CR-112153

ROTOR SYSTEMS RESEARCH AIRCRAFT PREDESIGN STUDY

FINAL REPORT
VOLUME II
CONCEPTUAL STUDY REPORT

by Steven A. Schmidt, Arthur W. Linden, et al.

SIKORSKY REPORT NO. SER 50775

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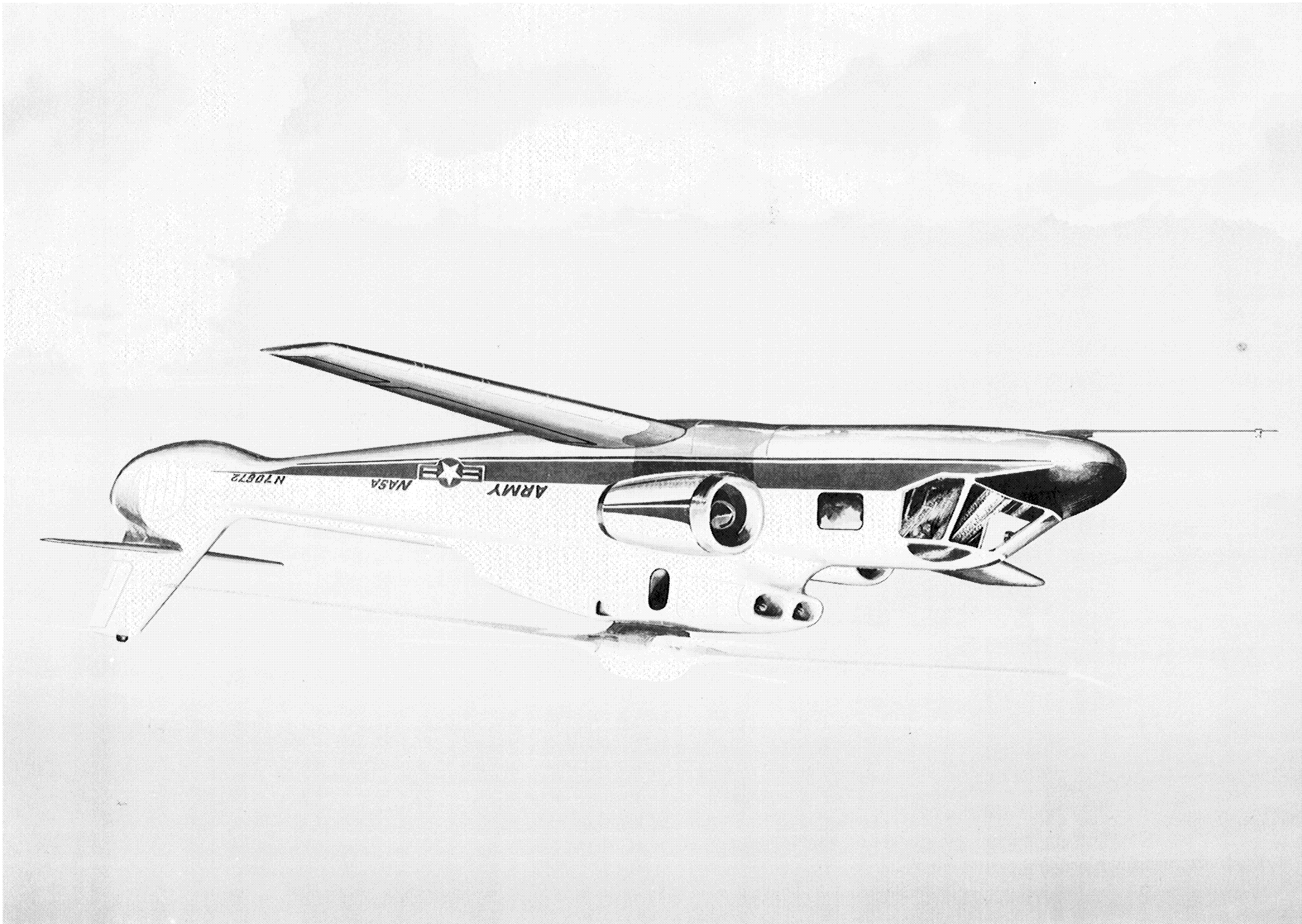
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
AND
UNITED STATES ARMY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FOREWARD

