigate as well as a note as to the limit MAP. The pilot should note on his card during the tests what Vmax is limited by (blade stall, power or controllability). The pilot should also make a statement either in his flight report or flight card as to the air conditions for the flight (smooth, slightly turbulent, etc). To obtain good level flight performance data the air must be smooth and in order to obtain these conditions early morning or late evening flights are scheduled.

Level flight data are taken at sufficient altitude spread to include the operating ranges of the helicopter in 10 kt increments. The minimum speed scheduled is approximately 30 kt because of the difficulty in stabilizing the helicopter and the accuracy of the airspeed indicator.

The helicopter is stabilized in level flight at the desired RPM and whatever MAP is required to maintain the desired IAS and constant altitude. Sufficient time should be allowed to determine if the aircraft is actually stabilized. The smoother the air the shorter the time required

| | | |
|---|---|---|
| WEATHER: Time<br>Wet Bulb<br>Dry Bulb<br>Baro | | CARD NO. 15<br>PTR BIS |
| AIRPLANE TYPE: HRS-1 | BU. NO.: 127783 | TIME T.O.<br>TIME LAND |
| DATE: 3/23/58 | | |
| PILOT: D. Blades | T.O.C.G.<br>GEAR DOWN $ UP $ | |
| CONDITION: Level Flight Perf | | T.O. GROSS WEIGHT: 7500 |

EXTERNAL CONFIGURATION: Normal Mixture

1500 ft. Hp

| CTR | RPM | Vi | MAP | FUEL | OAT | CAT | Remarks |
|---|---|---|---|---|---|---|---|
| | 2200 | 30 | | | | | |
| | | 40 | | | | | |
| | | 50 | | | | | |
| | | 60 | | | | | |
| | | 70 | | | | | |
| | | 80 | | | | | |
| | | 90 | | | | | |
| | | 100 | | | | | |
| | | 105 | | | | | |
| | | Vmax | | | | | |
| | | | | | | | |
| | | | | | | | |
| | | | Start | Stop | | | |
| Oscill Ctr # | | | | | | | |
| Photo Ctr | | | | | | | |
| | | | | | | | |
| | | | | | | | |
| | | | | | | | |
| | | | | | | | |

FLIGHT DATA-NATC-312 (Rev. 6-51)

Figure 3.3:1

to stabilize the helicopter. Stabilization on a particular point is accomplished by the pilot by holding altitude constant and manipulating throttle until the desired IAS is reached. At this time the pilot flies the desired IAS and checks the altimeter and rate of climb indicator making minor power adjustments to stop the rate of descent or ascent. If the data is to be taken at sea level it is easier to stabilize the aircraft just off the water as it no longer becomes necessary to check altitude variations. Caution of "slick" water should be exercised when using this method as it is difficult to accurately determine actual height above the water. Records should be approximately 2 to 3 sec in length. Another approach in obtaining level flight data and probably easier from a pilot's viewpoint is to maintain constant MAP instead of airspeed. For example; if a 60 kt, 2500 RPM and 1000 ft pressure altitude level flight point is desired, the pilot would obtain the constant RPM and altitude rather readily. Then as the airspeed is within ±5 kt from 60 kt, the pilot would note the MAP. Then with this MAP held constant he would let the airspeed settle down to a stabilized speed with a resultant speed close to the desired 60 kt point.

### 3.3.3 DATA REDUCTION PROCEDURE

The method of reducing level flight performance data is shown in figure 3.3:2 and is a similar method employed by NASA. This method uses the nondimensional values of P/L (rotor shaft power input), $\mu$ (tip speed ratio) and $C_T$ (thrust coefficient). As mentioned in paragraph 3.3.1, the power expended by a helicopter in forward flight may be summarized as follows:

$$HP_{Total} = HP_o + HP_i + HP \quad (1)$$

The above values may be expressed in terms of equivalent drag or energy dissipated per second.

Then

$$\left.\begin{aligned} HP_o &= D_oV \\ HP_i &= D_iV \\ HP_p &= D_pV \end{aligned}\right\} \quad (2)$$

$HP_{Total}$ = PV where P is the total equivalent drag force (not power)

## LEVEL FLIGHT (Power Req'd)

Card No. ______
Observers Sheet

Dry Bulb
Wet Bulb
Bar.Press.
$K_h$

Date

Test of

| CTR NO. | | | | | | | | | | |
|---|---|---|---|---|---|---|---|---|---|---|
| ① Hp | | | | | | | | | | |
| ② CAT | | | | | | | | | | |
| ③ OAT | | | | | | | | | | |
| ④ MAP | | | | | | | | | | |
| ⑤ $RPM_e$ | | | | | | | | | | |
| ⑥ $RPM_e^2$ | | | | | | | | | | |
| ⑦ $V_c$ | | | | | | | | | | |
| ⑧ $\rho_o/\rho$ | | | | | | | | | | |
| ⑨ $\sqrt{\rho_o/\rho}$ | | | | | | | | | | |
| ⑩ $V_T$ fps<br>⑦ x ⑨ x 1.689 | | | | | | | | | | |
| ⑪ $T_{as}$ | | | | | | | | | | |
| ⑫ $T_{co}$ | | | | | | | | | | |
| ⑬ $\sqrt{\frac{T_{as}}{T_{co}}}$ | | | | | | | | | | |
| ⑭ W | | | | | | | | | | |
| ⑮ BHP chart<br>①, ④⑤ | | | | | | | | | | |
| ⑯ $BHP_o$<br>⑬ x ⑮ | | | | | | | | | | |
| ⑰ $BHP_{oh}$<br>⑯ x $K_h$ | | | | | | | | | | |

Figure 3.3:2.1

Project Number	Project	Observers

| | | | | | | | | | | |
|---|---|---|---|---|---|---|---|---|---|---|
| 18 Fan Loss | | | | | | | | | | |
| 19 Tail Loss | | | | | | | | | | |
| 20 Trans. Loss | | | | | | | | | | |
| 21 Total Loss | | | | | | | | | | |
| 22 BHP R. Shaft 17 - 21 | | | | | | | | | | |
| 23 N Rotor RPM | | | | | | | | | | |
| 24 P/L $\frac{550 \times 22}{14 \times 10}$ | | | | | | | | | | |
| 25 $C_T$ $\frac{14 \times 8}{6 \times K_{CT}}$ | | | | | | | | | | |
| 26 $\Omega R$ | | | | | | | | | | |
| 27 $\mu$ 10/26 | | | | | | | | | | |

Figure 3.3:2.2

Substituting equation (2) in equation (1) then $P = D_o + D_i + D_p$ (3). Equation (3) is made dimensionless by dividing through by rotor lift. Thus $P/L = (\frac{D}{L})_o + (\frac{D}{L})_i + (\frac{D}{L})_p$. P/L is analogous to the drag lift ratio of an airplane and represents the drag that would absorb the same power at the velocity along the flight path as the power being supplied through the rotor shaft.

In the data reduction computations shown in figure 3.3:2, columns (1) thru (7) are test data instrumentation readings corrected for instrument error. Column (7) is obtained by employing the airspeed calibration chart after the indicated airspeed has been corrected for instrument error. Engine power column (15) is obtained from the engine manufacturer's power chart or by the use of a torquemeter. If a torquemeter is employed column (16) and (17) can be omitted. Columns (18) thru (20) are usually obtained from the contractor. Tail rotor power loss column (19) if applicable may be determined by a torquemeter if available. The determination of the values of $K_h$ and $K_{C_T}$ are explained in Section 3.1. Normally in the plot of test data values (P/L vs $\mu$ ) the value of $C_T$ will vary slightly because of weight change from fuel usage, RPM and slight altitude differences. Fairing of the curve should be accomplished to obtain a line of constant $C_T$.

### 3.3.4 TYPICAL RESULTS

The chart of P/L vs $\mu$ , figure 3.3:3 for constant values of $C_T$ is utilized to plot speed power polars at various gross weights, RPM's and altitudes as shown in figure 3.3:4. Figure 3.3:5 shows the maximum level flight speed as limited by blade stall or power available. The solid lines are based on power available limitations or the intersection of power required and power available at each individual altitude. The dashed line is based strictly on blade stall at the individual altitude.

Model HRS-1 Helicopter
BuNo 127783

MAIN-ROTOR-SHAFT POWER PARAMETER
vs
TIP-SPEED RATIO
STEADY LEVEL FLIGHT

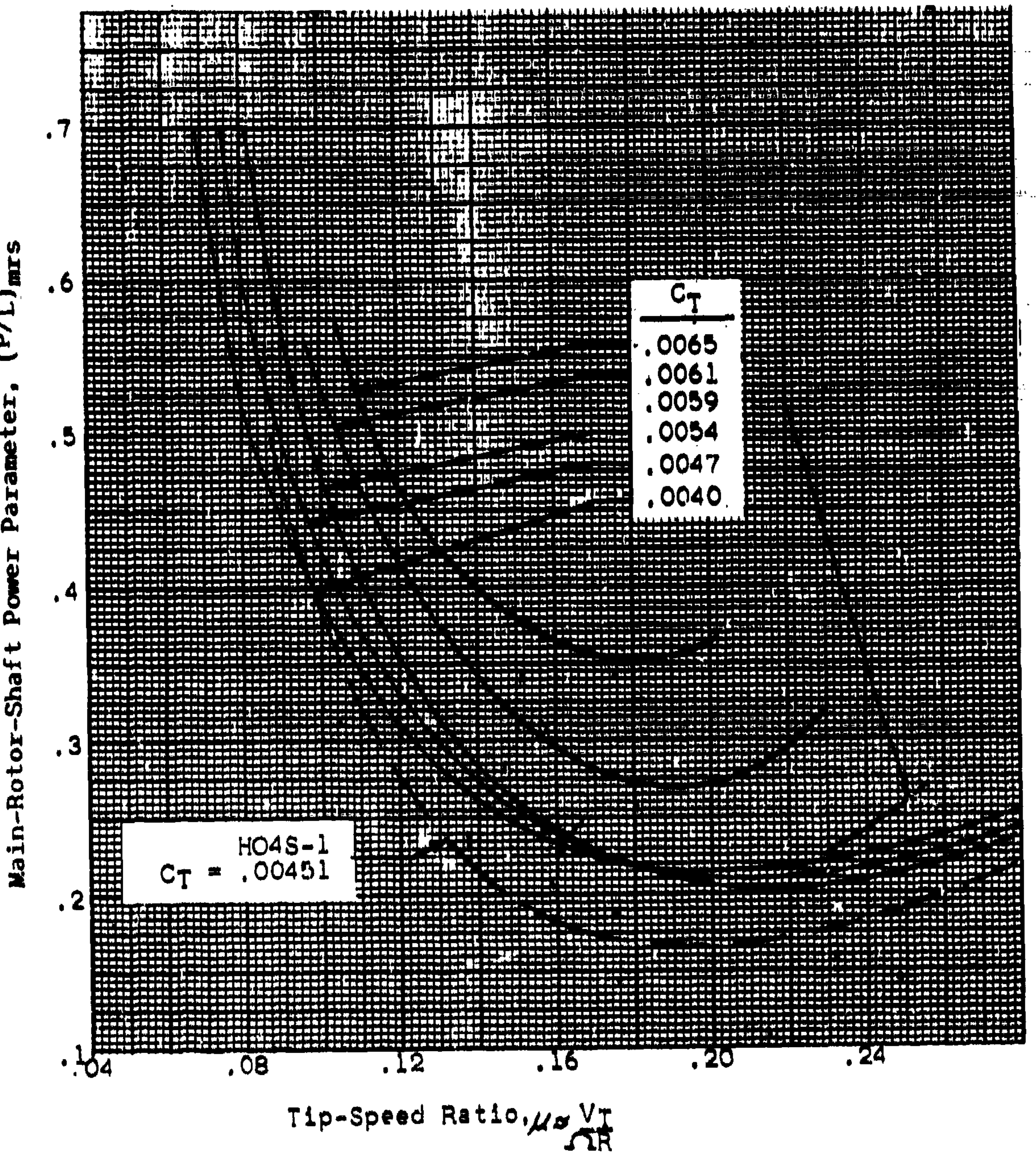



Figure 3.3:3

Model HRS-1 Helicopter
BuNo 127783

LEVEL FLIGHT PERFORMANCE
POWER REQUIRED AT 1500 FT ALTITUDE

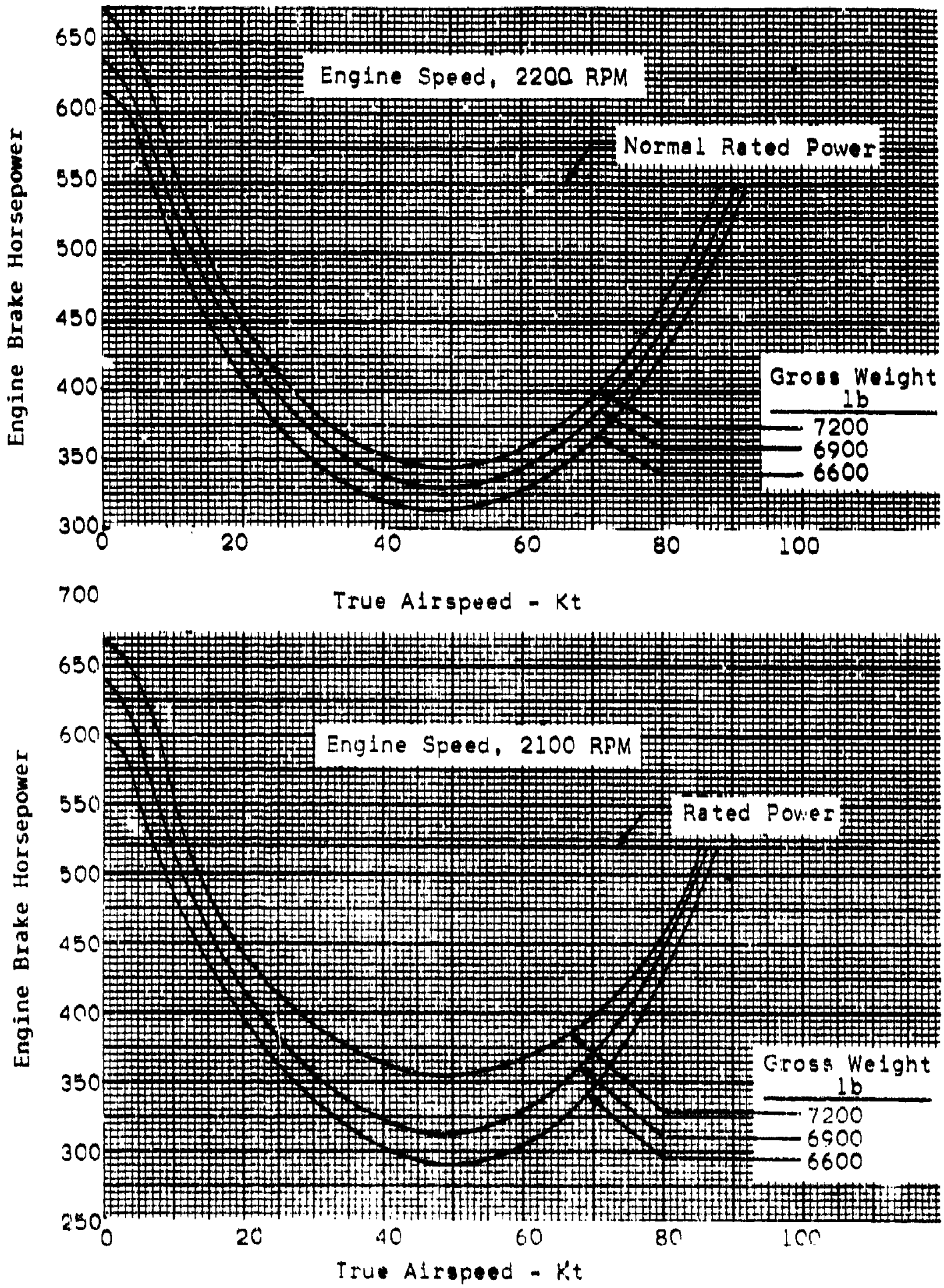


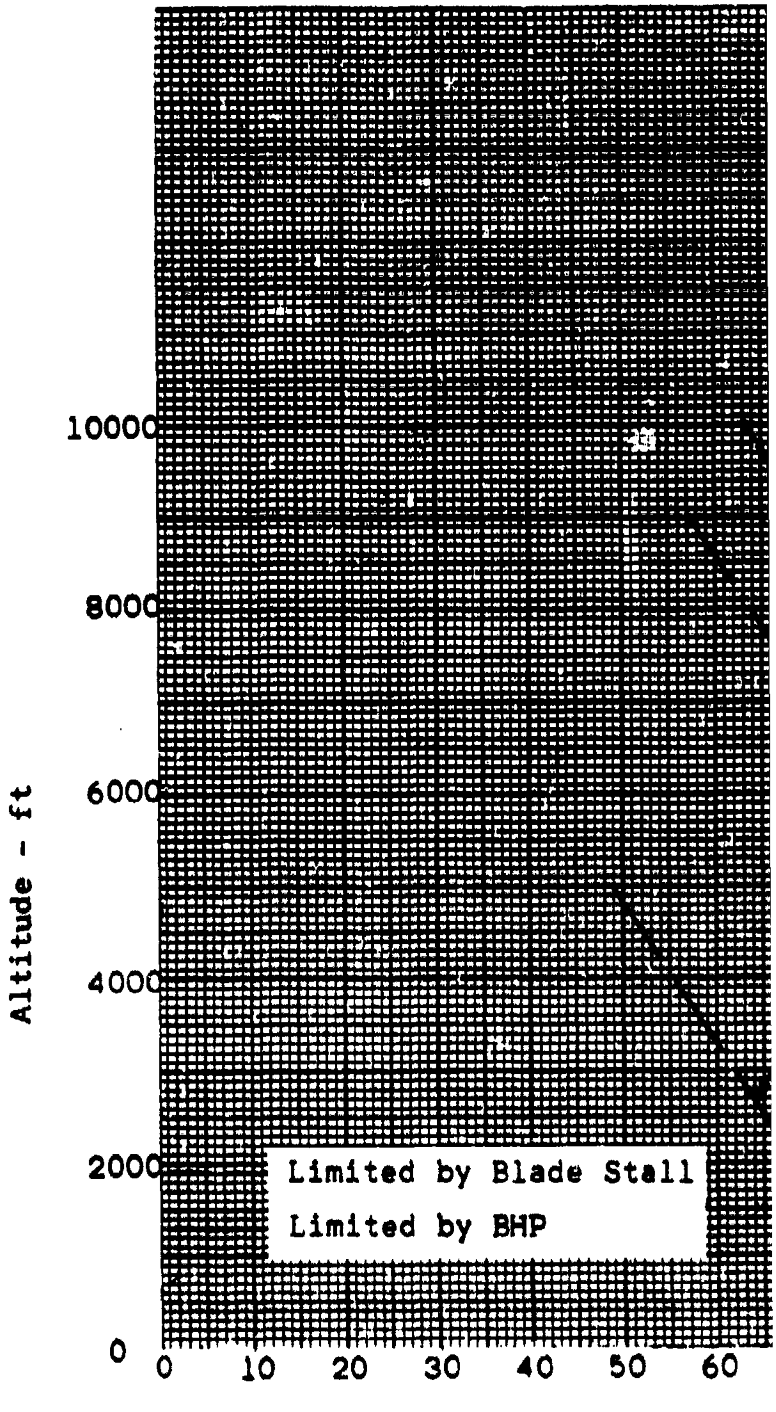

Figure 3.3:4

Model XHJP-1 Helicopt

BuNos 37976, 37977

MAXIMUM VELOCITY
LEVEL FLIGHT

2300 RPM




True Airspeed

Figure 3.3:5

# FORWARD SPEED CLIMBS AND BEST CLIMB SPEED

## Section 3.4

### 3.4.1 INTRODUCTION AND PURPOSE

The purpose of this chapter is to discuss one aspect of the climb performance of helicopters; namely, service ceiling climb at the speed for best climb. The discussion will include the sawtooth climb method of determining best speed for climb at various altitudes, flight techniques and methods, data reduction procedures, and sample tables and graphs which present the data.

Service ceiling is that altitude at which the rate of climb reduces to 100 ft/min at full throttle on a standard day. It is a very lengthy and laborious chore to continue flight above service ceiling to an altitude at which the rate of climb degenerates to zero ft/min, called absolute ceiling; thus the term "service ceiling" is a more realistic and widely used term than "absolute ceiling."

The problem of computing climb performance requires the solution to either one of two alternate problems.

1. Compute the rate of climb at a given airspeed for a given available power.

2. Compute the power required to climb at a given rate of climb and at a given airspeed.

The latter problem is encountered in the design stage, wherein the aircraft contractor attempts to determine what type of power plant will be required to attain given minimum allowable performance. The former problem is encountered in flight testing.

Computations made through utilization of the performance charts are made in order to plot a curve of rate of climb versus airspeed at a given take-off gross weight for various altitudes. Shown in figure 3.4:1 are typical results of preliminary sawtooth climb performance computations.

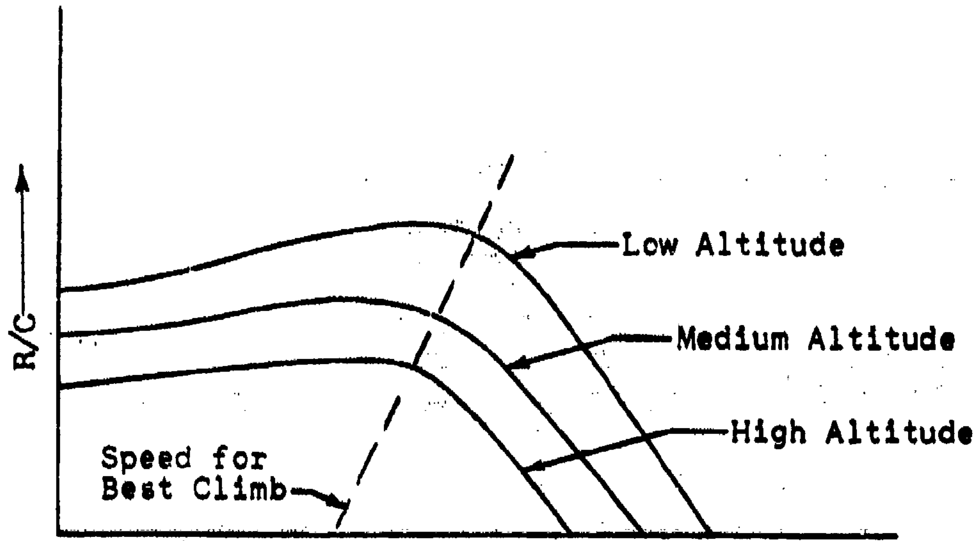


Figure 3.4:1

From these preliminary computations, it is possible to predict the schedule of best climbing speed versus altitude.

### 3.4.2 METHOD AND TECHNIQUE

Actual climb performance flight testing of the aircraft commences with a sawtooth climb evaluation to check the predicted best climbing speed schedule versus altitude. Prior to conducting sawtooth climbs the airspeed position error in climb is determined with a trailing pitot static bomb. This type of calibration is discussed completely in Part II. This calibration is usually conducted on the same flight that the position error in level flight is determined. In performing the sawtooth climbs, the helicopter is flown through a given altitude increment at a constant power setting and airspeed. The climb is started approximately 500 ft below the initial altitude marker to allow complete stabilization of the helicopter. Repeated runs are conducted through the same altitude increments at several airspeeds in the range near the estimated best climb speed. The above procedure is repeated at other altitudes. Instrumentation bursts are made of 3 sec duration at each 500 ft increment through each individual 1000 ft climb and the following items are recorded:

a. Altitude

b. Time

c. MAP

d. RPM

e. OAT

f. Fuel Consumed

g. CAT

h. IAS

Carburetor heat should be omitted unless required to prevent icing. A constant amount of carburetor heat, if required, should be employed for all climbs conducted at a particular altitude. Mixture control should also remain unchanged during the climb. In order to utilize flight time most efficiently, sawtooth autorotations are conducted in conjunction with sawtooth climbs in a similar manner as described above. Figure 3.4:2 is a sample flight card for such a flight.

Instrumentation records are processed to obtain true rate of climb, (observed corrected for ambient air conditions) and indicated airspeed, and to ascertain that each climb was conducted at the same power for the given altitude increment. The true rate of climb is corrected to standard rate of climb by including a correction due to deviation of test gross weight from standard. Shown in figure 3.4:3 is a plot of standard rate of climb versus true airspeeds for several altitudes. Figure 3.4:9 is a sample data reduction sheet used for sawtooth climb evaluation.

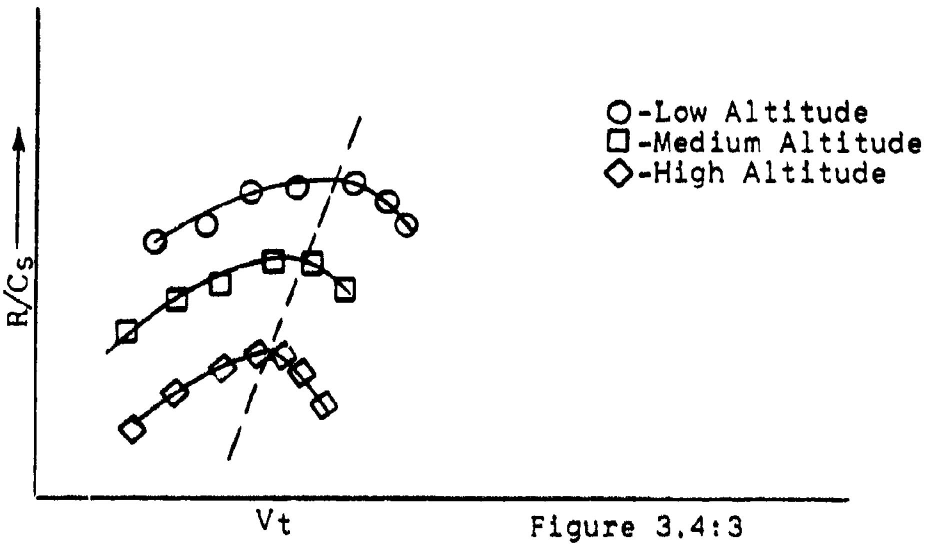



Figure 3.4:3

| WEATHER | Wet Bulb<br>Dry Bulb<br>Bar.Press. | | CARD NO. 10<br>PTR DIS 21139 |
|---|---|---|---|
| AIRPLANE TYPE HRS-1 | BU. NO. | TIME T.O.<br>TIME LAND | DATE 2/1/52 |
| PILOT Major Gill | | T.O.C.G. 131.0<br>GEAR DOWN % UP % | |
| CONDITION | | | T.O. GROSS WEIGHT 6900 |

EXTERNAL CONFIGURATION

Sawtooth Climb & Autorotation

500' - 1500' $H_p$ Normal Mixture

| $V_i$ | RPM | MAP | CAT | OAT | FUEL | TIME | CTR |
|---|---|---|---|---|---|---|---|
| 30 | 2250 | 36.5 | | | | | |
| 40 | ↓ | | | | | | |
| 50 | ↓ | | | | | | |
| 60 | ↓ | | | | | | |
| 60 | ↓ | | | | | | |
| 50 | ↓ | | | | | | |
| 40 | ↓ | | | | | | |
| 30 | ↓ | | | | | | |
| 30 | 2200 | 35 | | | | | |
| 40 | ↓ | | | | | | |
| 50 | ↓ | | | | | | |
| 60 | ↓ | | | | | | |
| 60 | ↓ | | | | | | |
| 50 | ↓ | | | | | | |
| 40 | ↓ | | | | | | |
| 30 | ↓ | | | | | | |
| | | | | | | | |
| | | | | | | | |
| | | | | | | | |
| | | | | | | | |

FLIGHT DATA-PNMC-NATC-312 (REV. 6-51)

SAMPLE FLIGHT CARD FOR SAWTOOTH CLIMB

Figure 3.4:2

Subsequent to the completion of sawtooth climb performance tests, the schedule of best climbing speed versus altitude is established and the service ceiling climb performance evaluation is begun. The test pilot is given a schedule of power settings and airspeed versus altitude, and executes climbs to the service ceiling at various gross weights. Service ceiling climbs should be conducted under smooth air conditions, unlimited ceiling and visibility and free of temperature inversions. A typical flight card for such a climb is presented as figure 3.4:4.

### 3.4.3 DATA REDUCTION PROCEDURE AND TYPICAL RESULTS

Figure 3.4:10 is a sample data reduction sheet which is employed to reduce the flight data into curves of indicated brake horsepower versus indicated rate of climb for various take-off gross weights. This plot is shown in figure 3.4:5. A straight line relationship exists between $R/C_i$ and $BHP_i$ provided constant angle of attack and efficiency for varying $V_T$ and altitude are maintained. The deviations from constant and $\eta$ result in a curvature in the relationship but does not affect the suitability of the method when used for reducing observed data to standard conditions. A substantial simplification is obtained by omitting the correction to standard weight which provides data that plot satisfactorily although additional curvature is introduced by omitting the weight correction. The latter deviation is employed by this activity in the reduction of service ceiling climb data to standard conditions, therefore climb data is based on take-off gross weights.

An important phase of service ceiling climb performance flight testing is the determination of standard power for insertion into column 2 of figure 3.4:11. If the aircraft is equipped with a strain gage type torquemeter, the determination of standard power is simplified, since torquemeter power readings are not affected by ambient air conditions. However, the power readings of the helicopter which are undergoing flight tests without a torquemeter are appreciably affected in climb performance by ambient air conditions. At altitudes above the critical altitude, the engine is incapable of producing more power than the full-open setting that the throttle allows. Consequently, at a given altitude, on colder days the full-open throttle setting will allow a greater mass flow rate of air through the engine than on warmer days. To facilitate determination of standard engine power available at altitudes above critical, this activity conducts several service ceiling climbs at a given take-off

WEATHER Wet Bulb
Dry Bulb
Bar.Press.

CARD NO. 20

PTR DIS 21139

| AIRPLANE TYPE | BU. NO. | TIME T.O. / TIME LAND | DATE |
|---|---|---|---|
| HRS-1 | | | 2/10/52 |

PILOT Major Gill

T.O.C.G. 131.0 GEAR DOWN % UP %

CONDITION

T.O. GROSS WEIGHT 6900

EXTERNAL CONFIGURATION

Service Ceiling Climb

Normal Mixture

| ALT | RPM | Vi | MAP | CAL | OAT | FUEL | CTR |
|---|---|---|---|---|---|---|---|
| 0 | 2250 | 50 | 36.5 | | | | |
| 500 | | | | | | | |
| 1000 | | | | | | | |
| 1500 | | | | | | | |
| 2000 | | | | | | | |
| 2500 | | | | | | | |
| 3000 | | | | | | | |
| 3500 | | | | | | | |
| 4000 | | | | | | | |
| 4500 | | | | | | | |
| 5000 | | | | | | | |
| 5500 | | | | | | | |
| 6000 | | | | | | | |
| 7000 | | | | | | | |
| 8000 | | | | | | | |
| 9000 | | | | | | | |
| 10000 | | | | | | | |
| 11000 | | | | | | | |
| 12000 | | | | | | | |

FLIGHT DATA-PNUC-NATC-312 (Rev. 6-51)

SAMPLE FLIGHT CARD FOR SERVICE CEILING CLIMB

Figure 3.4:4

gross weight and varying ambient air conditions. At altitudes below critical, the test pilot must follow a power schedule which has taken into account the ambient air conditions on power at a given manifold schedule, i.e., on colder days the schedule must specify manifold pressures lower than those specified in the engine manufacturer's chart, in order to perform the climb at normal rated power.

Power Available With
Ambient Condition

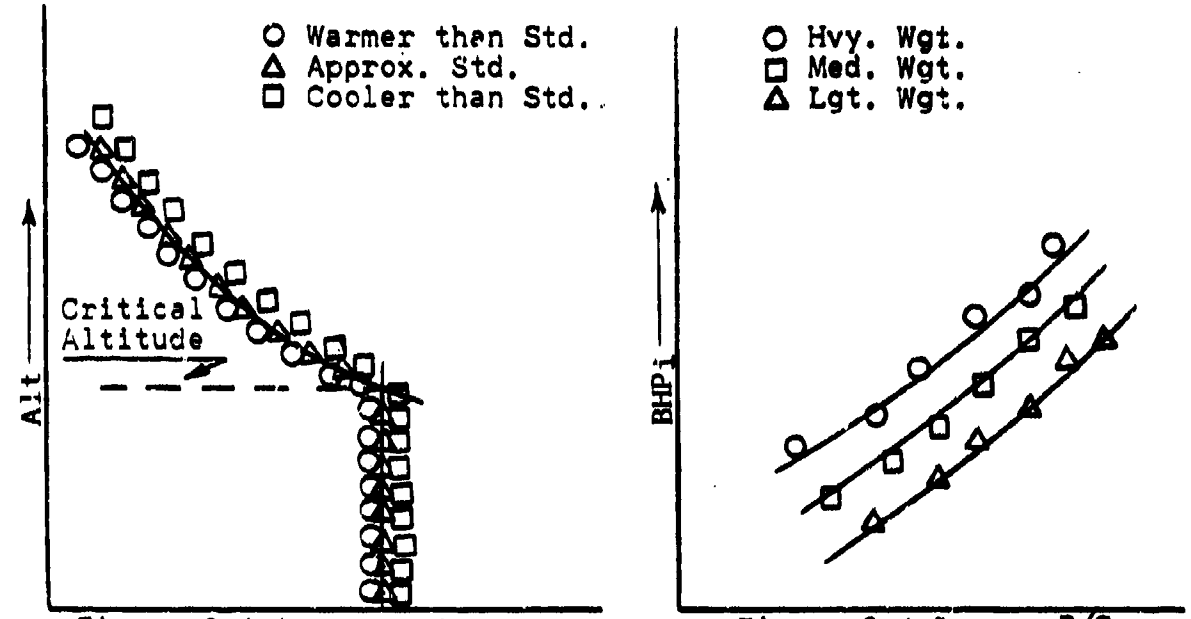



Figure 3.4:6 Figure 3.4:5

Shown in figure 3.4:6 is a plot of $BHP_s$ versus altitude, where $BHP_s$ is obtained from column 24 of figure 3.4:10. If the helicopter is not equipped with a torquemeter, $BHP_s$ is obtained by the method described in figure 3.4:10, notes a and b. A mean line is faired through the points above critical and the critical altitude is established as that altitude where the faired high altitude line intersects a line drawn from the sea level NRP value. The sea level NRP value is obtained from the engine manufacturer's chart with the assumption that the engine is capable of delivering this rated power by manipulations of manifold pressure for variations in ambient air conditions. Subsequent to the establishment of the standard power available schedule versus altitude, the rates of climb attained by employing this specified rated power are obtained by entering the curve of $BHP_i$ versus $R/C_i$ shown in figure 3.4:5 with standard power available values for a given altitude.

A curve of standard rate of climb versus altitude is then obtained by the process of figure 3.4:11.

In determining the time required to climb to a given altitude, the following assumption is made: the rate of climb at an altitude midway between two specified altitudes is the average rate of climb between these two altitudes. Thus, the time required to climb to 1000 ft equals the rate of climb at 500 ft divided into 1000 ft (see figure 3.4:7).

The final presentation of climb results is shown in figure 3.4:8.