This document was prepared by Sikorsky Aircraft, a Division of United Aircraft Corporation, Stratford, Connecticut, under Contract NAS1-11228 to the National Aeronautics and Space Administration and the U.S. Army. It is subdivided into five volumes as follows:

Volume I	Summary and Conclusions
Volume II	Conceptual Study Report
Volume III	Predesign Report
Volume IV	Preliminary Draft Detail Specification
Volume V	Development Plan Report

The report covers work conducted during the period December 1971 - July 1972.

SUMMARY

The Rotor System Research Aircraft (RSRA) Predesign Study was performed in three parts. Parts I and II are reported in this volume.

Part I of the study determined the overall feasibility of the aircraft technical requirements and concepts for a Rotor System Research Aircraft. Part I concluded that the concepts and requirements were feasible with minor modifications as recommended by Sikorsky. A brief synopsis of Part I is included as an Appendix to this report.

Part II of the Predesign Study compared two aircraft against the RSRA requirements. One of these is an all new aircraft specifically designed as an RSRA vehicle. A new main rotor, transmission, wings, and fuselage are included in this design. The second aircraft uses an existing Sikorsky S-61 main rotor, an S-61 roller gearbox which is currently under development in a U.S. Army supported program, and a highly modified Sikorsky S-67 airframe. The wing for this aircraft is a new design. Both aircraft employ a fan-in-fin anti-torque/yaw control system, T58-GE-16 engines for rotor power, and TF34-GE-2 turbofans for auxiliary thrust.

Each aircraft meets the basic requirements and goals of the program. The all new aircraft has inflight variable main rotor shaft tilt, a side-by-side cockpit seating arrangement, and is slightly faster in the compound mode. It is also somewhat lighter since it uses new dynamic components specifically designed for the RSRA. The existing component aircraft could be delivered earlier and at a substantial reduction in total program cost.

Preliminary development plans, including schedules and costs, have been prepared for both of these aircraft. It is projected that two copies of the existing component aircraft could be delivered to the government approximately three months ahead of the all new aircraft. There is no subsystem or component development, beyond that which is currently being funded, that is required before initiation of the aircraft development. This is due to the approach being used to provide the RSRA with certain of its unique capabilities. The main rotor force measuring system for the basic aircraft consists of a load cell mounting mechanism for the main gearbox. This system will handle many types of rotor systems for testing on the RSRA without the need for any type of active or passive vibration suppression system. However, a parallel program is recommended to develop the Sikorsky universal active vibration suppression device, which can also be used as a rotor balance system, so that the RSRA can handle rotors with large variations in blade passage frequency, such as slowed rotors.

Another area of concern in the development of the RSRA is in the aircraft flight control system. A fully redundant fly-by-wire system would be expensive and would extend the aircraft development schedule. Therefore, Sikorsky recommends a combination electrical/mechanical control system which will provide the required versatility at lower cost and within the aircraft development schedule.

Part II concluded with the recommendation that the existing S-61/S-67 dynamic components and an all new airframe offered the best design approach for the RSRA program. This design will best meet the technical requirements and goals while offering least risk and cost.

TABLE OF CONTENTS

ROTOR SYSTEMS RESEARCH AIRCRAFT

PREDESIGN STUDY *

INTRODUCTION

This Conceptual Study Report summarizes the results of Part II of the Rotor System Research Aircraft (RSRA) Predesign Study which has been performed by Sikorsky Aircraft under Contract No. NAS1-11228. An oral presentation of these results was given to NASA/Army personnel on May 22, 1972 at the Langley Research Center.