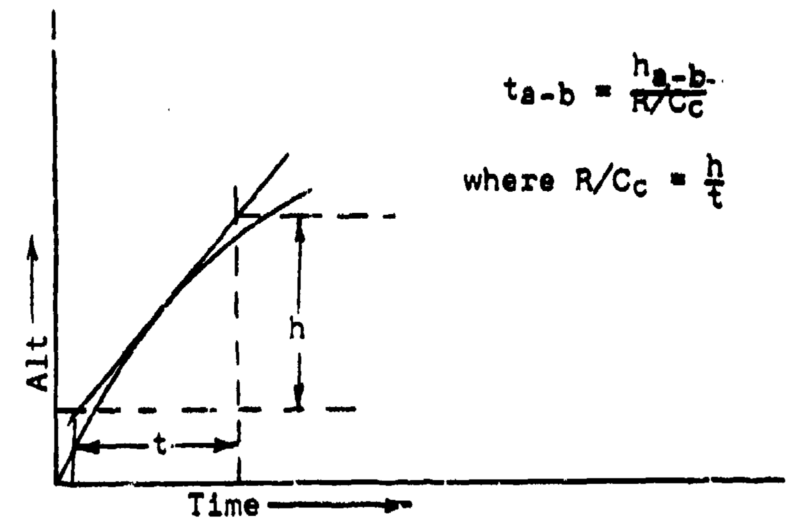



Figure 3.4:7

Model HRS-1 Helicopter

CLIMB PERFORMANCE AT 50 KT
INDICATED AIRSPEED USING SETTINGS
FOR NORMAL RATED POWER
AND TAKE-OFF RATED POWER

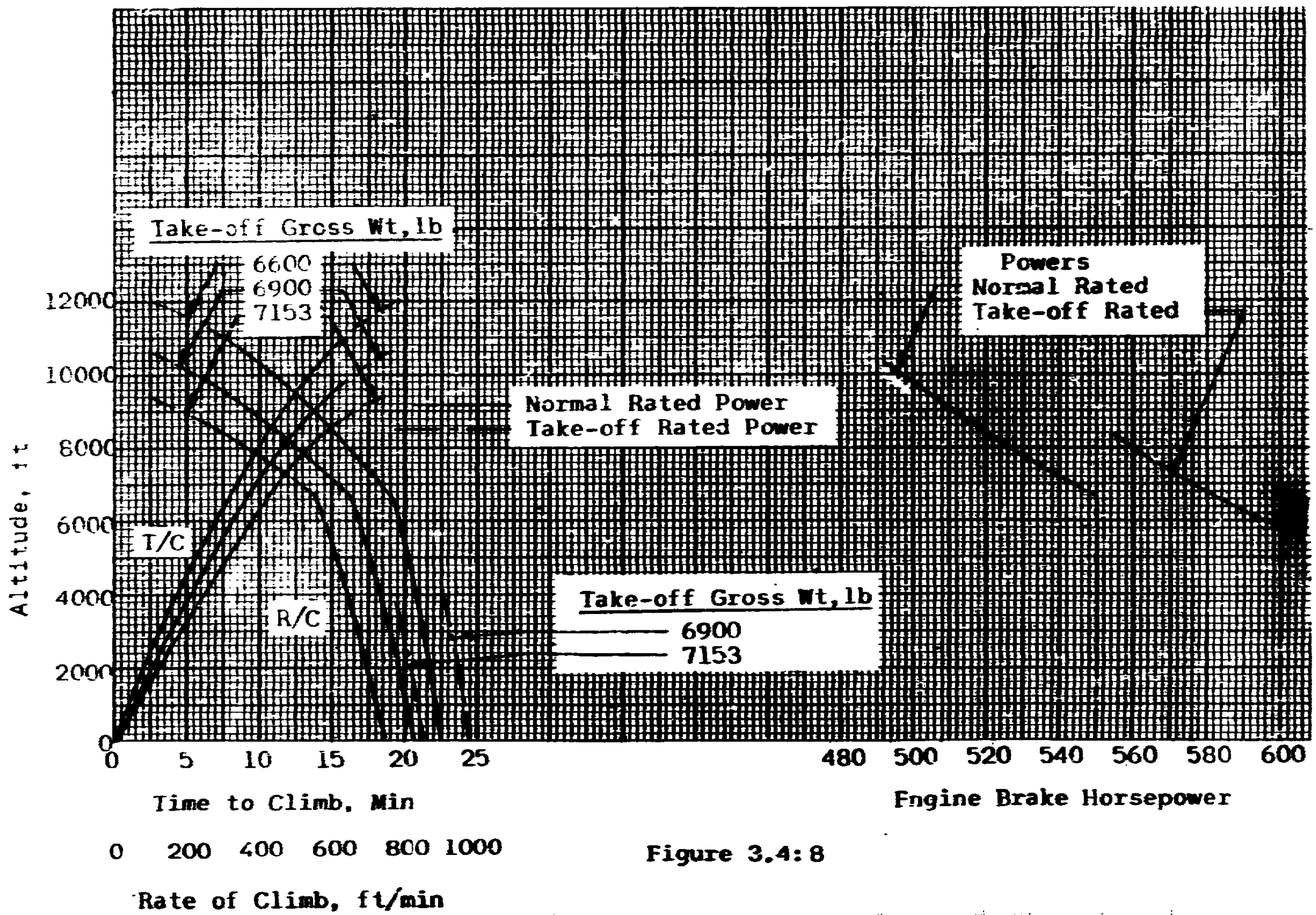



Figure 3.4:8

Observers Sheet — Table I

Date — Test of SAWTOOTH CLIMB EVALUATION

| Film Ctr. | | | | | | | | | |
|---|---|---|---|---|---|---|---|---|---|
| 1 $V_i$ | | | | | | | | | |
| 2 $H_p$ | | | | | | | | | |
| 3 Time | | | | | | | | | |
| 4 MAP | | | | | | | | | |
| 5 RPM | | | | | | | | | |
| 6 $BHP_{ch}$ | | | | | | | | | |
| Plot Columns 3, 4, 5 & 6 vs Column 2 for each $V_i$, Column 1 | | | | | | | | | |
| 7 R/Co | | | | | | | | | |
| 8 $t_{ao}$ | | | | | | | | | |
| 9 $T_{ao}$ 8 + 273 | | | | | | | | | |
| 10 $T_{as}$ From 2 | | | | | | | | | |
| 11 $\frac{T_{ao}}{T_{as}}$ | | | | | | | | | |
| 12 R/Ct 7 x 11 | | | | | | | | | |
| 13 $W_s$ | | | | | | | | | |
| 14 $W_o$ | | | | | | | | | |
| 15 $\frac{W_o}{W_s}$ 14 ÷ 13 | | | | | | | | | |
| 16 $\sqrt{\frac{W_o}{W_s}}$ $\sqrt{15}$ | | | | | | | | | |

Figure 3.4:9.1

Observers Sheet Table I (Cont')

Date Test of SAWTOOTH CLIMB EVALUATION

| | | | | | | | | | | |
|---|---|---|---|---|---|---|---|---|---|---|
| 17 $R/C_s$<br>12 x 16 | | | | | | | | | | |
| 18 Vc<br>Calibration Chart | | | | | | | | | | |
| 19 $V_T$<br>$V_c\sqrt{\frac{\rho_o}{\rho}}$ | | | | | | | | | | |
| 20 Plot<br>17 vs 19 | for various values of 2 | | | | | | | | | |

Figure 3.4:9.2

Table II

SERVICE CEILING CLIMB

*Corrected

Aircraft________

Wet Bulb
Dry Bulb
Bar.Press.
Hum.Factor

| | | | | | | | | |
|---|---|---|---|---|---|---|---|---|
| 1 $H_p$ * | | | | | | | | |
| 2 RPM* | | | | | | | | |
| 3 MAP* | | | | | | | | |
| 4 OAT* | | | | | | | | |
| 5 CAT* | | | | | | | | |
| 6 CDP* | | | | | | | | |
| 7 Elapsed Time, min | | | | | | | | |
| 8 BHP Chart | | | | | | | | |
| 9 $H_p$ (Even Increm) | 0 | 1000 | 2000 | 3000 | 4000 | 5000 | 6000 | 6500 |
| 10 $t_{ao}$ | | | | | | | | |
| 11 $t_{co}$ | | | | | | | | |
| 12 $T_{ao}$ | | | | | | | | |
| 13 $T_{co}$ | | | | | | | | |
| 14 $T_{as}$ | | | | | | | | |
| 15 $R/C_o$ | | | | | | | | |
| 16 $\sqrt{\frac{T_{as}}{T_{co}}}$ | | | | | | | | |
| 17 $\frac{T_{ao}}{T_{as}}$ | | | | | | | | |
| 18 BHP Graph | | | | | | | | |
| 19 $BHP_{oh}$ 18 x 16 x $K_h$ | | | | | | | | |

* Determination of $BHP_s$

a. Part Throttle - $BHP_s$ = normal rated or take-off rated power as guaranteed until critical is obtained

b. Full Throttle $BHP_s$ Available
$BHP_s$ = $BHP_{oh}$ x Factor From (Engine Manufacturers Power Chart) Wright Case II Chart a function of MAP, CDP, $H_p$, $t_{ao}$, and $t_{as}$.

c. $BHP_s$ (Torquemeter employed) Above Critic

Take Torquemeter engine power with following corrections:

1. RPM Variation from desired RPM - (Straight ratio)

2. $\sqrt{\frac{T_{ao}}{T_{as}}}$

Figure 3.4:10.1

Table II

SERVICE CEILING CLIMB

*Corrected

Aircraft________

Wet Bulb
Dry Bulb
Bar.Press.
Hum.Factor

| | | | | | | | | |
|---|---|---|---|---|---|---|---|---|
| 20 $R_{CT}$ 15 x 17 | | | | | | | | |
| 21 $\sqrt{\rho_o/\rho}$ for 9 and 10 | | | | | | | | |
| 22 $BHP_i$ 19 ÷ 21 | | | | | | | | |
| 23 $R/C_i$ 20 ÷ 21 | | | | | | | | |

Figure 3.4:10.2

# SERVICE CEILING CLIMB

## Final

Aircraft ________ Date__________

| | | | | | | | | | |
|---|---|---|---|---|---|---|---|---|---|
| ① $H_p$ | | | | | | | | | |
| ② $\sqrt{\theta/\rho}$ | | | | | | | | | |
| ③* $BHP_s$ | | | | | | | | | |
| ④ $BHP_{is}$ 3 ÷ 2 | | | | | | | | | |
| ⑤ $R/C_{is}$ from plot of 22 vs 23 | | | | | | | | | |
| ⑥ $R/C_{ts}$ 5 x 2 | | | | | | | | | |
| ⑦** T/C | | | | | | | | | |

* See Figure 3.4:6 for determination of $H_s$ vs $BHP_s$ curve

** See Figure 3.4:7

Figure 3.4:11

## FUEL CONSUMPTION

## Section 3.5

### 3.5.1 INTRODUCTION AND PURPOSE

The purpose of obtaining fuel consumption data under various flight conditions is to determine the range and endurance characteristics of the helicopter. Operationally, one of the most important items of performance is the optimum distance that can be flown on a given fuel quantity or the optimum time that the helicopter can remain airborne. Fuel consumption tests are generally conducted to determine compliance with contract guarantees. These guarantees are generated from a design consideration to a particular type of mission. The helicopter Detail Specification usually regulates the percentage of time required for each flight regime when determining range and endurance characteristics (hover 60%, level flight 40% etc, % of fuel remaining for reserve and warm-up and take-off).

### 3.5.2 METHOD AND TECHNIQUE

Prior to conducting fuel consumption tests the aircraft's fuel tank is calibrated to determine fuel gage error and the exact fuel load of the helicopter. The ideal method of calibrating the fuel tank is to drain all the fuel from the helicopter and perform the entire calibration utilizing the Flight Test Division weighing scales. Exact delta fuel increment weights can then be directly compared to the gage readings. Performing the calibration in this manner requires a standby fire truck and removal of all aircraft from the hangar. Since this is impractical for obvious reasons, the calibration is conducted outside the hangar in the following manner:

a. Helicopter drained of all removable fuel.

b. Fuel truck supplies fuel in 5 gal increments - corresponding fuel gage read at each increment.

c. Specific gravity of fuel determined midway thru calibration with hydrometer.

d. Three weighings are conducted with fuel tank empty, half full and full fuel.

### 3.5.3 DATA REDUCTION PROCEDURE

The method of reducing fuel consumption data is shown in figure 3.5:2. The determination of engine brake horsepower as shown in figure 3.5:2 is based on the engine manufacturer's power chart. However, if a torquemeter is available engine power is determined by this means and item (15) and (16) of figure 3.5:2 are omitted.

### 3.5.4 TYPICAL RESULTS

Fuel consumption data, reduced as described above, are first plotted in the form of fuel flow vs BHP as shown in figure 3.5:3. Usually some scatter is present and the data must be faired for each RPM, mixture and altitude. Normally helicopters are associated with low altitude regime of flight which reduces the extent of the fuel consumption investigation. A 1000 ft flight variation has minor effect on the fuel flow data. After the plot of figure 3.5:3 has been made a cross plot of the faired lines is performed and plotted in the form of SFC vs BHP as shown in figure 3.5:4. Utilizing the level flight charts obtained in Section 3.3 final fuel flow charts are presented as shown in figures 3.5:5 and 3.5:6. The maximum point on the chart of figure 3.5:6 gives the speed for maximum range. An approximate method for obtaining the speed for maximum range is to draw a line from the origin of the level flight horsepower required chart tangent to the curve as shown in figure 3.5:7. The difference in the determination of maximum range airspeed as shown in figure 3.5:6 and 3.5:7 is approximately 2 kt.

From the above data, the fuel tank calibration is plotted with the actual aircraft weighings as the guiding standard.

In addition to the fuel tank calibration, the fuel flowmeter and carburetor are also calibrated. The carburetor is calibrated at the Aeronautical Engine Laboratory at Philadelphia whereas the fuel flowmeter is calibrated locally by the Instrumentation Branch of the Flight Test Division. A further check on the metering characteristics of the carburetor should be made periodically utilizing flight test data and plotting metering suction differential (MSD) vs fuel flow as shown in figure 3.5:1.

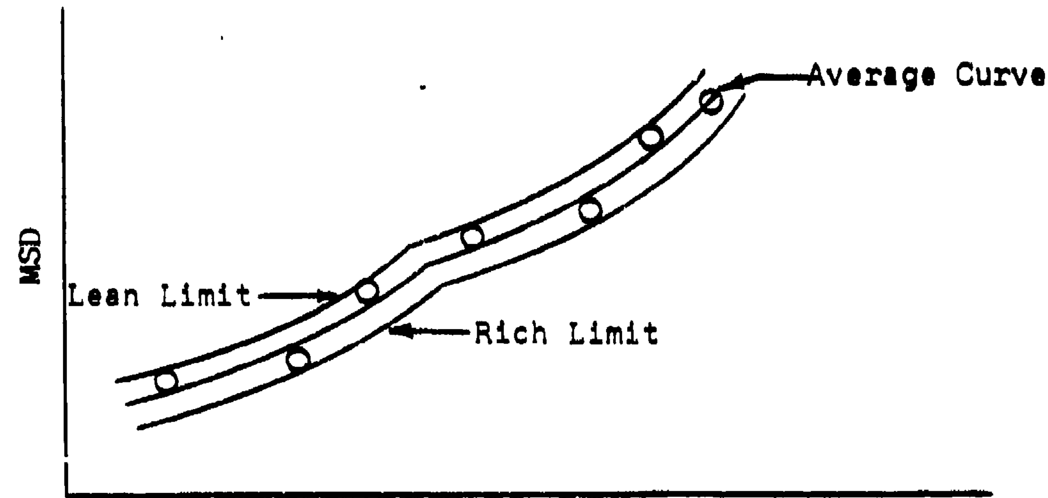



Figure 3.5:1

These data are then checked with the manufacturer's tolerances to assure that the helicopter has a typical carburetor.

Fuel consumption tests are usually conducted in conjunction with level flight performance and hover tests. Fuel flowmeter readings are obtained at each stabilized flight condition. A typical flight card for the test is contained in Section 3.1 and 3.3. Test data coverage should include the operating range of gross weight, airspeed, altitude and RPM.

Project Work Sheet (Calibration)

TYPE HELICOPTER ____________ BuNo__________ Gross Wt__________

CARD NO ____

*OBSERVED DATA CORRECTED FOR INSTR ERROR

| | | | | | | | | |
|---|---|---|---|---|---|---|---|---|
| (1) CTR No | | | | | | | | |
| (2) IAS*, kt | | | | | | | | |
| (3) OAT* °C | | | | | | | | |
| (4) Press Alt* ft | | | | | | | | |
| (5) $\sqrt{\rho_o/\rho}$ | | | | | | | | |
| (6) CAS kt from Airspeed Calibration Curve | | | | | | | | |
| (7) $V_T$, kt = (5) x (6) | | | | | | | | |
| (8) RPM* | | | | | | | | |
| (9) MAP* "Hg | | | | | | | | |
| (10) MSD* | | | | | | | | |
| (11) Tas | | | | | | | | |
| (12) CAT* | | | | | | | | |
| (13) Tco | | | | | | | | |
| (14) $\sqrt{\frac{Tas}{Tco}}$ | | | | | | | | |
| (15) $BHP_{chart}$ | | | | | | | | |
| (16) $BHP_o$ (14) x (15) | | If torquemeters are installed determine $BHP_{req'd}$ utilizing $\frac{TORQ(ft\text{-}lb) \times RPM}{5252}$ = BHP | | | | | | |
| (17) $BHP_{oh}$ (16) x $K_h$ | | | | | | | | |
| (18) *Fuel Flow, $\frac{lb}{hr}$ | | | | | | | | |
| (19) GPH | | | | | | | | |
| (20) SFC (19) ÷ (16) | | | | | | | | |