The objective of the RSRA Predesign Study is to define the most feasible research aircraft configuration for use by the government in performing research on a variety of helicopter and compound rotor systems at all speeds from hover to 300 knots. In addition, the Predesign Study must identify additional component research and technology developments that, if pursued in the scheduled development time, will improve the research capabilities of the RSRA.

Part I of the Study was concerned with determining the overall feasibility of the technical requirements and concepts envisioned by the government for the RSRA. Engineering trade-off studies were performed to determine the desirability of any changes or additions to minimize program time and cost. Two potential aircraft designs were developed to meet the requirements. One of these was an all new aircraft specifically designed as an RSRA vehicle. The second used existing aircraft components wherever feasible to reduce aircraft cost. Part I results are discussed in Appendix I of this report.

Part II of the Predesign Study was involved with further preliminary design of these two aircraft, including preliminary development plans and costs At the beginning of Part II, the government modified the aircraft technical requirements to reflect the results of the Part I study, and the designs were changed accordingly. With the conclusion of Part II, sufficient further analyses, design, and cost estimating was performed so that the government could select which features of the two aircraft designs should be included in a single RSRA configuration to be studied in Part III.

Part III of the Study was involved with the further analysis of this one aircraft configuration. This included further preliminary design and a more detailed analysis of development plans and costs. At the end of Part III, the government had a detailed definition of a Rotor Systems Research Aircraft, with a development plan and projected costs.

* The contract research effort which has led to the results in this report was financially supported by USSAMRDL (Langley Directorate).

THE ALL NEW AIRCRAFT DESIGN

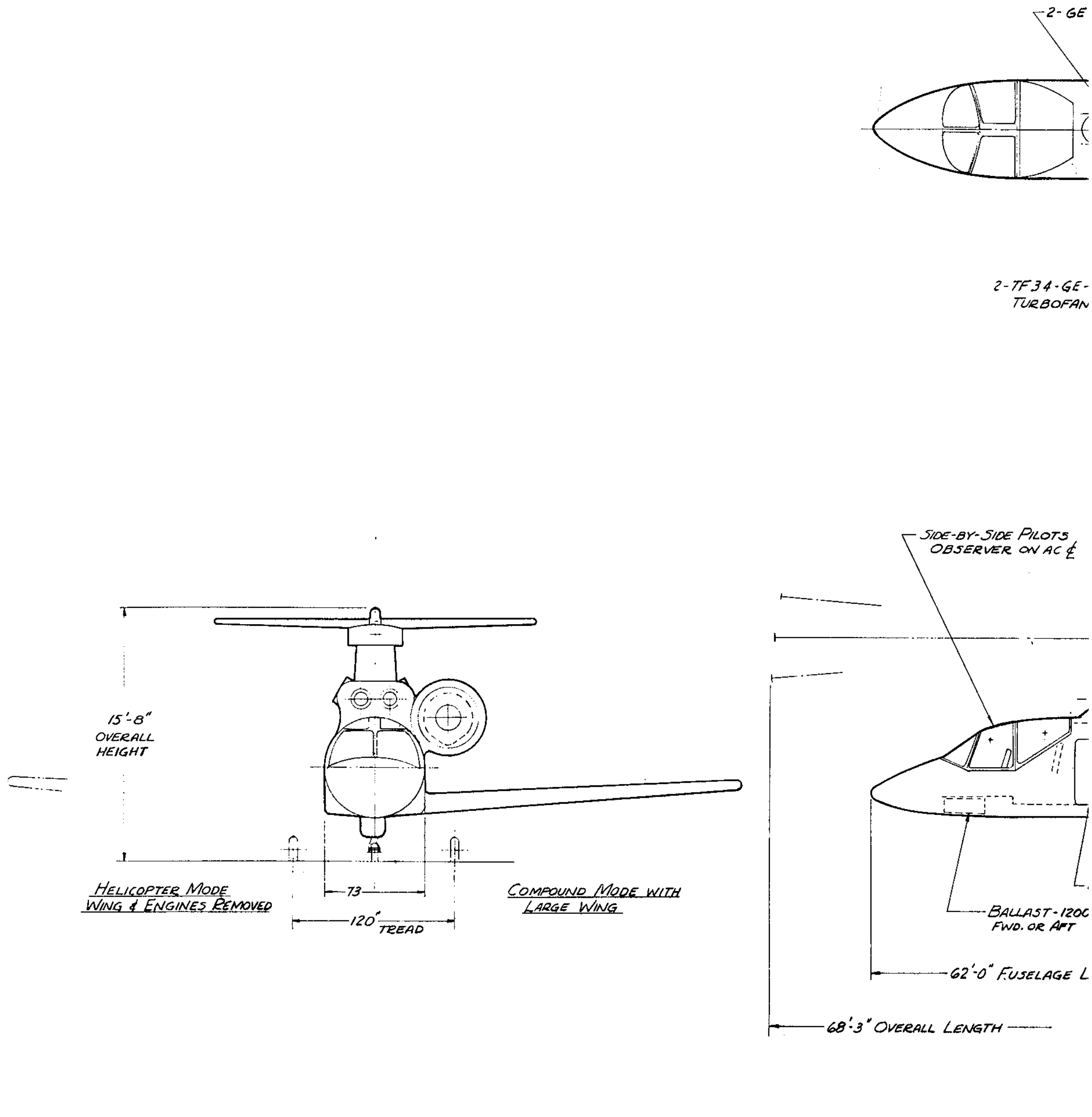
The all new aircraft design which evolved from the Part II study is illustrated on the opposite page. It is a 300 knot compound helicopter which has a five bladed 55.7 foot diameter rotor and a new main gearbox which is designed to provide inflight variable main rotor shaft tilt. An all new airframe is used which is specifically designed to provide the special features required for the RSRA. A side-by-side seating arrangement was chosen from human factors considerations.

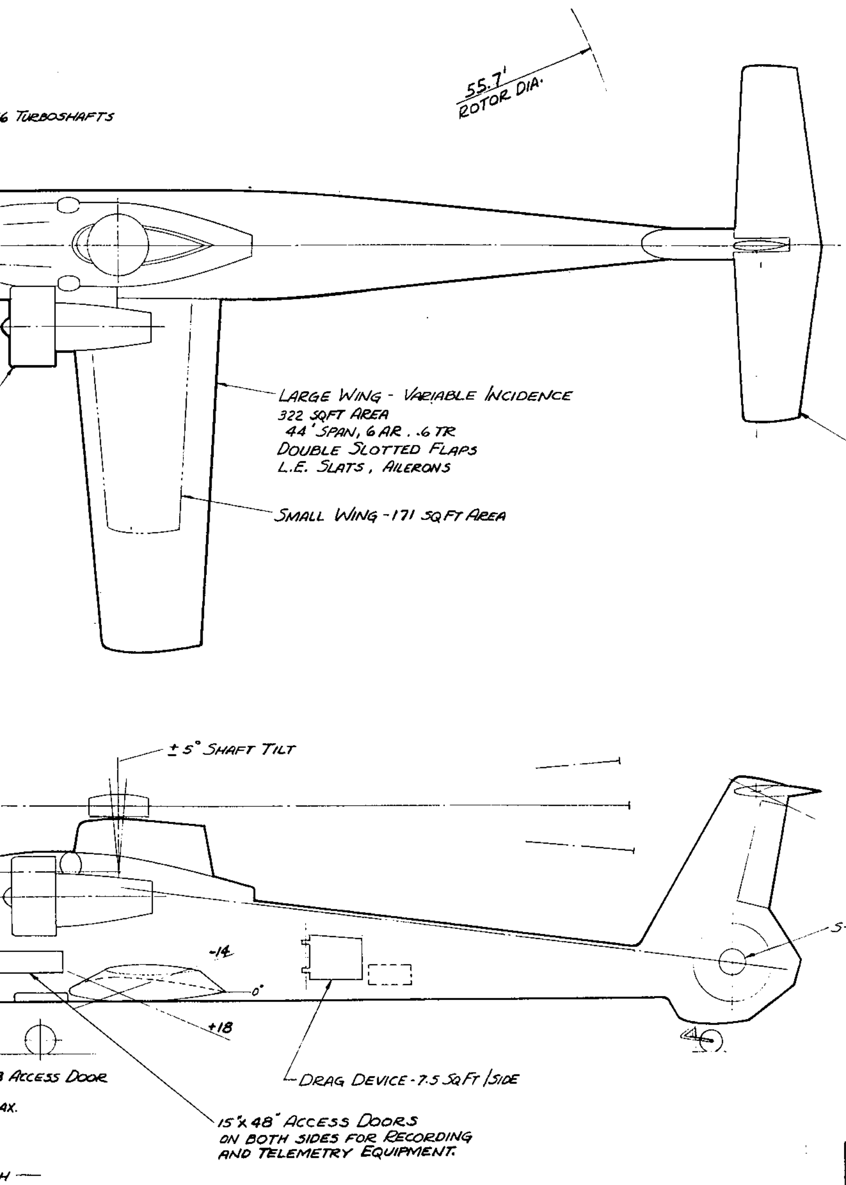
The unique features of this aircraft include:

- A wing capable of supporting full aircraft design gross weight at speeds as low as 120 knots.
- A variable wing incidence mechanism to vary wing angles of attack in flight.
- A variable rotor shaft tilt mechanism to vary rotor shaft angle in flight.
- Load cell instrumentation systems to measure all rotor and wing forces as well as auxiliary propulsion and tail rotor thrust.
- An electrical/mechanical control system to provide testing versatility with low cost and risk.
- A crew escape system, including a mechanism to severe the rotor blades before escape.
- A ballast system to vary aircraft center of gravity and inertia.
- Drag brakes to vary aircraft parasite drag.
- A fixed wing type landing gear and braking system to permit fixed wing landings at speeds up to 120 knots.
- A fan-in-fin anti-torque/yaw control fan.

The propulsion system on this aircraft uses two General Electric TF-34 fan engines for auxiliary propulsion. These are existing engines and are completely separate from the rotor propulsion engines. The rotor drive engines are two GE-T58-16's. These produce 1870 horsepower each. They were chosen as the most powerful derivative of the basic T-58 series. The next most powerful existing engines available are the small version of either the GE T-64 or the Lycoming T-55, both of which are in the 2800 horsepower category. Both of these are considerably heavier than the T-58, and would increase the weight of the aircraft. The T-58 also has the advantage of being a rear drive engine, which helps to alleviate any tail heavy balance problems with the aircraft.

The landing gear configuration selected uses a tail wheel, with the main gear forward. The logic for this decision stems from the fact that the primary mission of the RSRA is rotor system testing, and a tail wheel is definitely preferred during nose high helicopter type landings. Although a nose wheel might be preferable for the high speed fixed wing type of landing, these would, in fact, rarely be performed and are strictly secondary to the more conventional helicopter type of landings. The tail wheel configuration is also somewhat lighter since the nosewheel arrangement requires a tail skid or bumper in addition to the basic three point gear.


2-GE
2-TF34-GE-
TURBOFAN
SIDE-BY-SIDE PILOTS
OBSERVER ON AC ℄
15'-8"
OVERALL
HEIGHT
HELICOPTER MODE
WING & ENGINES REMOVED
73
120" TREAD
COMPOUND MODE WITH
LARGE WING
BALLAST-120C
FWD. OR AFT
62'-0" FUSELAGE L
68'-3" OVERALL LENGTH



DESIGN GW - 24,392 LB.