Figure 3.5:2

Model H-23D Helicopter
ASN 57-2987

FUEL FLOW
vs
ENGINE BRAKE HORSEPOWER

3000 - 3200 RPM

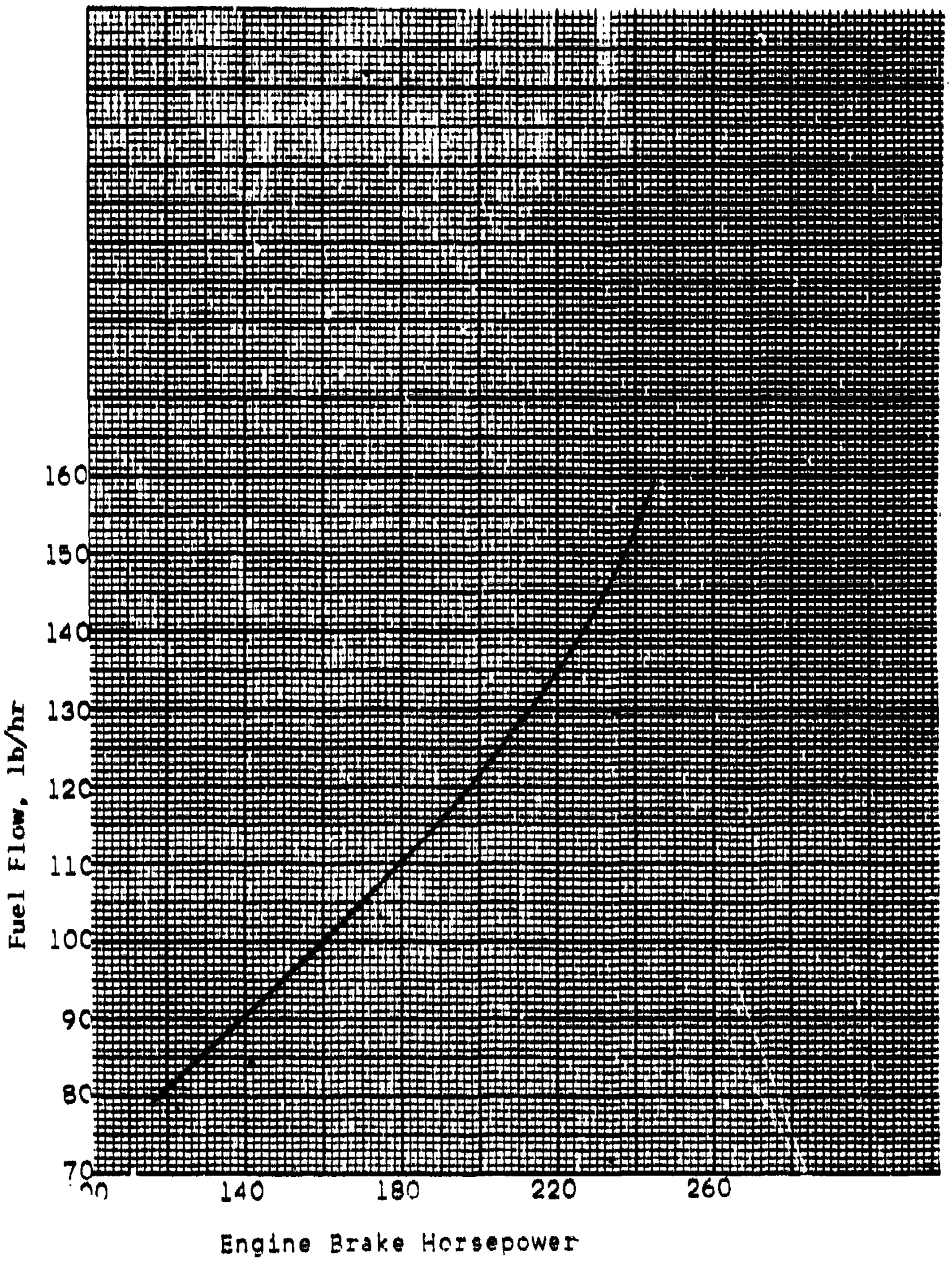



Figure 3.5:3

Model H-23D Helicopter
ASN 57-2987

SPECIFIC FUEL CONSUMPTION
vs
ENGINE BRAKE HORSEPOWER

3000 - 3200 RPM

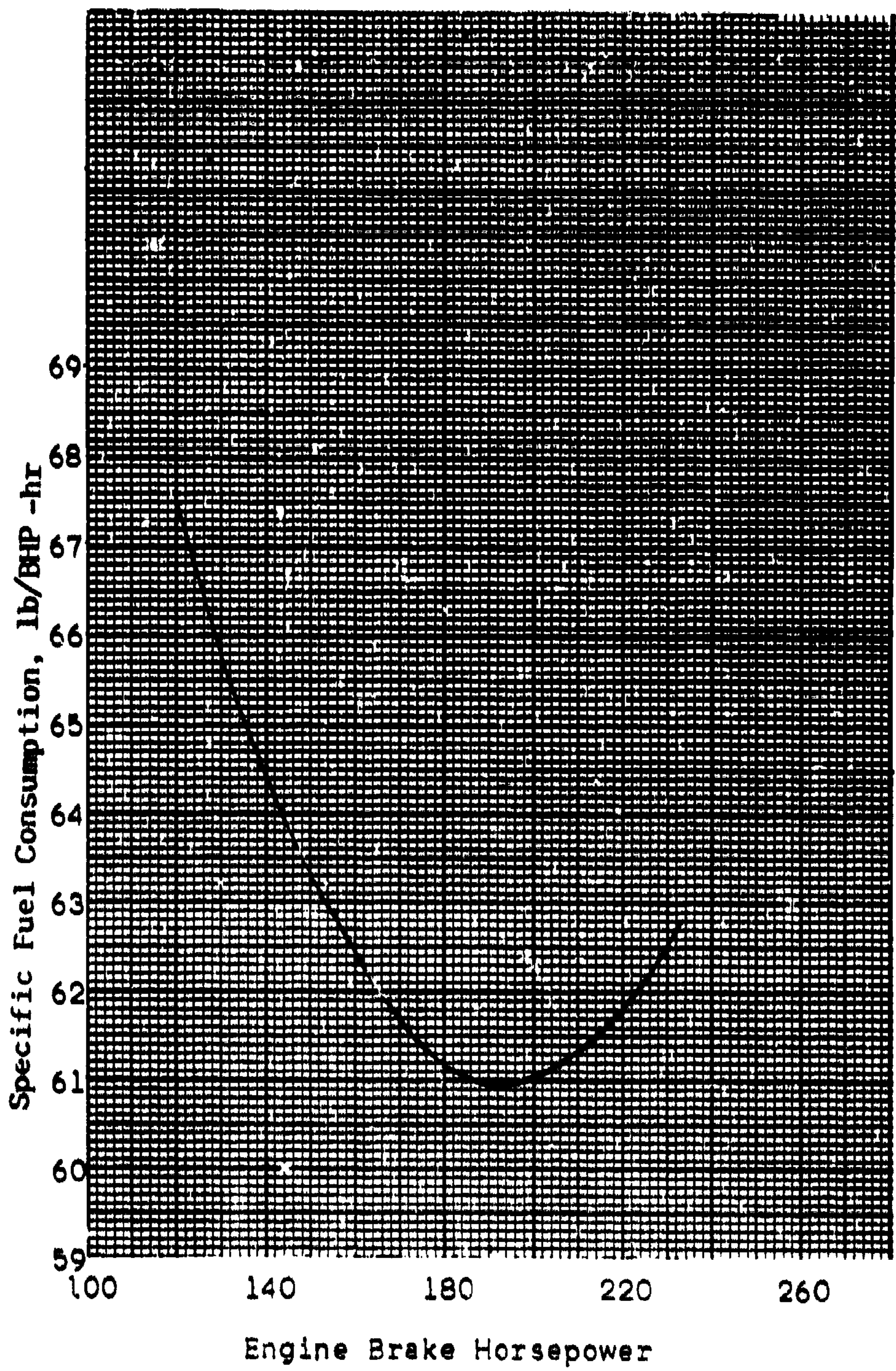



Figure 3.5:4

Model H-23D Helicopter
ASN 57-2987

FUEL FLOW
vs
TRUE AIRSPEED

Gross Weight - 2600 lb
Engine Speed - 3200 RPM

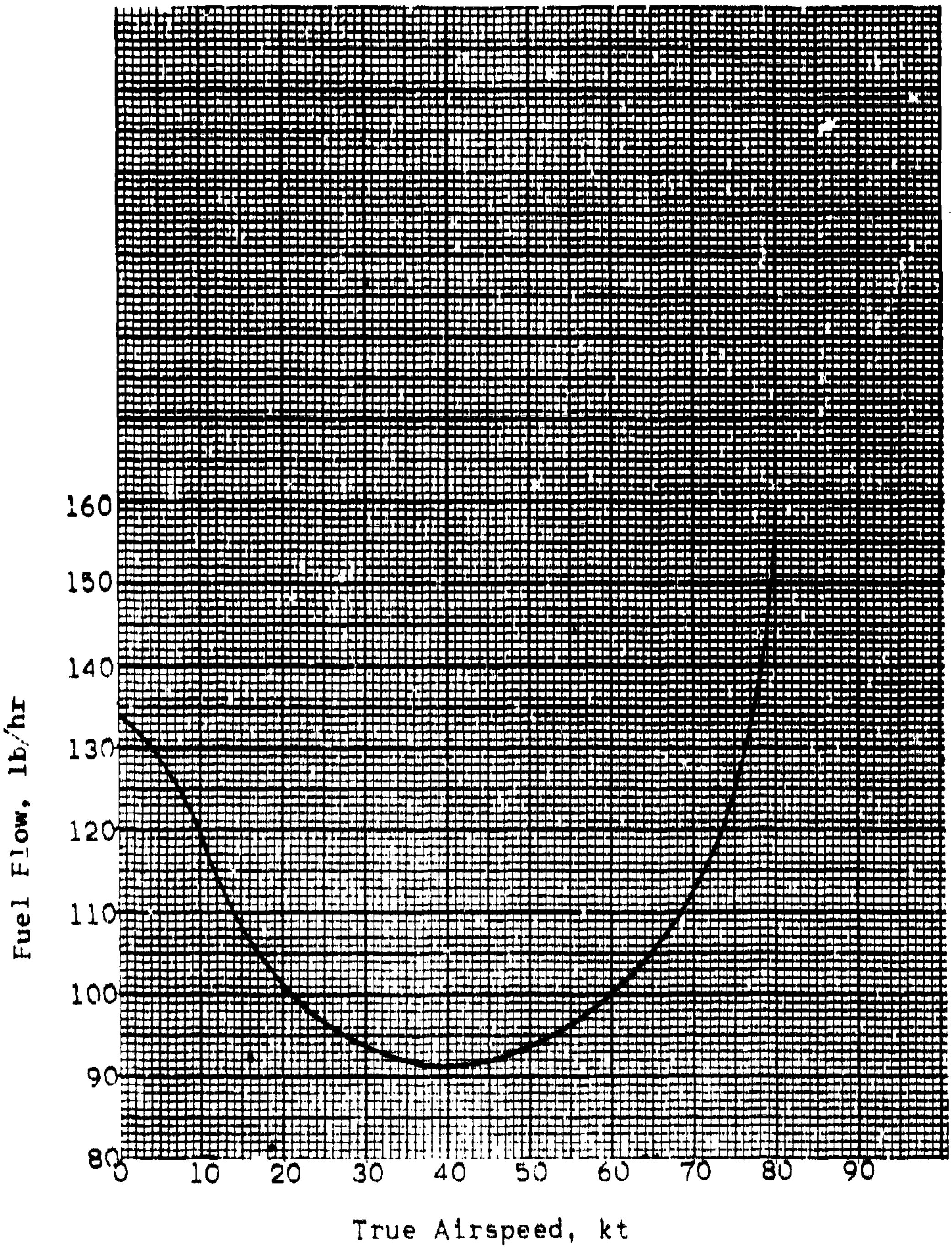



Figure 3.5:5

Model H-23D Helicopter
ASN 57-2987

SPECIFIC RANGE
vs
TRUE AIRSPEED

Gross Weight - 2600 lb
Engine Speed - 3200 RPM

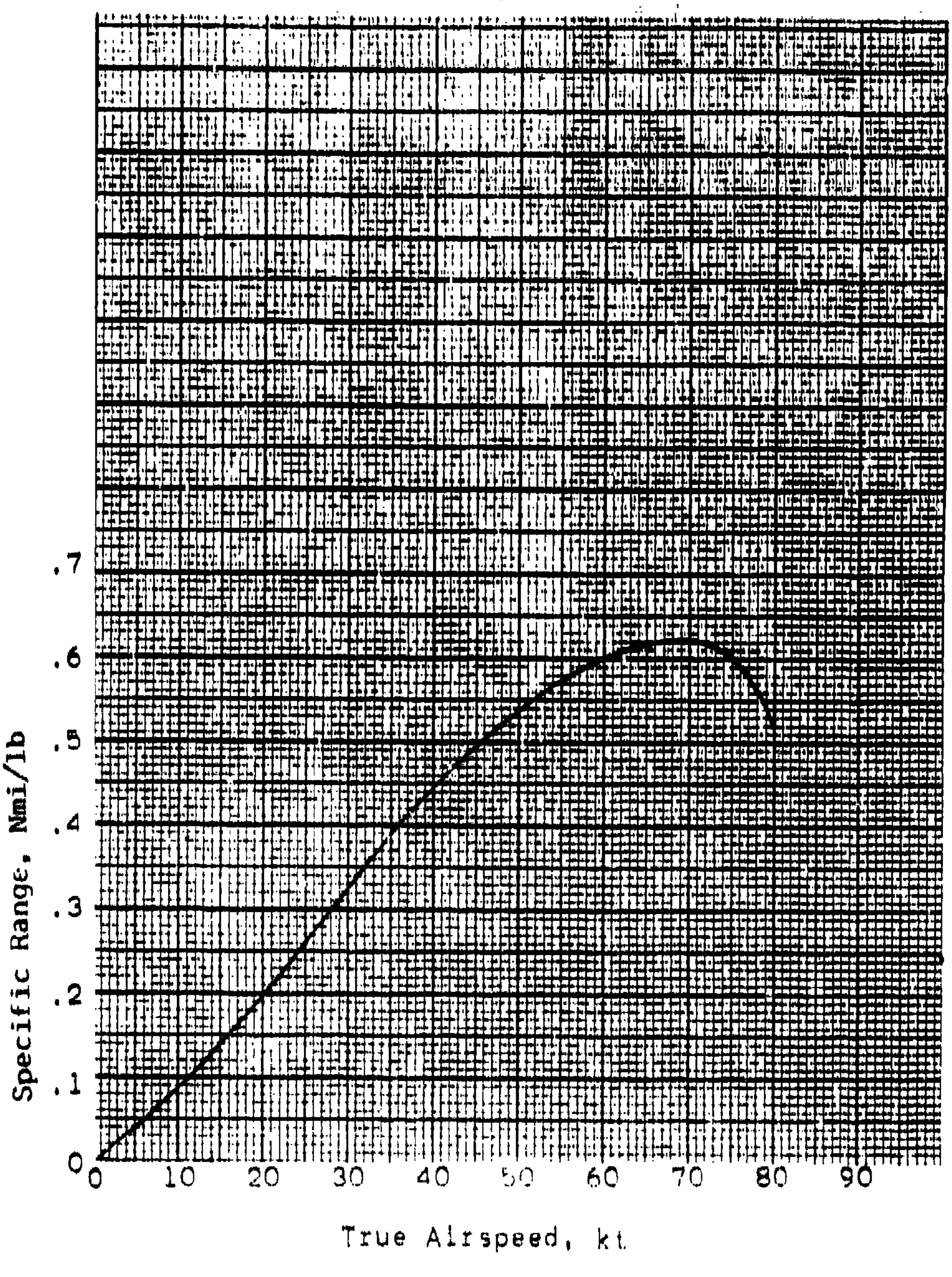



Figure 3.5:6

Model H-23D Helicopter
ASN 57-2987

LEVEL FLIGHT POWER REQUIRED

Tangent Method of Determining
Best Range Airspeed

Engine Speed - 3200 RPM
Pressure Altitude- Sea Level
Gross Weight - 2600 lb

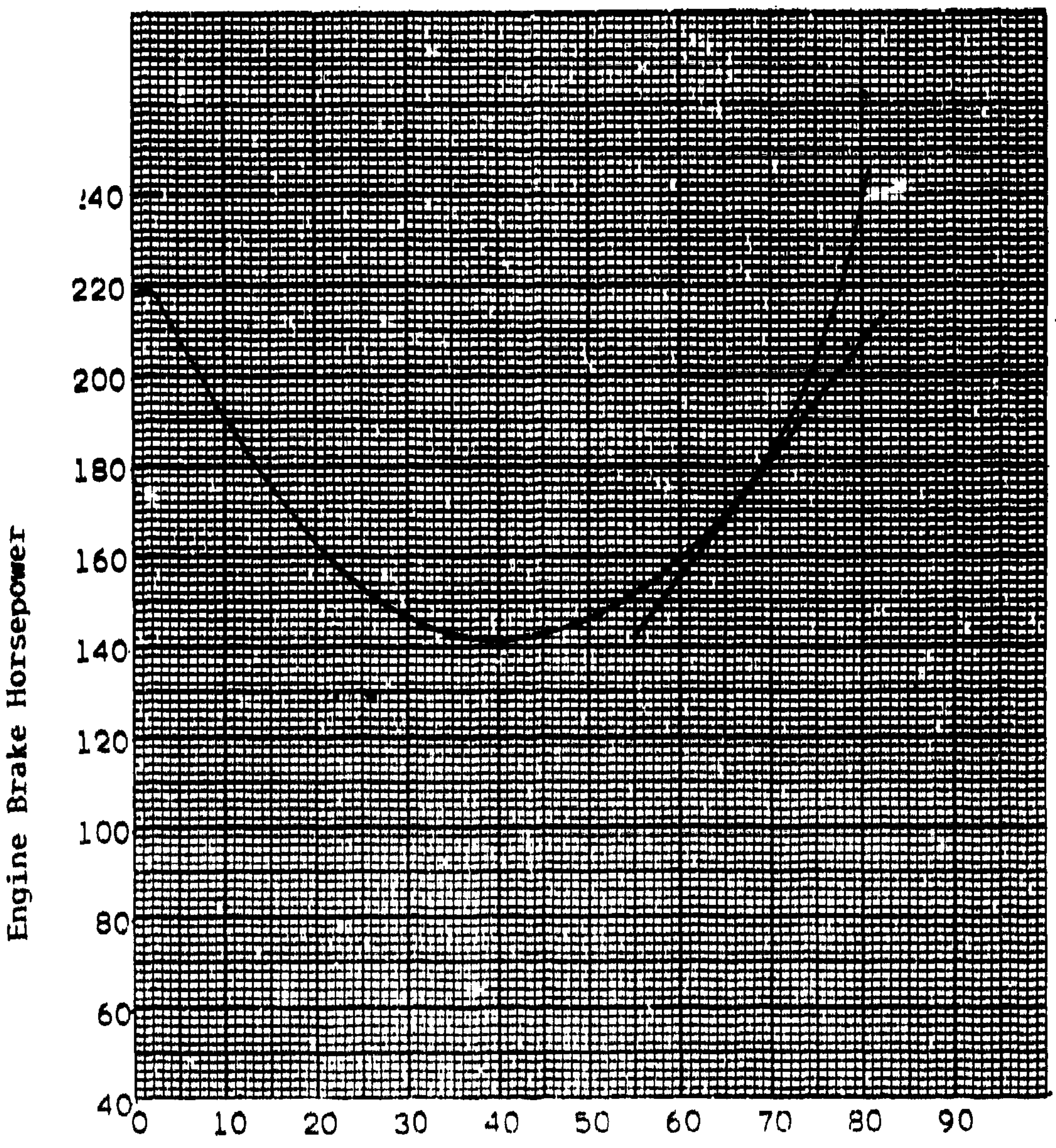



Figure 3.5:7

## GLOSSARY OF TERMS AND SYMBOLS

| | |
|---|---|
| Gross weight, lb | $W$ |
| True airspeed, ft/sec | $V_T$ |
| Density of air, slugs/ft$^3$ | $\rho$ |
| Blade radius, ft | $R$ |
| Rotor angular velocity, radians/sec | $\Omega$ |
| Tip-speed ratio $\approx \frac{V_T}{\Omega R}$ | $\mu$ |
| Thrust coefficient, $\frac{W}{\pi R^2(\Omega R)^2\rho}$ | $C_T$ |
| Rotor shaft horsepower | $BHP_{rs}$ |
| Rotor shaft power parameter, $\frac{550\ BHP_{rs}}{V_T W}$ | P/L |
| Rotor shaft power coefficient, $\frac{550\ BHP_{rs}}{\pi R^2(\Omega R)^3\rho}$ | $C_p$ |
| Rotor figure of merit, $\frac{.707 C_T^{3/2}}{C_p}$ | M |
| Rotor hub height above ground, ft | Z |
| Vertical rate of climb, ft/sec | $V_v$ |
| Standard altitude | $H_s$ |
| Thrust constant, $\frac{(\pi R^4)(.1047)^2\rho_o}{(GR)^2}$ | $K_{CT}$ |
| Power constant $\frac{K_{CT}\ R(.1047)}{(GR)\ 550}$ | $K_{Cp}$ |
| Gear ratio | GR |
| Absolute standard air temperature, °K | $T_{as}$ |

| | |
|---|---|
| Absolute carburetor air temperature, °k | $T_{co}$ |
| Absolute observed outside air temperature, °k | $T_{ao}$ |
| Standard air temperature, °C | $t_{as}$ |
| Outside air temperature, °C; OAT | $t_{ao}$ |
| Carburetor air temperature °C; CAT | $t_{co}$ |
| Manifold pressure, "Hg | MAP |
| Pressure altitude, ft | $H_p$ |
| Chart horsepower | $BHP_{ch}$ |
| Humidity correction factor | $K_h$ |
| Engine brake horsepower | $BHP_e$ |
| Brake horsepower corrected for temperature | $BHP_o$ |
| Brake horsepower corrected for humidity | $BHP_{oh}$ |
| True rate of climb, ft/sec | $R/C_T$ |
| Calibrated airspeed, kt | $V_c$ |
| Normal rated power | NRP |
| Take-off rated power | TOP |

Subscripts: "o" observed

"i" indicated

"s" standard

NOTE:

* represents correction made for instrument error.