PROGRAM		NAME	DATE	FUNCTIONAL	NAME	DATE
	DRAWN BY	G HOWARD	5-2-72	AC & SS		
	DESIGNER			AD & D		
	SYS DESIGN			STRUCT & MATL		
	SYS ENGINEER			AERO MECHANICS		
	ACFT DESIGN			SD & E		
	CHIEF DESIGN			CHIEF TEST ENGR		
	TASK MANAGER			CHIEF ENGINEER		
				ENGRG MANAGER		
				MFG/MFG ENGRG		

REPORT NO. | PAGE NO. | FIGURE NO.

ROTOR SYSTEM RESEARCH AIRCRAFT - RSRA
GENERAL ARRANGEMENT - ALL NEW DESIGN
NASA / ARMY

Sikorsky Aircraft | CODE IDENT. NUMBER 78286 | DS - 509 - 3 - 4
SCALE 1/40 | WBS No. | REV.
SHEET OF

The main changes that were made from the part one design (see Appendix I) are in changing the cockpit from a tandem to a side by side seating arrangement, replacing the convertible propulsion system with separate systems for rotor and cruise propulsion, replacing the tail rotor with an anti-torque fan, and using two different wing designs.

Basic characteristics of this aircraft are listed in Table 1 on page 6.

THE EXISTING COMPONENT AIRCRAFT

The general arrangement of the existing component aircraft is shown on the opposite page. It uses a Sikorsky S-61 main rotor system, and a 3700 horsepower roller main gearbox which is now under a U.S. Army Development Program. Its airframe is derived from the Sikorsky S-67 Blackhawk, although extensive modifications are required to meet the RSRA technical requirements. This aircraft has all of the features and capabilities of the all new design except that it does not have inflight variable main rotor shaft tilt and it uses a tandem cockpit arrangement.

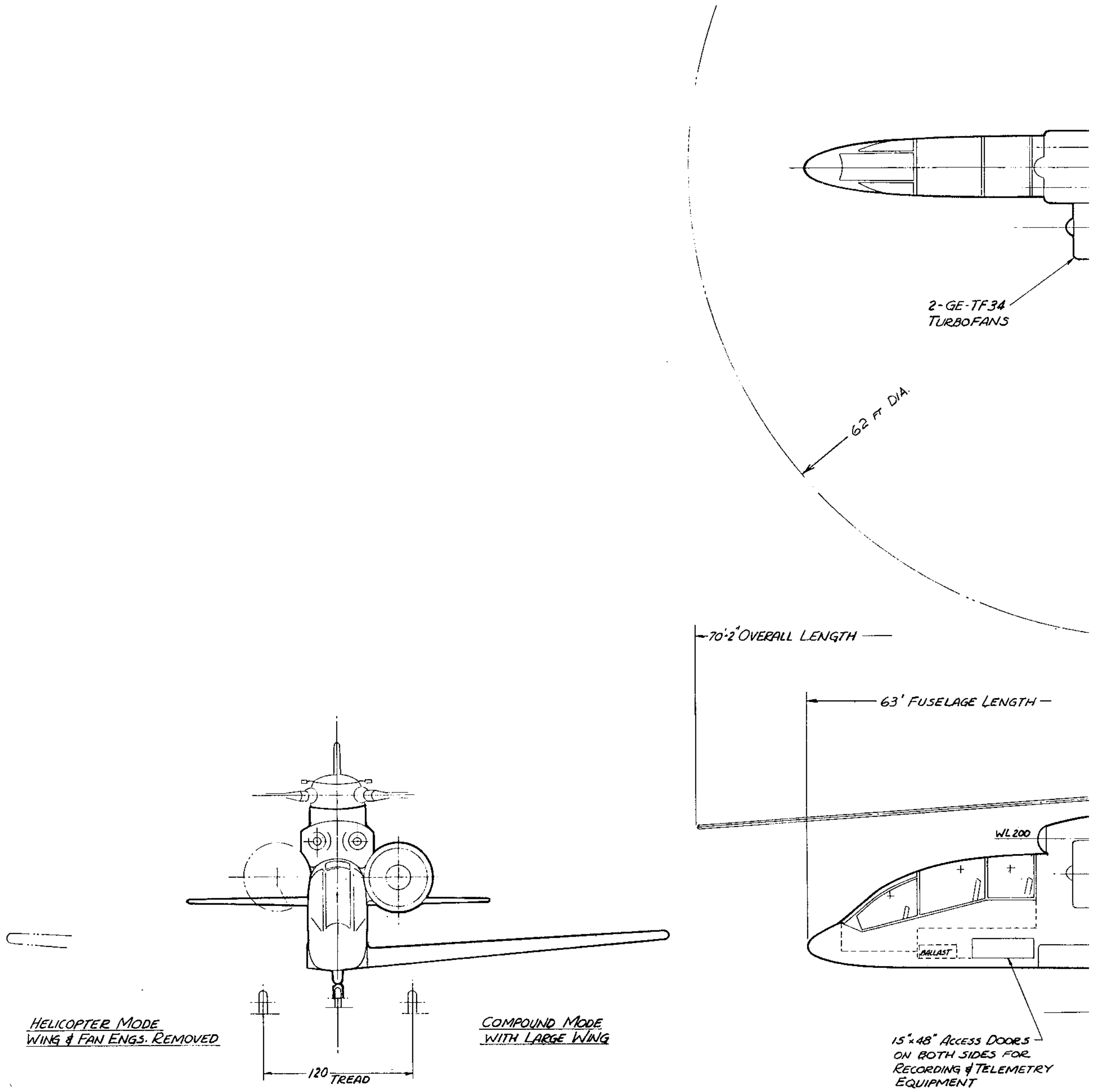
The gearbox is mounted such that it can be shimmed to allow ground adjustable shaft tilt of ±2 degrees. Thomas couplings in the tail drive shaft allow this deflection. The power required of the fan-in-fin can be accommodated by the roller gearbox and beefed up components in the tail shaft drive. The S-61 rotor system which is the basic rotor for this aircraft, is a five-bladed, 62 foot diameter rotor, with a 1.52 ft chord. The twist of the blades is -3°.