# DEFINITIONS OF TERMS

## Section 1

AXES - The three lines representing the intersection of the longitudinal, lateral, and directional planes. The roll (x) axis runs through the aircraft CG, fuselage reference line in the plane of symmetry. The pitch (y) axis is perpendicular to the X-axis in the plane of symmetry. The yaw (z) axis is perpendicular to the X-axis and the plane of symmetry.

BREAK-OUT FORCE - The force required to move the pilot control from trim position prior to obtaining appreciable deflection or control change, (includes friction force).

CONFIGURATION - The disposition of the elements and parameters which affect the flying qualities of the helicopter.

CONTROL - The means by which moments and forces are generated to produce movements of the helicopter about and along its axes.

CONTROL COUPLING - The establishment of moments about a given axis following a control input about another axis.

CONTROL FORCE GRADIENT - The slope of the longitudinal control-force/airspeed plot about a given trim speed, or the slope of the directional control-force/sideslip angle plot about zero sideslip for a given trim speed.

CONTROL HARMONY - The relation of all pilot control forces, movements and response during all maneuvers.

CONTROL POWER - The moment created by a given control displacement for a given configuration.

CONTROL RESPONSE - The response of the helicopter to control input, composed of lag and sensitivity.

CONTROLLABILITY - The quality of an aircraft that is determined by the ease of operating its pilot controls and their effectiveness in achieving changes about and along all axes. Controllability is another name for flying qualities.

DYNAMIC STABILITY, CONTROL-FIXED - The oscillatory characteristics of a helicopter with controls fixed subsequent to a disturbance.

DYNAMIC STABILITY, CONTROL-FREE - The oscillatory characteristics of a helicopter, with power control system inoperative, with controls free subsequent to a disturbance.

DYNAMIC STABILITY - The characteristics of the oscillatory movements of a helicopter in response to its static stability, inertia, and damping tendencies following an initial disturbance from equilibrium.

EFFECTIVENESS - The rate of increase of angular velocity subsequent to lag, in reaction to pilot control application.

EXTERNAL CONFIGURATION - The particular disposition of externally appendant elements to the helicopter at which a flying qualities investigation is to be conducted.

FLYING QUALITIES - Stability and control characteristics and the pilot's impressions of the ease of flying the helicopter on a given mission in steady flight and in maneuvers.

FRICTION FORCE - The force required to overcome the friction of the pilot control system prior to obtaining a pilot control deflection or change.

LAG - The time delay between application of pilot control and start of reaction.

LONGITUDINAL, LATERAL AND DIRECTIONAL PLANES - The longitudinal plane is defined by the roll and yaw axes, the lateral plane by the pitch and yaw axes, and the directional plane by the pitch and roll axes.

MANEUVERABILITY - Same as Controllability but not concerned with pilot effort.

MANEUVERING LONGITUDINAL STABILITY - The variation of normal acceleration with control force.

MASK - A term describing the condition existing when the pilot control force stability is hidden by control system friction.

NEGATIVE DYNAMIC STABILITY - If in response to its static stability, inertia, and damping tendencies a helicopter performs oscillations of increasing amplitude (divergent oscillations), it is said to be dynamically unstable or to possess negative dynamic stability.

NEGATIVE STATIC STABILITY - The development of aggravative moments upon deviation of the helicopter from desired trim attitude with respect to a flight path, or trim velocity with respect to the surrounding air mass.

NEUTRAL DYNAMIC STABILITY - If in response to its static stability, inertia, and damping tendencies a helicopter returns to equilibrium but overshoots and performs continuous oscillations of equal amplitude, it is said to possess neutral dynamic stability.

NEUTRAL STATIC STABILITY - The absence of moments upon deviation of the helicopter from desired trim attitude with respect to a flight path, or trim velocity with respect to the surrounding air mass.

PILOT CONTROL - The means available to a pilot to change control (rudder pedals, cyclic stick, etc).

PILOT CONTROL FORCE - The force required to move the pilot control from the control trim position in order to obtain a response.

PILOT CONTROL MARGIN - The amount of pilot control remaining to produce attitude changes at the aircraft limit speeds.

PILOT CONTROL POSITION - The deflection of the pilot's control from a given reference point, expressed in percent of total travel.

PILOT EFFORT - The energy expended by the pilot in flying the helicopter composed of control force, displacement and displacements per second.

PILOT LATERAL CONTROL MOTION - The variation of pilot lateral control position with changes in trim sideward speed.

PILOT LONGITUDINAL CONTROL MOTION - The variation of pilot longitudinal control position with changes in trim forward speed.

POSITIVE DYNAMIC STABILITY - If in response to its static stability, inertia, and damping tendencies a helicopter returns to equilibrium by performing a single motion (aperiodic or deadbeat damping), or by performing oscillations of decreasing amplitude (convergent oscillations), it is said to be dynamically stable or to possess positive dynamic stability.

POSITIVE PILOT LATERAL CONTROL MOTION - Further pilot lateral control displacement in the direction of trim sideward speed increase.

POSITIVE PILOT LONGITUDINAL CONTROL MOTION - Further pilot longitudinal control displacement in the direction of trim forward speed increase.

POSITIVE STATIC STABILITY - The development of corrective moments upon deviation of the helicopter from desired trim attitude with respect to a flight path, or trim velocity with respect to the surrounding air mass.

PULSE CONTROL INPUT - Displacement of pilot's control a given amount, holding at the displaced position for a specified time interval, and returning to the original position.

SENSITIVITY - The pilot's impression of the combination of the rate of increase of angular velocity and the peak value of angular velocity subsequent to lag, in reaction to pilot control application.

STABILITY - Related to the behavior of an aircraft after it is disturbed slightly from trim.

STABILITY CONFIGURATION - The particular power and collective position setting, rotor speed, CG position, gross weight, airspeed and altitude at which a flying qualities investigation is to be conducted.

STATIC LATERAL AND DIRECTIONAL STABILITY - The development of lateral and directional moments upon deviation of the helicopter from desired trim attitude as evidenced by the variation of pilot's lateral and directional control positions and forces with variation of sideslip angle about zero sideslip angle at a given trim speed.

STATIC LONGITUDINAL STABILITY - The development of longitudinal moments upon deviation of the helicopter from desired trim attitude as evidenced by the variation of pilot's longitudinal control position and control force with variation of aircraft speed about a given trim speed.

STATIC STABILITY - The development of moments upon deviation of the helicopter from desired trim attitude with respect to a flight path, or trim velocity with respect to the surrounding air mass.

STEP CONTROL INPUT - Displacement of pilot's control a given amount and holding at the displaced position.

TRIM - That condition of flight in which the forces and moments on the aircraft are equal to zero, and the forces on the pilot's controls are equal to zero.

TRIM CHANGE - The pilot control forces and/or displacements involved in opposing moments that arise from a configuration change.

TRIMMABILITY - The effort required to trim the pilot's controls and their tendencies to remain fixed at the desired trim condition.

INTRODUCTION

Section 2

4.2.1

In the preceding chapters, basic aerodynamics laws have been developed and have been applied to the determination of the complete helicopter's performance characteristics. The next consideration is how does the helicopter fly, and how well can it be controlled. The ability of the aircraft to fly smoothly at a constant altitude, attitude, and airspeed, to recover from the effects of atmospheric disturbances, and to respond adequately to the controls of the pilot is covered by a group of characteristics called stability and control characteristics, or collectively, with pilot effort considered, flying qualities.

4.2.2

Flying qualities are evaluated at the Naval Air Test Center with the basic mission of the helicopter as a reference datum and in accordance with tests outlined in MIL-H-8501. An aircraft may possess satisfactory flying qualities in general, yet it may be unsuitable for certain missions because of either poor stability or poor control. Maneuverability of two aircraft may be similar where one aircraft has weak stability and weak control and the other has strong stability and strong control, yet the flying qualities of the latter aircraft are superior because of the decreased pilot fatigue factor, and consequently increased controllability.

4.2.3

Consider two aircraft under evaluation, A and B. In aircraft A, where strong control and weak stability are inherent. the pilot is more capable of overpowering the tendencies of the aircraft to stray from the flight path; while in aircraft B, possessing weak control and strong stability, it is necessary for the pilot to move the control quickly and over large displacements to disturb it from its steady stable path. Aircraft A is better suited for a mission requiring short hauls in and out of cramped quarters, while aircraft B is better suited for missions such as AEW or instrument flight.

4.2.4

Discussions concerning flying qualities refer to the following general equations:

(a) Flying Qualities = Stability + Control + Pilot Effort = Controllability

(b) Maneuverability = Stability + Control

(c) Flying Qualities = Maneuverability + Pilot Effort

4.2.5

Considering aircraft A and B above, equation (b) would yield similar results, yet aircraft B would be considered superior from the results of either equation (a) or (c), because of the "Pilot Effort" factor. In unstable aircraft, "Pilot Effort" is higher than in stable aircraft. Consequently for general helicopter requirements, aircraft B is said to possess flying qualities superior to those of aircraft A. However, only when the mission is considered can conclusions be drawn concerning the superiority of one aircraft to another in flying qualities.

## STATIC LONGITUDINAL STABILITY

### Section 3

### 4.3.1 PURPOSE

The purpose of these tests is to determine if the movements of the pilot's longitudinal control involved in changing speed and attitude are logical in direction, and if the helicopter may be trimmed at any desired speed at will. Static longitudinal stability may be indicated in terms of longitudinal control surface position and longitudinal control forces for fixed wing aircraft where elevator measurements are feasible. However, in helicopters, the measurements are made at the pilot's cyclic control; consequently, all "control" terms mentioned in Section 3 refer to "pilot's control".

### 4.3.2 STATIC LONGITUDINAL STABILITY

If an aft longitudinal control displacement and a pull force is required to hold the aircraft in a nose-up attitude, accompanied by a decreased airspeed; and if a forward longitudinal control displacement and a push force is required to hold the aircraft in a nose-down attitude, accompanied by an increase airspeed, the aircraft possesses positive static longitudinal stability about a given trim speed. The slope of the curve of stick force versus airspeed is indicative of the amount of force tending to snap the longitudinal control back to trim upon release.

### 4.3.3

In aircraft where high rotor loads would induce fatigue or produce immovable forces, some measure of control system force elimination must be provided. In eliminating forces from the control system, the designer must be aware of the possibility of eliminating all forces, so that no "feel" is available to the pilot. To prevent this, designers usually install stick-centering devices to provide a small resisting force which is proportional to control displacement from trim. Inasmuch as most helicopters are equipped with control system force eliminators and artificial "feel", the static longitudinal stability investigations are not indicative of the forces on the control surfaces tending to return the aircraft to trim or to displace it farther from trim, but are rather indicative only of the "apparent" stability, which is sensed by the pilot.

4.3.4 METHOD OF TESTS

Static longitudinal stability tests are conducted in the following manner:

Trim the aircraft in steady level flight in a given stability configuration. After recording the longitudinal control position and control force, slowly move the longitudinal control to a position accompanying a lower airspeed than trim speed, maintaining the same collective position and throttle setting as at trim. Record the control position, control force and new trim speed. Then move the longitudinal control to a position accompanying a new airspeed higher than trim, still maintaining the original throttle and collective position. Record the new position, force and airspeed.

Repeat the process for separate trim speeds from hover through Vmax. Then vary elements of the configuration to determine their effects.

Static longitudinal stability data are presented in figures 4.3:1 and 4.3:2.

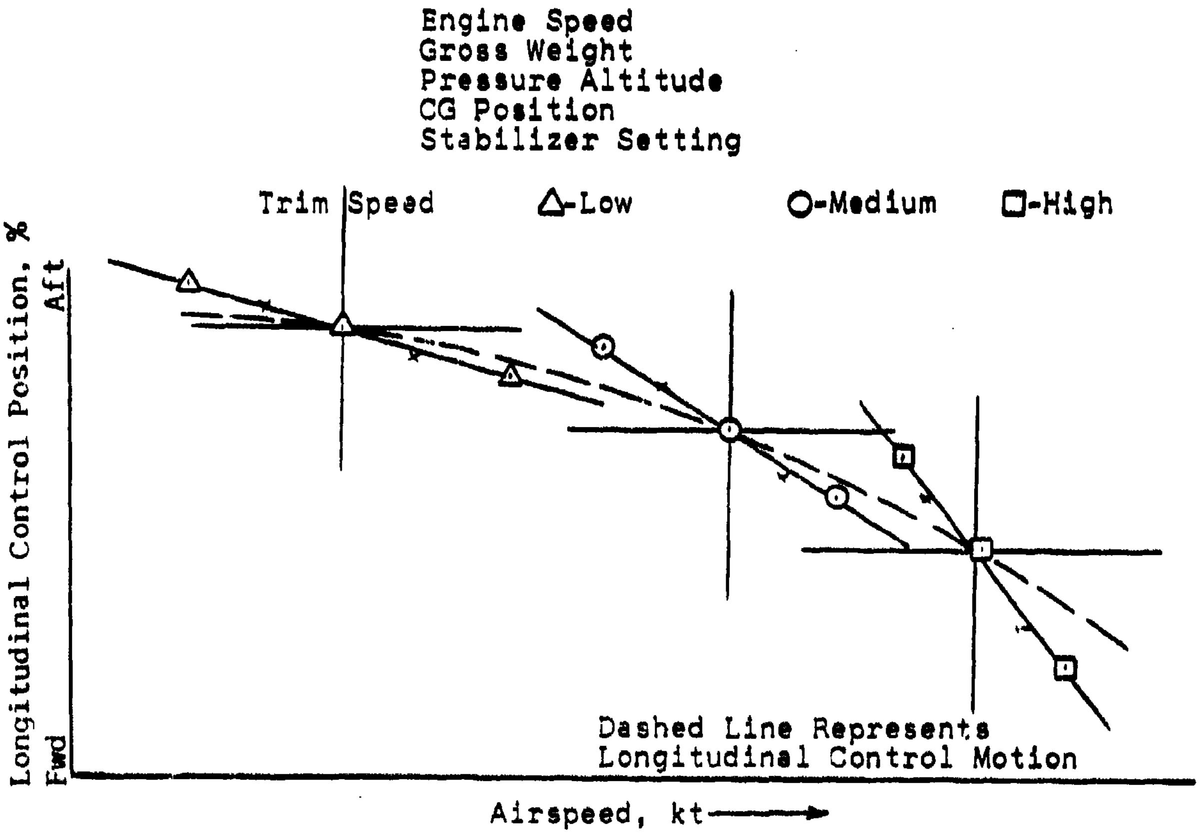



Figure 4.3:1

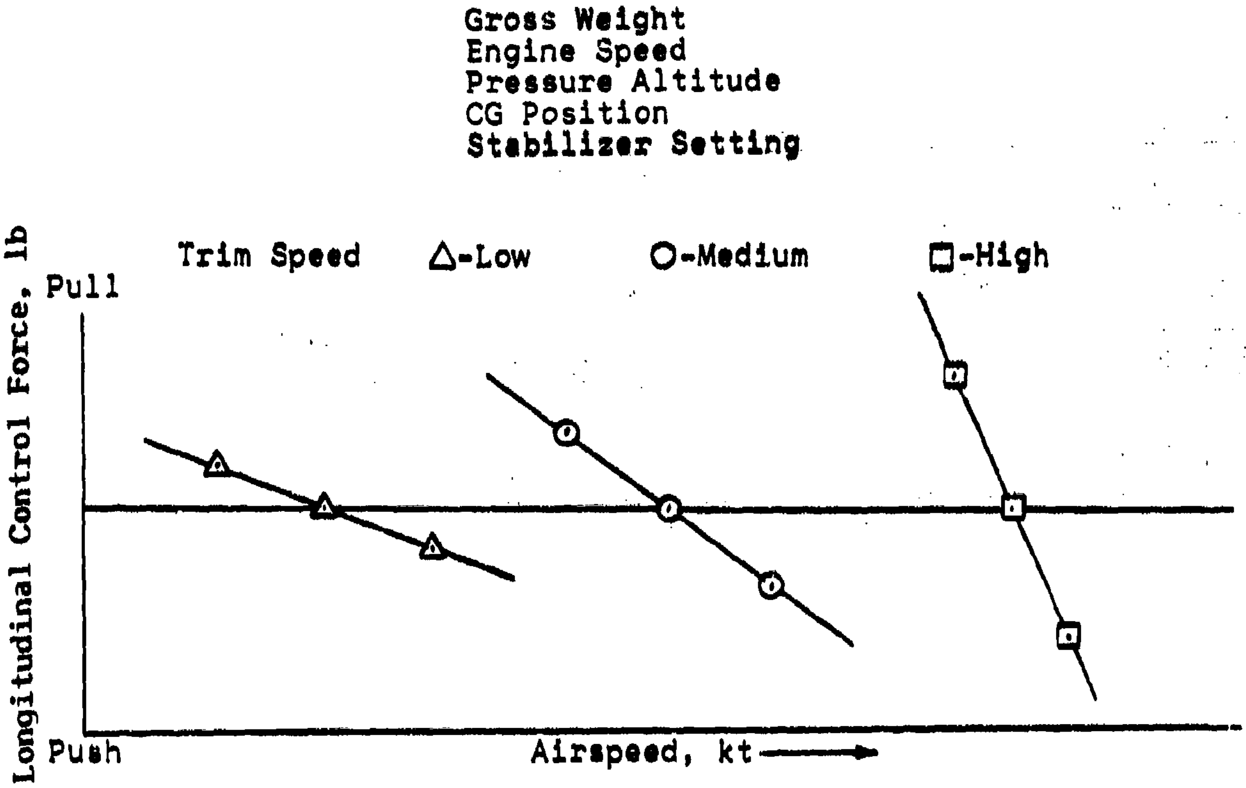



Figure 4.3:2

### 4.3.5 CONTROL MOTION

It should be noted that the plot of longitudinal control position vs airspeed provides data for the control phase of the stability and control tests. Control motion, namely the variation of control position, in trim, with airspeed, may be evaluated by connecting the trim points of the static stability plots (dashed line in figure 4.3:1). Control motion may be obtained in another way by recording longitudinal control positions in level flight, normally accomplished during the level flight phase of the aircraft and engine performance investigation.

### 4.3.6

Sample flight data cards for determination of static longitudinal stability are presented in figures 4.3:3 and 4.3:4.

SAMPLE STATIC LONGITUDINAL

STABILITY FLIGHT DATA CARD

(with automatic instrumentation)

WEATHER

CARD NO. 1

BKNO 21225

| AIRPLANE TYPE | BU. NO. | TIME T.O. / TIME LAND | DATE |
|---|---|---|---|
| HSS-1N | 145672 | 0600 / 0700 | 13 Jan 1959 |

PILOT: Segner

T.O.C.G. 137.5" (fwd) GEAR DOWN % UP %

CONDITION: Effect of Trim Speed on Static Stab.

T.O. GROSS WEIGHT: 12,629

EXTERNAL CONFIGURATION: Drop tanks installed, stabilizer 10° a/c nose up

Static Longitudinal Stability

| CTR No. | $V_i$ | RPM | ALT | FUEL | FORCE | POS | |
|---|---|---|---|---|---|---|---|
| 001 | 30* | 2200 | 1500 | | | | *Trim |
| 002 | 25 | | | | | | |
| 003 | 35 | | | | | | |
| | | | | | | | |
| 004 | 40* | | | | | | *Trim |
| 005 | 35 | | | | | | |
| 006 | 45 | | | | | | |

FLIGHT DATA-PRNC-NATC-312 (Rev. 6-55)

Figure 4.3:3

SAMPLE STATIC LONGITUDINAL

STABILITY FLIGHT DATA CARD

(without automatic instrumentation)

LONGITUDINAL STABILITY AND CONTROL RECORD
PRNC-NATC-418 (Rev. 5-55)

CARD NUMBER 1

| AIRPLANE TYPE | PILOT | XTD-818 |
|---|---|---|
| HSS-1N | Major Segner | 21225 |

| BUREAU NUMBER | T.O. WEIGHT | DATE |
|---|---|---|
| 145672 | 12,629 | 13 Jan 1959 |

C.G. 137.5"(fwd) GEAR DOWN ______ % MAC

TIME T.O. 0600 LAND 0700

EXTERNAL CONFIGURATION

Drop tanks installed, stabilizer 10° a/c nose up

ELEVATOR EFFECTIVENESS

| STAB. CONFIG. | V TRIM | FRICTION | FUEL |
|---|---|---|---|
| 2200 RPM | 30,40,50 | | START 379 |
| | ALT. 1500 | | END 268 |

| TABS | | POWER | SHORT PERIOD | TRIMMABILITY |
|---|---|---|---|---|
| | E | | | |
| | R | | | |
| | A | | | |

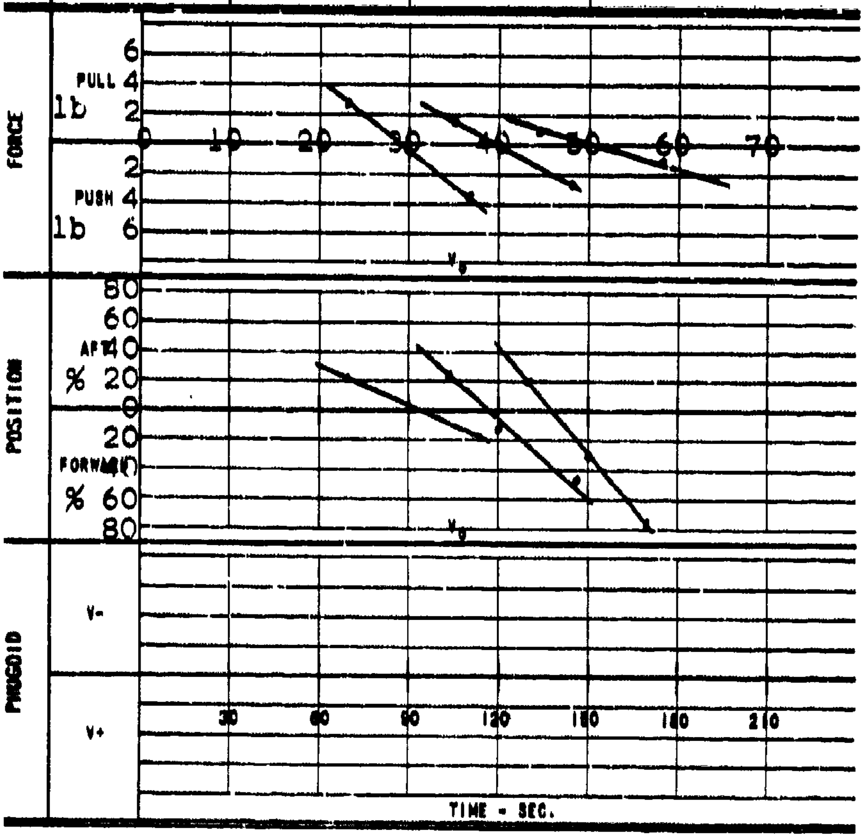



Figure 4.3:4

### 4.3.7 LONGITUDINAL TRIMMABILITY

As indicated in paragraph 4.3.1, one of the main reasons for conducting a static longitudinal stability investigation is to determine its effect on the helicopter's ability to remain at trim. Rather than assume that the static stability data obtained proves that the helicopter will be satisfactory in flight, a more direct method of evaluation is to specifically determine how well or how close it will hold trim speed under various conditions of flight. In addition, an examination of the suitability of the trimming device control system is necessary, i.e., an investigation of the speed of actuation at various airspeeds, the ability to trim precisely without continually overshooting, etc. These tests are combined into one test called "trimmability". It involves the ability to precisely trim the helicopter at a given speed and the helicopter's ability to subsequently remain at that speed. Also included in this test is the damping of small phugoid oscillations about trim which, in rough air, determine partially the helicopter's ability to return to trim speed.

4.4.1

The purpose of a dynamic longitudinal control-fixed stability investigation is to observe the oscillations of the aircraft in response to its positive static longitudinal stability, damping, and inertia, in order to determine its ability to recover with control fixed from gust disturbances.

4.4.2

Dynamic longitudinal control-fixed stability tests are conducted in the following manner: In a given stability configuration the aircraft is suddenly displaced from trim by a sudden longitudinal pulse control input while holding all other controls fixed, and a time history of the ensuing oscillation is recorded. Should the longitudinal oscillations decrease in amplitude, the aircraft is said to be dynamically stable longitudinally; should they increase in amplitude, the aircraft is said to be dynamically unstable longitudinally. Oscillations about the other axes are investigated to determine the degree of control coupling.

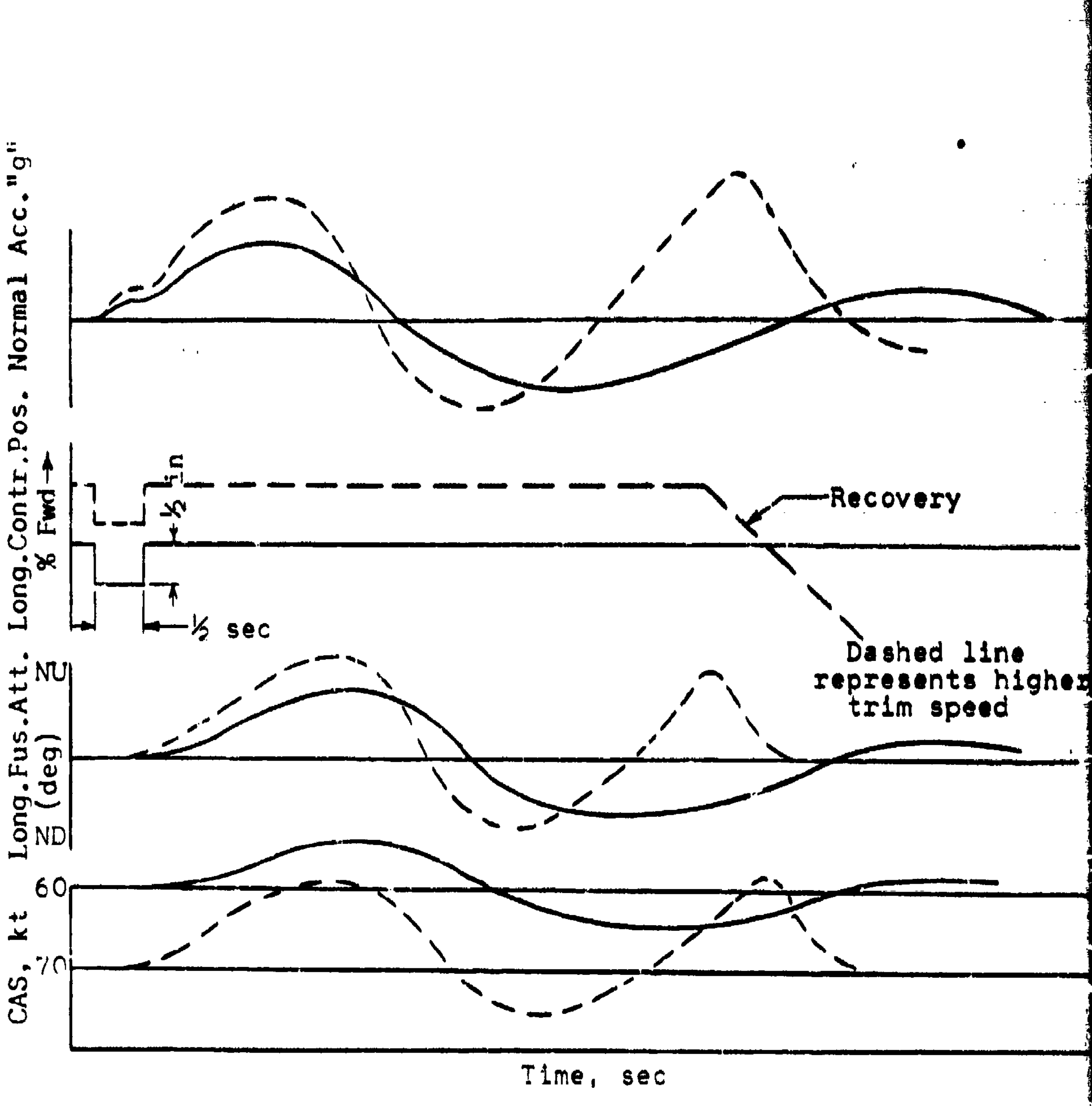
It should be noted that the magnitude of the pilot's control displacement and the magnitude of the time interval in the pulse control input must be carefully chosen so that violent reactions are avoided. A typical pulse control input is: displace control ½ in, hold ½ sec, and return. Typical curves are presented in figure 4.4:1.

4.4.3

A second phase of dynamic longitudinal control-fixed stability testing involves the aircraft's response to a step control input. A typical step control input is: displace the pilot's control 1 in. and hold. Data are presented in the same manner as for the test of 4.4.1.

4.4.4

The first one or two sec of dynamic longitudinal stability tests described in Section 4.4.2 provide information



Normal Acc. "g"
Long. Contr. Pos. % Fwd →
½ in
½ sec
Recovery
Dashed line represents higher trim speed
Long. Fus. Att. (deg)
NU
ND
CAS, kt
60
70
Time, sec


Figure 4.4:1

about the helicopter's "short-period" oscillation characteristics, i.e. its characteristics immediately following a sudden disturbance. Of interest also are the aircraft's "long-period" oscillation characteristics, namely, its motion involving changes in speed and altitude at essentially constant angle of attack when displaced from trim. This differs from the "short-period" oscillation, which occurs at essentially constant speed and altitude while the angle of attack and normal acceleration vary rapidly. The longer the period of the "phugoid", (the long period oscillation), the greater the capabilities of the aircraft for hands-off and instrument flight.

4.4.5

For the conduct of the phugoid tests, the helicopter is trimmed at a specified speed and stability configuration. The airspeed is then increased approximately 10 to 20% (but not into an unstable speed range) above trim and stabilized. The stick is then returned to trim and a time history of airspeed is recorded. Owing to control coupling, it is usually necessary to maintain "wings level" with lateral stick during the oscillation.

4.4.6

Data are presented as follows:

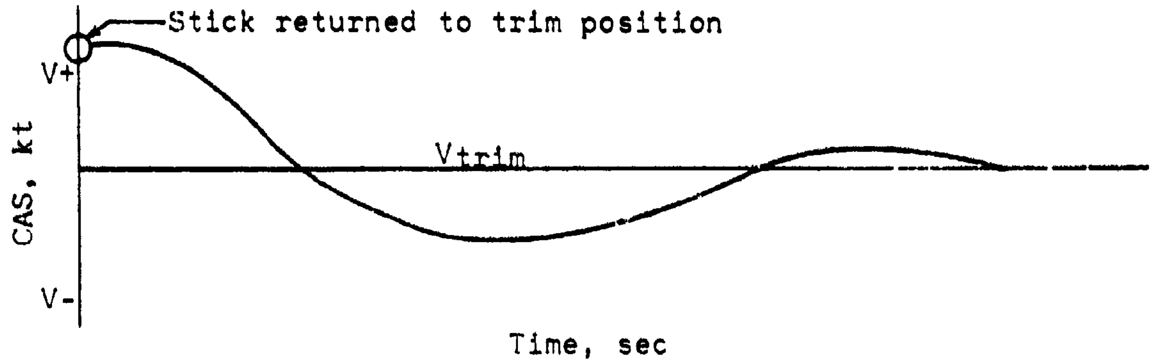



Figure 4.4:2

4.5.1

If the static longitudinal stability tests result in a stable gradient of control force through trim, an investigation may be conducted to determine the dynamic longitudinal control-free stability. Forces in the longitudinal control system, which would normally be resisted by the pilot, are allowed to move the longitudinal control during this test, to determine the possibilities of hands-off flight. These tests are not conducted on aircraft with irreversible power boost systems.

4.5.2

Tests are conducted in the same way as the dynamic longitudinal control-fixed stability tests, with the exception that after the longitudinal control is suddenly displaced and held for a short time, it is released.

4.5.3

Data are presented and obtained as shown in figures 4.5:1 and 4.5:2, respectively.

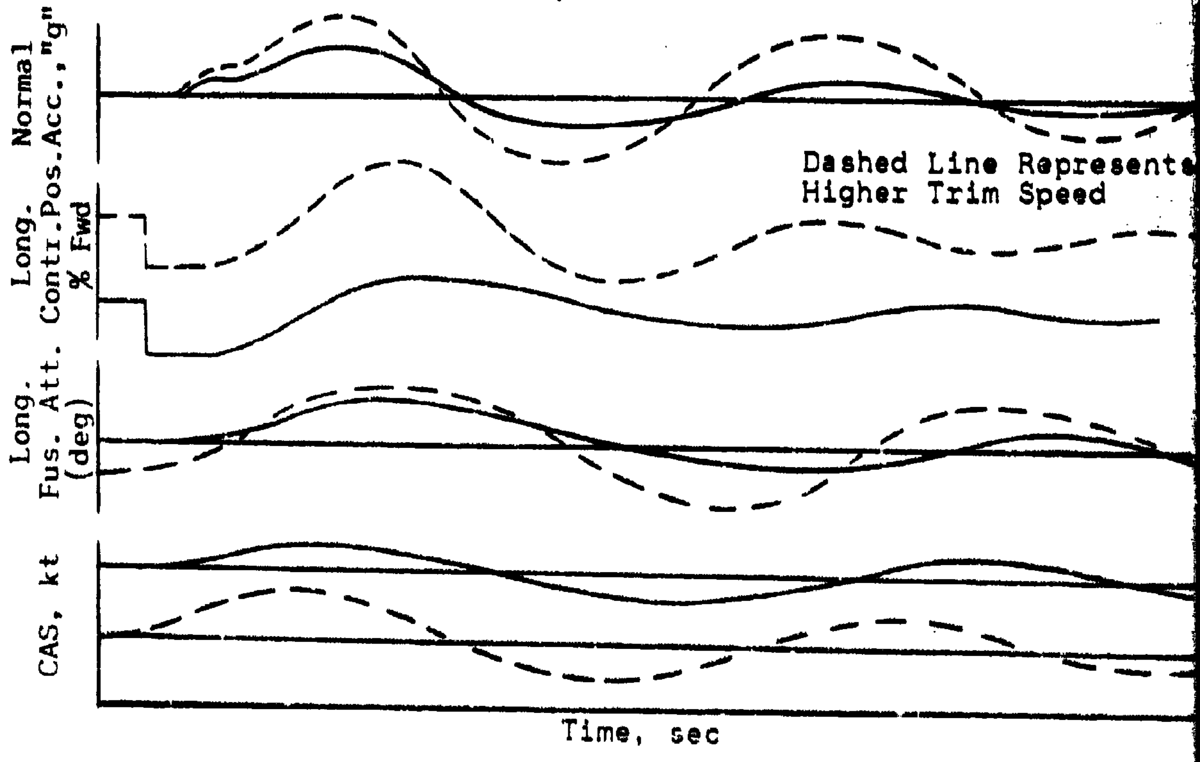



Figure 4.5:1

SAMPLE DYNAMIC LONGITUDINAL CONTROL-FIXED

AND CONTROL-FREE STABILITY FLIGHT DATA CARD

(with automatic instrumentation)

WEATHER

CARD NO. 2

XX 21225

| AIRPLANE TYPE | BU. NO. | TIME T.O. / TIME LAND | DATE |
|---|---|---|---|
| HSS-1N | 145672 | 0600 / 0710 | 14 Jan 1958 |

PILOT: Cdr. McClanan

T.O.C.G. 148.6"(Aft) GEAR-DOWN UP

CONDITION: Dynamic Stability @ 30 kt, 1500 ft

T.O. GROSS WEIGHT: 14,160

EXTERNAL CONFIGURATION: Clean, Stabilizer 10° A/C nose down

Test of (a) Dynamic Longi. Control-Fixed Stability
(b) Dynamic Longi. Control-Free Stability

| TEST | CTR No. | RPM | TRIM SPEED | CONTROL INPUT |
|---|---|---|---|---|
| A | 010 | 2200 | 30 | ½" Pull and hold |
| | 011 | | | 3/4" Pull and hold |
| | 012 | | | 1" Pull and hold |
| | 013 | | | ½" Push and hold |
| | 014 | | | 3/4" Push and hold |
| | 015 | | | 1" Push and hold |
| | 016 | | | ½" Pull, hold ½sec.retur |
| | 017 | | | ½" Push,hold ½sec.return |
| B | 018 | 2200 | 30 | ½"Pull,hold ½sec,release |
| | 019 | | | 3/4"Pull,hold ½sec,rel. |
| | 020 | | | 1"Pull,hold ½sec,release |
| | 021 | | | ½"Push,hold ½sec,release |
| | 022 | | | 3/4"Push,hold ½sec,rel. |
| | 023 | | | 1"Push,hold ½sec,release |

FLIGHT DATA-PRNC-NATC-317 (Rev. 6-55)

Figure 4.5:2

## MANEUVERING LONGITUDINAL STABILITY

### Section 6

4.6.1

In flying qualities evaluations of fixed-wing aircraft, an important phase is concerned with maneuvering longitudinal stability, namely, the variation of elevator force with normal acceleration produced by changes in flight path. These normal accelerations may be produced by a pull-out or a steady turn. Ultimately, mission considerations are applied to the data obtained to determine the acceptability of the maneuvering longitudinal stability.

4.6.2

With the advent of helicopter flying qualities evaluations, maneuvering longitudinal stability tests using fixed-wing methods were attempted by NASA with little success. Control force was unmeasurable and was replaced by longitudinal pilot control position as a pertinent parameter. Steady turns were accomplished, but pilot longitudinal control position could not be measured because of stick shake, high feedback force, and high number of pilot control movements to maintain the steady curved path. Pull-outs comprised dangerous regimes of flight because of the possibility of entering a loop.

4.6.3

In view of the difficulties encountered, maneuvering longitudinal stability, as tested for fixed-wing aircraft, is not investigated in helicopters. The closest approximation to maneuvering longitudinal stability is the test for the response to a step control input, described in Section 4.4.3.

## STATIC LATERAL STABILITY

### Section 7

4.7.1

Rotary wing aircraft exhibit different lateral stability characteristics in sideward flight than in forward flight. Most helicopters possess negative stability at speeds less than 30 kt, both longitudinally and laterally. Since few helicopters are capable of exceeding 30 kt in sideward flight, a lateral stability investigation in sideward flight is impractical; consequently static lateral stability is investigated in forward flight.

4.7.2

The purpose of these tests is to determine if the movements of the pilot's lateral control involved in changing sideward speed and roll attitude are logical in direction. Static lateral stability may be indicated in terms of lateral control position and lateral control force for fixed-wing aircraft, where aileron measurements are feasible. However, in helicopters the measurements are made at the pilot's cyclic control; consequently, all "control" terms mentioned in Section 7 refer to "pilot's control". A desirable flying quality requirement is that right roll tendencies are produced with left sideslips with the lateral control fixed, and conversely for right sideslips. Consequently, an aircraft is said to possess positive static lateral stability if left lateral displacement and force are required to hold the trim airspeed during left sideslip. Similar statements may be made concerning right sideslip.

4.7.3

Static lateral stability are obtained by first establishing unyawed flight, and subsequently noting the increasing or decreasing amount of cross-lateral control required to maintain the sideslip as the aircraft is flown at increasing sideslip angles at the trim power and airspeed, allowing the aircraft to descend. If the amount of lateral control increases to the left with increasing left sideslip angles, and if the lateral force increases to the left with increasing left sideslip angles, the aircraft is said to possess positive static lateral stability to the left. Similar statements can be made concerning right sideslips.

Data are presented and obtained as illustrated in figures 4.7:1, 4.7:2, and 4.7:3.

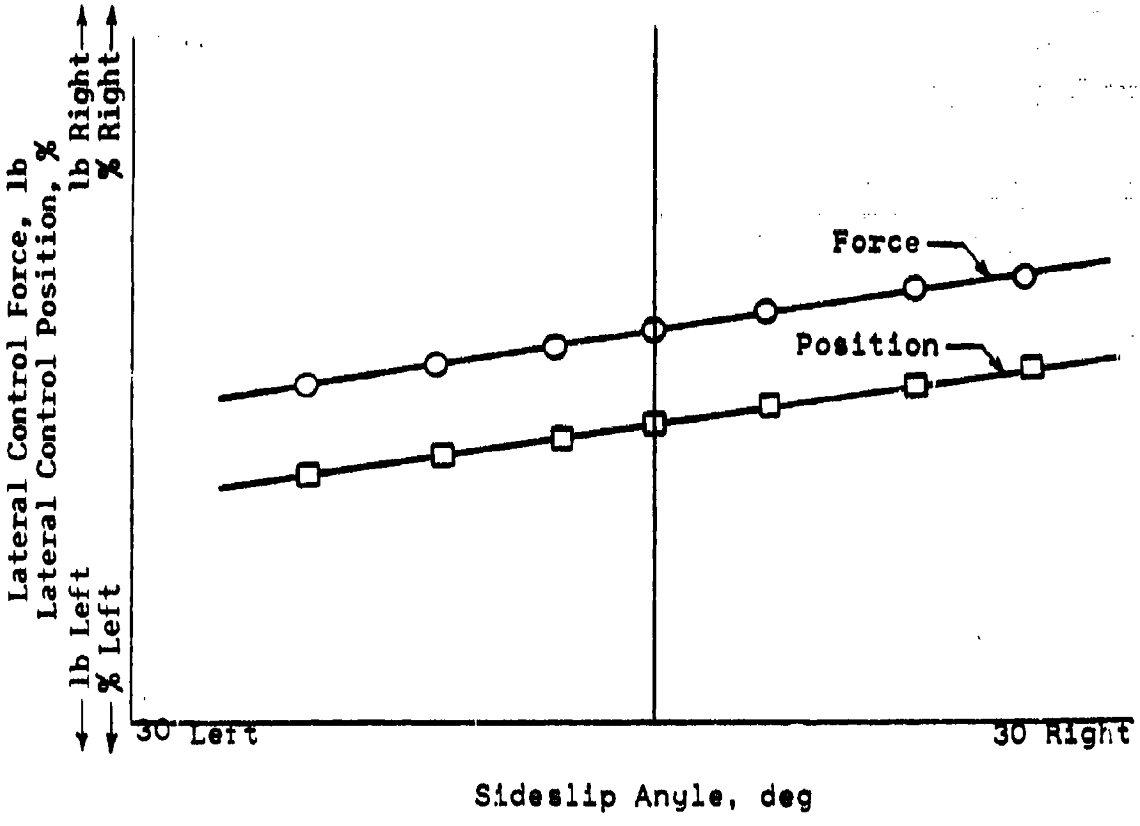



Figure 4.7:1

## SAMPLE STATIC LATERAL-DIRECTIONAL

## STABILITY FLIGHT DATA CARD

## (with automatic instrumentation)

WEATHER

CARD NO. 3

[illegible] 21225

AIRPLANE TYPE HSS-1N | BU. NO. 145672 | TIME T.O. 0600 | TIME LAND 0700 | DATE 15 Jan 1958

PILOT LCDR Sigafoos | T.O.C.G. 137.8" (mid) | GEAR DOWN % UP %

CONDITION Static Lateral-Directional Stab. | T.O. GROSS WEIGHT 13,164

EXTERNAL CONFIGURATION Landing Gear Extended, Stabilizer "Mid"

Test of: (a) Lateral-Directional Stability @50 kt
(b) Lateral-Directional Stability @70 kt

| TEST | CTR No. | RPM | ALT | TRIM SPEED | | DIR. F/POS | LAT. F/POS |
|---|---|---|---|---|---|---|---|
| A | 020 | 2250 | 1500 | 50 | 0° | | |
| | 021 | | | | 5°R | | |
| | 022 | | | | 10°R | | |
| | 023 | | | | 15°R | | |
| | 024 | | | | 20°R | | |
| | 025 | | | | 5°L | | |
| | 026 | | | | 10°L | | |
| | 027 | | | | 15°L | | |
| | 028 | | | | 20°L | | |
| | | | | | | | |
| B | 029 | 2250 | 1500 | 70 | 0° | | |
| | 030 | | | | 5°R | | |
| | 031 | | | | 10°R | | |
| | 032 | | | | 15°R | | |
| | 033 | | | | max. rudder | | |
| | 034 | | | | 5°L | | |
| | 035 | | | | 10°L | | |
| | 036 | | | | 15°L | | |
| | 037 | | | | max. rudder | | |

FLIGHT DATA-PRNC-NATC-512 (Rev. 6-51)

Figure 4.7:2

## SAMPLE STATIC LATERAL-DIRECTIONAL

## STABILITY FLIGHT DATA CARD

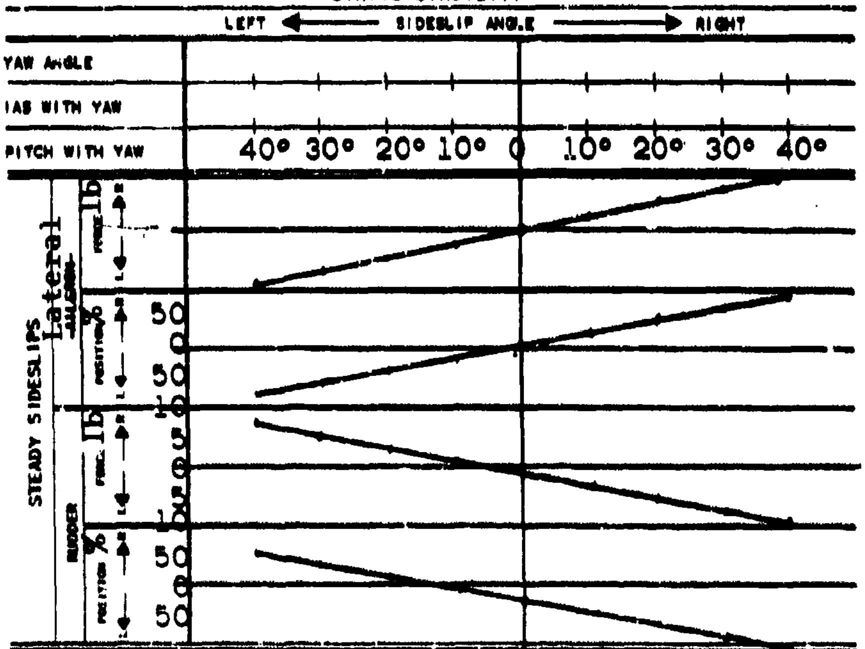
## (without automatic instrumentation)

**LATERAL - DIRECTIONAL STABILITY**
PRNC-NATC-632 (Rev. 10-57)

CARD NO. 4

| AIRPLANE | PILOT | DATE | [illegible] |
|---|---|---|---|
| HSS-1N | McConnell | 16 Jun '58 | 21225 |
| BUNO 145672 | T O. WEIGHT 12,202 | C.G. 137.2"(mid) | |
| STAB. CONFIG. | TRIM CONDITION 50 IAS 1500' ALT. TABS ___ A ___ R ___ E | | |
| EXTERNAL CONFIG. Stabilizer "mid" gear extended | TIME BEGIN 0800 END 0900 | POWER SETTINGS 2200 RPM | |
| RUDDER EFFECTIVENESS | | AILERON EFFECTIVENESS | |

STATIC STABILITY




| MAXIMUMS | RUDDER L ___ % R ___ % | AILERON L ___ % R ___ % | ANGLE OF BANK L ___ % R ___ % |
|---|---|---|---|

REMARKS

Figure 4.7:3

# DYNAMIC LATERAL STABILITY

## Section 8

4.8.1

The purpose of the dynamic lateral stability investigations is to observe the rolling oscillations of the aircraft in response to its positive static lateral stability, damping, and inertia, in order to determine its ability to recover from lateral gust disturbances.

### 4.8.2 DYNAMIC LATERAL CONTROL-FIXED STABILITY

If the static lateral stability tests result in a stable gradient of lateral control position with respect to sideslip angle, an evaluation of the dynamic lateral control-fixed stability may be conducted. From trimmed flight at a given forward speed, the aircraft is given a pulse lateral control displacement, while maintaining rudder, throttle, and collective fixed. The ensuing oscillation is observed to note if the lateral fuselage attitude time history converges or diverges. If the lateral fuselage attitudes converge, the aircraft is said to possess positive dynamic lateral control-fixed stability. During this test, no effort is made to counteract the directional oscillations induced, and measurements are made to determine the roll to yaw ratio. Typical curves are shown in figure 4.8:1.

### 4.8.3 DYNAMIC LATERAL CONTROL-FREE STABILITY

If the static lateral stability tests result in a stable gradient of lateral control force with respect to sideslip angle, an investigation may be conducted to determine its dynamic lateral control-free stability. From trimmed flight, at a given forward speed, the aircraft is given a step lateral control input which is held a short time and subsequently released. The ensuing oscillation is observed to note if the time histories of lateral fuselage attitude and lateral control position converge or diverge. If the oscillations converge, the aircraft is said to possess positive dynamic lateral control-free stability. These tests are not conducted in aircraft with irreversible power boost systems. Typical curves are shown in figure 4.8:2.

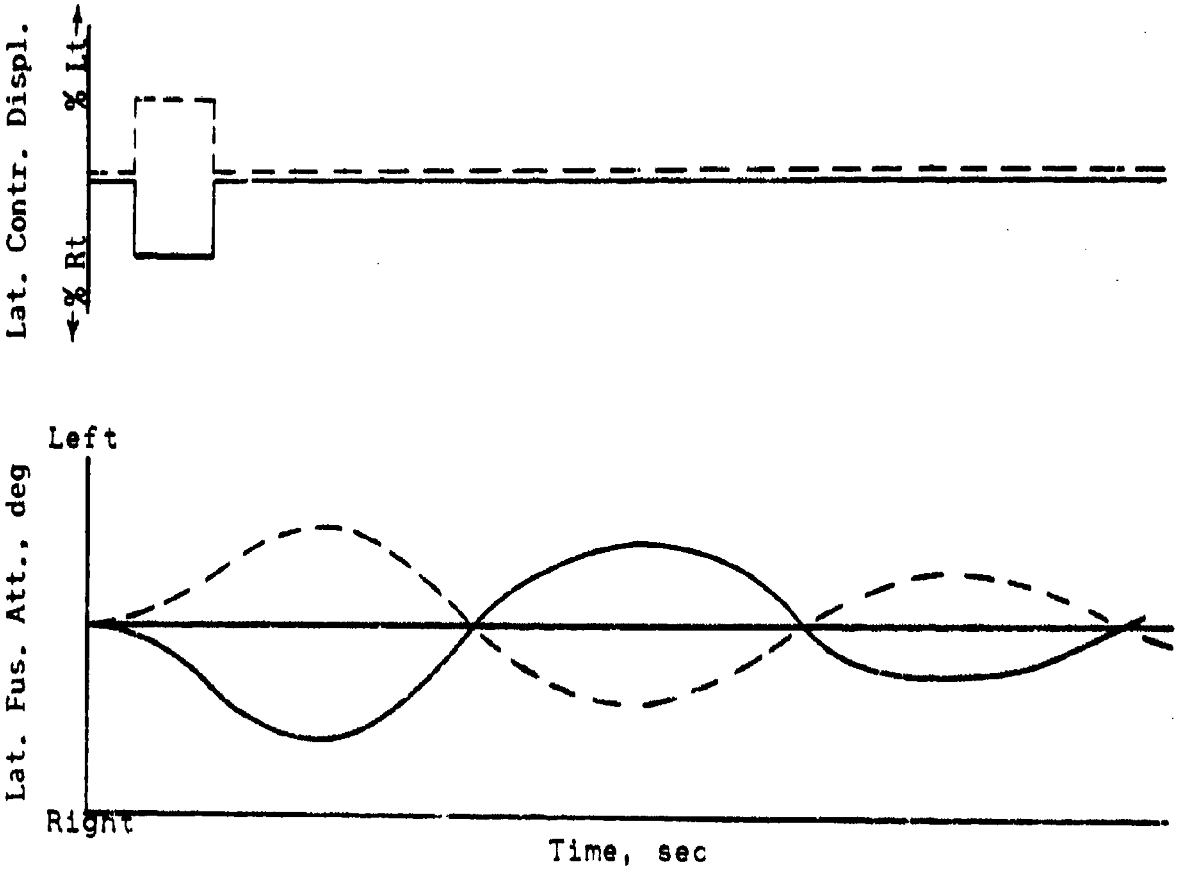



Figure 4.8:1

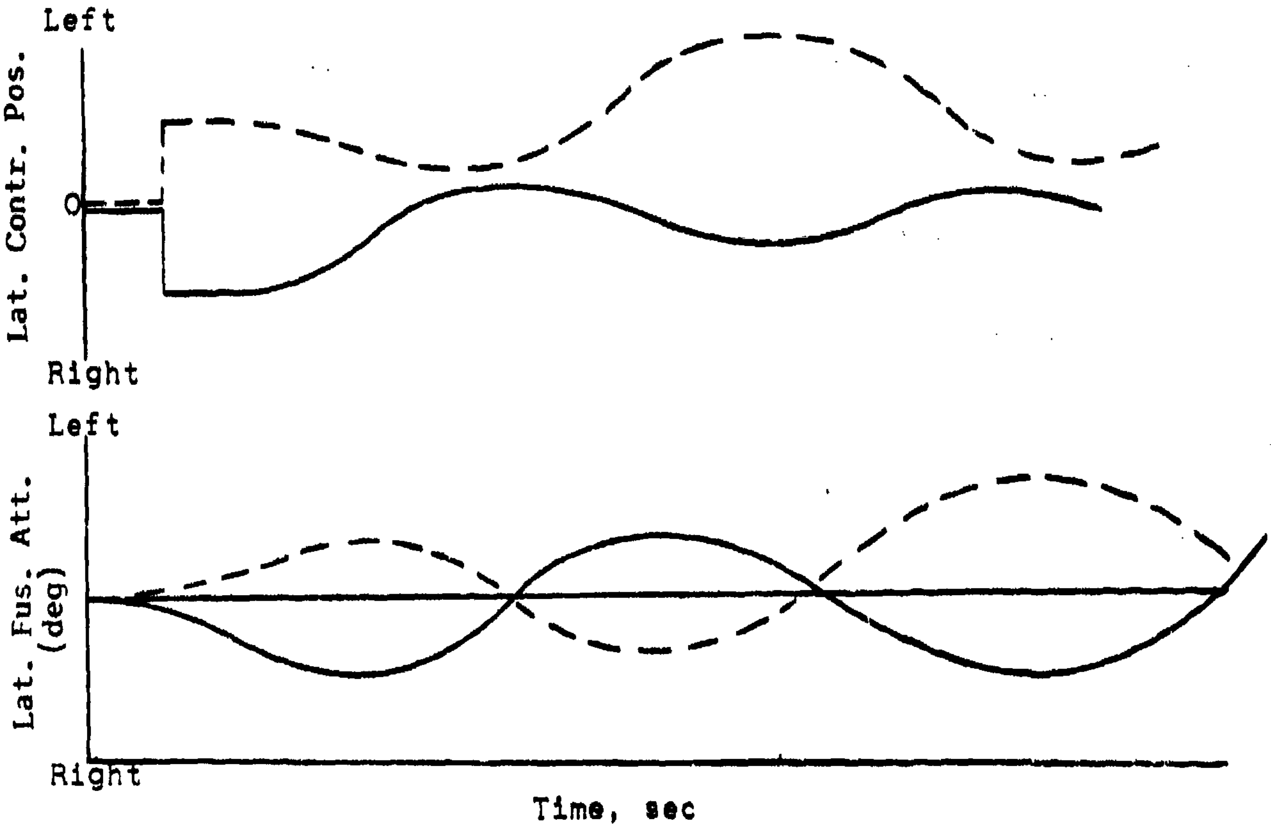



Figure 4.8:2

## STATIC DIRECTIONAL STABILITY

### Section 9

4.9.1

The purpose of these tests is to determine if the movements of and forces on the pilot's rudder pedals involved in changing heading are logical in direction. Static directional stability for fixed-wing aircraft may be indicated in terms of directional control position and directional control force where rudder measurements are feasible. However, in helicopters, the measurements are made at the pilot's pedal; consequently, all "control" terms mentioned in Section 9 refer to "pilot's control".

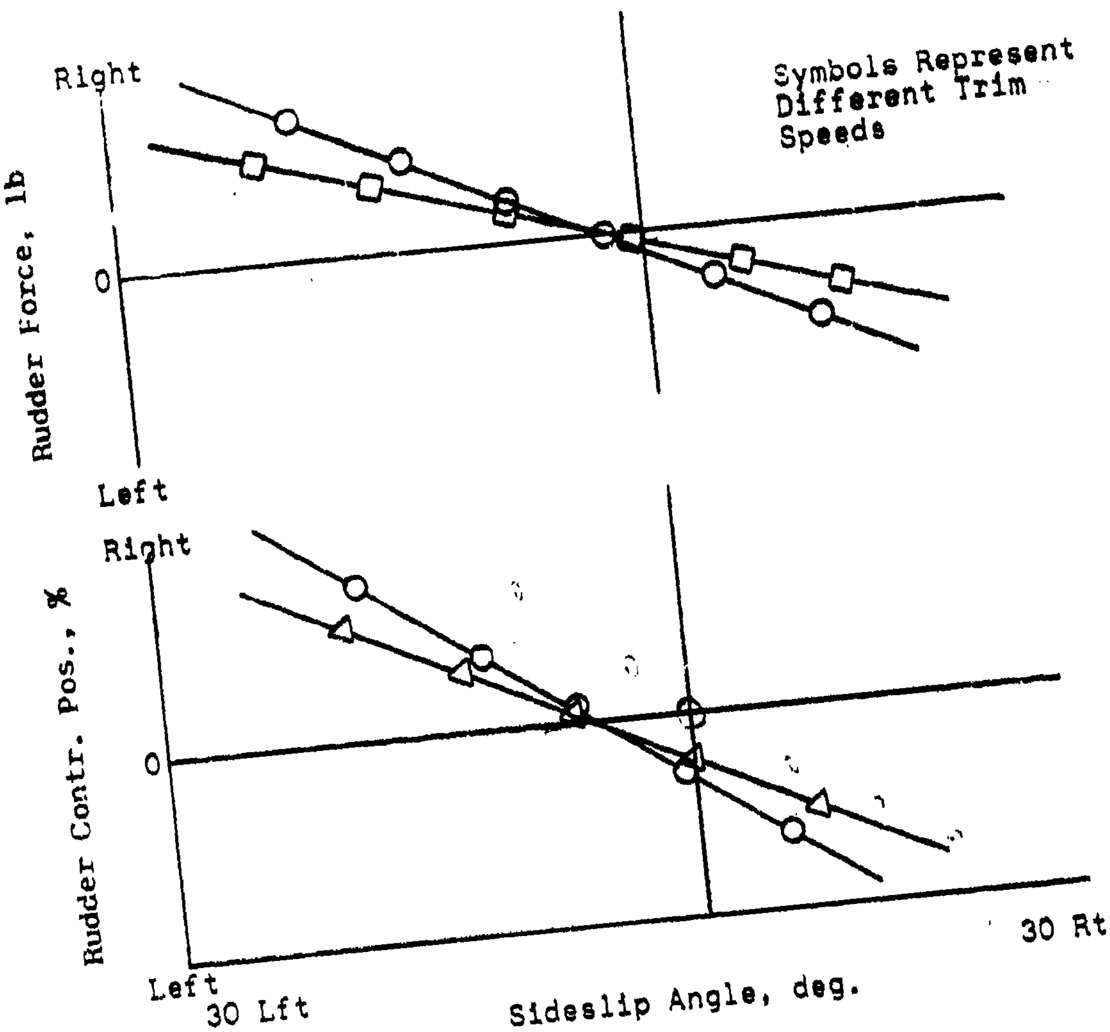
4.9.2 STATIC DIRECTIONAL CONTROL-FIXED STABILITY

If an increase in right rudder displacement and right rudder force is required to hold the aircraft in an increased left sideslip, and an increase in left rudder displacement and left rudder force is required to hold the aircraft in an increased right sideslip, the aircraft possesses positive static directional stability.

4.9.3

Static directional stability tests are conducted in the following manner:

Stabilize at a given trim speed in forward flight and record airspeed, rudder position, rudder force, and sideslip angle. Holding airspeed, collective and throttle constant, obtain sideslip angles to right and left allowing the aircraft to descend, recording rudder position, airspeed, rudder force, and sideslip angle for each new angle. If more right rudder position and force are required for increasing left sideslip angles, the aircraft is said to possess positive static directional stability to the left. Similar statements may be made concerning right sideslip angles. Typical curves are shown in figure 4.9:1.



Symbols Represent Different Trim Speeds
Right
Rudder Force, lb
0
Left
Right
Rudder Contr. Pos., %
0
Left
30 Lft
30 Rt
Sideslip Angle, deg.


Figure 4.9:1

4.10.1

The purpose of the dynamic directional stability investigations is to observe the yawing oscillations of the aircraft in response to its static directional stability, damping, and inertia, in order to determine its ability to recover from directional gust disturbances.

4.10.2 DYNAMIC DIRECTIONAL CONTROL-FIXED STABILITY

If the static directional stability tests result in a stable gradient of rudder displacement with respect to sideslip angle, an investigation may be conducted to determine the dynamic directional control-fixed stability. With the aircraft in steady level flight at a given trim speed, displace the rudder pedal a given amount in either direction, hold a short time, and return, maintaining constant throttle, collective and cyclic. Obtain time histories of sideslip angle, rudder position, airspeed, RPM, and altitude. If the ensuing sideslip oscillation converges, the aircraft is said to possess positive dynamic directional control-fixed stability.

Data are presented as follows:

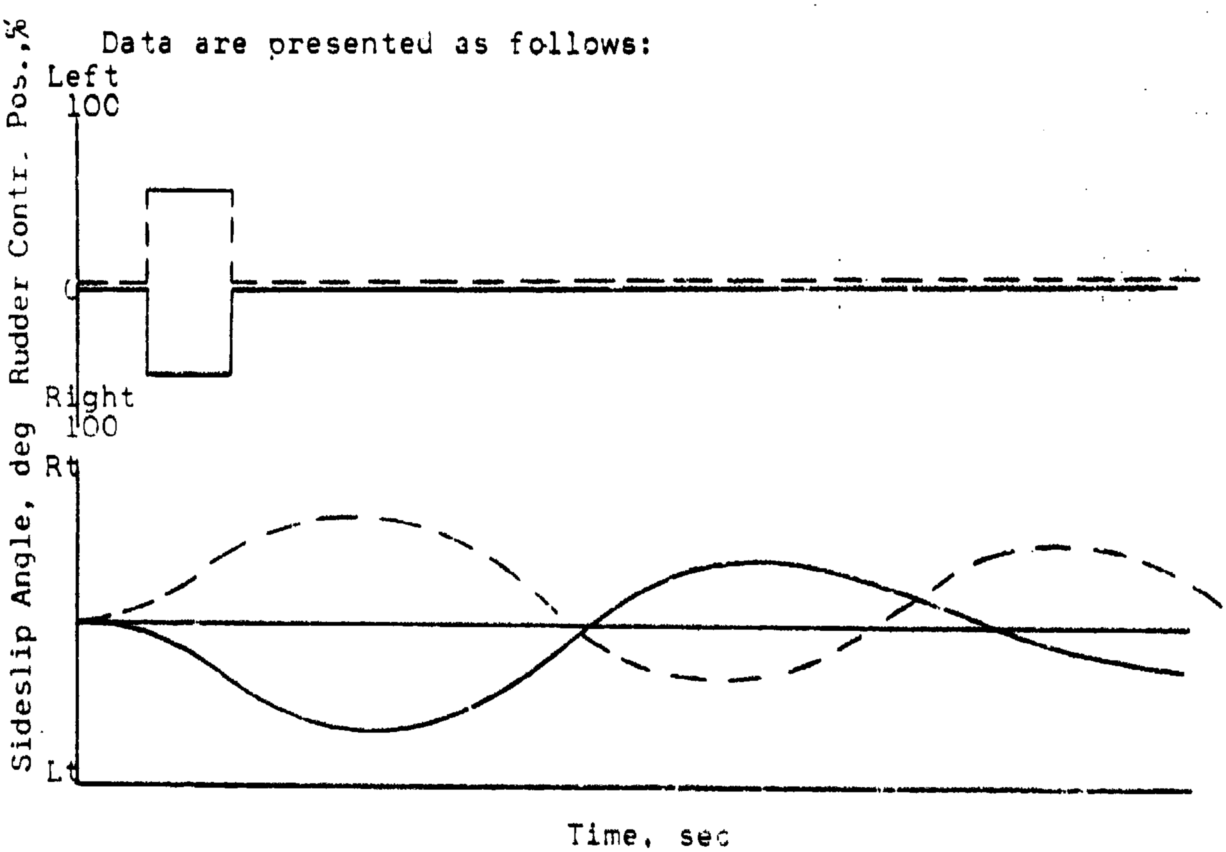



Figure 4.10:1

### 4.10.3 DYNAMIC DIRECTIONAL CONTROL-FREE STABILITY

If the static directional stability tests result in a stable gradient of rudder force with respect to sideslip angle, an investigation may be conducted to determine the dynamic directional control-free stability. Tests are conducted in the same manner as the dynamic directional control-fixed stability investigation, with the exception that in these tests, the rudder pedal is displaced, held a short time, and released. If the ensuing oscillations of sideslip angle, rudder pedal position, airspeed, and RPM are convergent, the aircraft is said to possess positive dynamic directional control-free stability. These tests are not conducted on aircraft with irreversible power control systems.

Data are presented as follows:

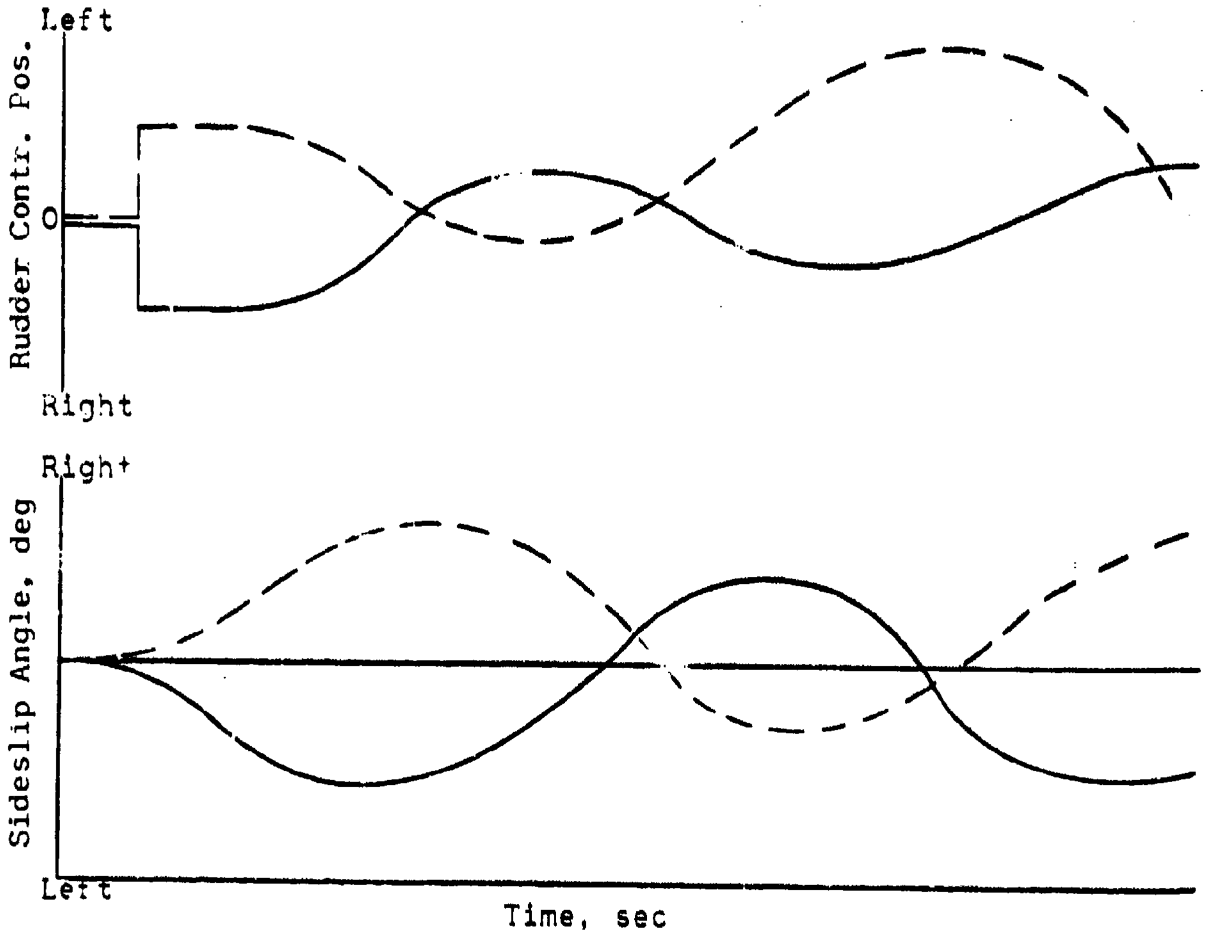



Figure 4.10:2

4.11.1

The control investigation is conducted to determine the ease of operation of the controls and their ability to produce movement of the helicopter about and along its axes. The ease of operation of the controls may be determined by measuring friction and breakout forces and control harmony. The ability of the controls to produce movement of the helicopter about and along its axes may be evaluated by tests for control response and control motion.

4.11.2

The control response of a helicopter may be defined by describing two components: (a) the lag, and (b), the sensitivity. At trimmed flight conditions at a given stability configuration, the helicopter is subjected to a step control input of specified duration and displacement, to obtain a time history of the response. The time history is plotted as follows:

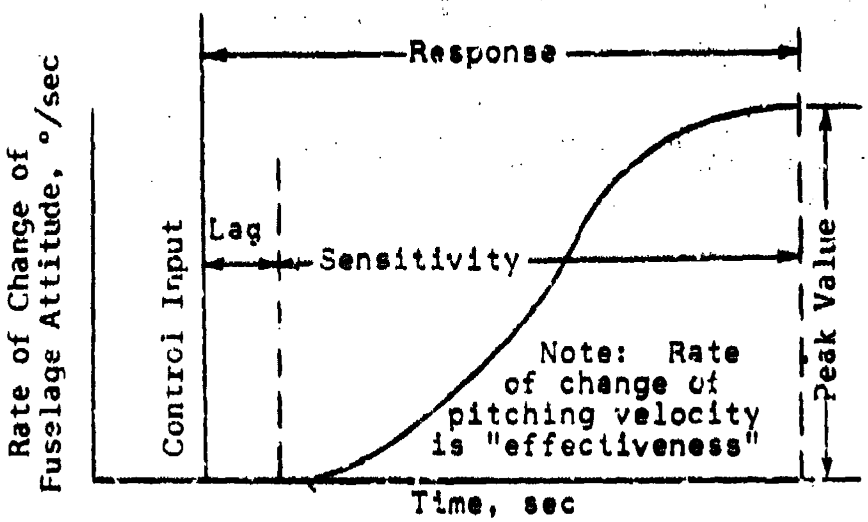



Figure 4.11:1

4.11.3

Response characteristics are investigated through a representative range of stability configurations and a table is prepared as follows:

CONTROL RESPONSE

Control Input: Longitudinal and Lateral, 1" Pull and Hold
Directional: 3" Kick and Hold

Longitudinal

| Regime of Flight | Trim Speed kt | Lag sec | Effectiveness °/sec/sec | Peak Rate °/sec |
|---|---|---|---|---|
| Level Flight | | | | |
| Autorotation | | | | |
| Climb | | | | |

Lateral

| Regime of Flight | Sideward Speed kt | Lag sec | Effectiveness °/sec/sec | Peak Rate °/sec |
|---|---|---|---|---|
| Right Sideward | | | | |
| Left Sideward | | | | |
| Autorotation | | | | |

Directional

| Regime of Flight | Trim Speed kt | Lag sec | Effectiveness °/sec/sec | Peak Rate °/sec |
|---|---|---|---|---|
| Right Sideslip | | | | |
| Left Sideslip | | | | |
| Autorotation | | | | |

## 4.11.4 CONTROL MOTION AND CONTROL MARGIN

Control motion tests are conducted to determine the variation of the pilot's longitudinal control position with trim speed; and lateral control position with sideward speed. It should be noted that these tests differ from static stability tests, wherein the variation of pilot's control position with speeds displaced from a given trim speed is determined. The distinction between the two evaluations is shown in figure 4.3:1 and is discussed in Section 4.3.4.

From a pilot's standpoint, a positive gradient of control position with trim speed is not as important a flying quality as a positive static stability characteristic, although a positive gradient improves trimmability. Consequently, the results of a control motion investigation have as their main objective the determination of control margin. Adequate control margin is considered equivalent to either 10% of the total control travel, or that margin sufficient to produce 10% of the maximum pitching acceleration. $V_{max}$, if not limited by power available or blade stall, is limited by control margin.

## 4.11.5 VERTICAL CONTROL

Tests are conducted on helicopters to determine vertical control characteristics. The aircraft is hovered in calm wind conditions at a constant engine speed and time histories of collective control position and height are obtained. A desirable vertical control characteristic requires that the height above the ground remain within ± 1 ft of the original height, with movements less than ± ½ in. of the collective control stick.

## 4.11.6 CONTROL HARMONY

"Harmony of control" is achieved in a helicopter when the forces and effectiveness of the controls are such that all normal maneuvers can be performed without the pilot being aware of using markedly different effort on any one of the controls required to execute the maneuver. Because a pilot is capable of applying considerable force with his legs, the rudder forces should be higher than those he exerts with his hand. Further, his capability in the longitudinal plane is greater than that in the lateral direction. For th' reason, "harmony" will be achieved when the lateral forces equired are less than the longitudinal requirements. To consider the pilot's longitudinal and lateral capability further, he is able to apply pull forces of greater magnitude than he can push. This is in keeping with his and the helicopter's tolerance of negative and positive accelerations. In a lateral direction the pilot's physical conformation permits a greater force to be applied to the left than to the right. Actual values of the ratio of the longitudinal, lateral and rudder forces have been established. These values are not realistic unless applied to a specific maneuver, because the deflections (and consequently the forces) are a function of the stability and control characteristics of the helicopter. Because of this it is considered that the ideal harmony ratio be applicable to only one maneuver, the rolling pullout.

4.11.7

Another facet of the control harmony investigation is the consideration of the relationship of friction force to breakout force in a given control. In an aircraft where friction force is low and breakout force is relatively high, the pilot encounters unexpectedly high forces when executing maneuvers, and eventually resorts to the aggravating technique of continuously re-trimming during maneuvers. Consequently, the control harmony investigation should include tests to determine the ratio of friction force to breakout force.

4.11.8

The control harmony of a helicopter will be most noticeable during the period in which the pilot is checking out in the helicopter. For this reason, control harmony should be evaluated in the early stages of the flying qualities test program, and repeated during the latter portion of the project to check the pilot's ability to become accustomed to the ratio of forces present.