The design load factor for both of these aircraft is 4.0, ultimate load factor is 6.0. This change requires extensive airframe modifications over the S-67 fuselage, and very little of the existing fuselage is being retained.

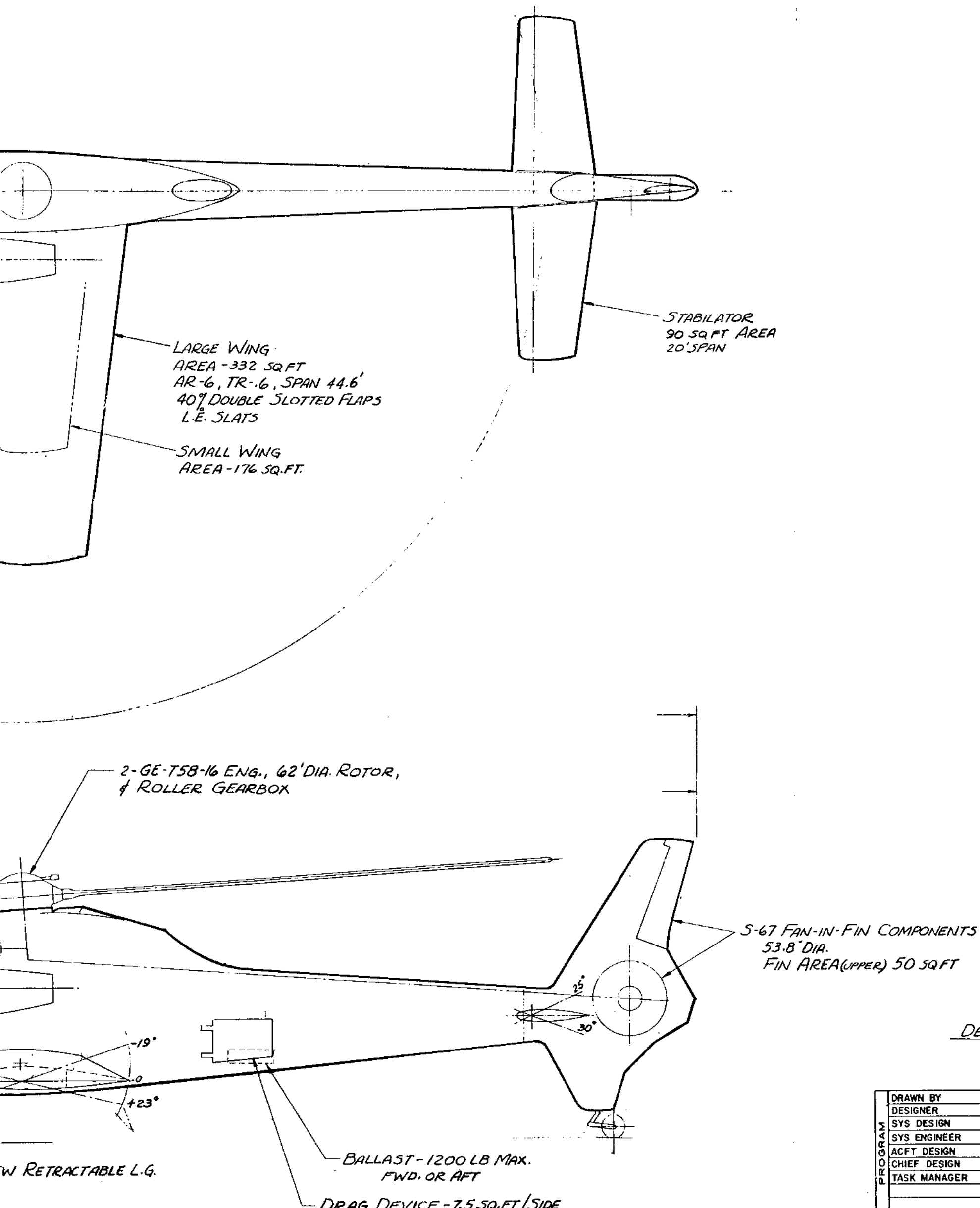
Sikorsky has recently been awarded a contract from the U.S. Army to design, construct and flight test an anti-torque fan on the S-67 Blackhawk helicopter. The existing component aircraft has been configured with that fan arrangement.

The basic changes from the design showed at the end of Part I include the replacement of the tail rotor with an anti-torque fan, the inclusion of two separate wings rather than one, and an increase in the design structural load factor. In addition, other changes have been made to meet the technical requirements as modified at the end of Part I, and the nose has been extended to alleviate a tail heavy balance situation.

Basic characteristics for this aircraft are also listed in Table 1 on page 6.



2-GE-TF34
TURBOFANS
62 FT DIA.
70'-2" OVERALL LENGTH
63' FUSELAGE LENGTH
WL 200
BALLAST
15" x 48" ACCESS DOORS
ON BOTH SIDES FOR
RECORDING & TELEMETRY
EQUIPMENT
HELICOPTER MODE
WING & FAN ENGS. REMOVED
COMPOUND MODE
WITH LARGE WING
120 TREAD



DESIGN GW - 25,150 LB

PROGRAM		NAME	DATE	FUNCTIONAL	NAME	DATE
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	DESIGNER			AD & D		
	SYS DESIGN			STRUCT & MATL		
	SYS ENGINEER			AERO MECHANICS		
	ACFT DESIGN			SD & E		
	CHIEF DESIGN			CHIEF TEST ENGR		
	TASK MANAGER			CHIEF ENGINEER		
				ENGRG MANAGER		
				MFG/MFG ENGRG		

REPORT NO. | PAGE NO. | FIGURE NO.

ROTOR SYSTEM RESEARCH AIRCRAFT (RSRA)
GENERAL ARR. - EXISTING COMPONENT DESIGN
NASA

Sikorsky Aircraft | CODE IDENT. NUMBER 78286 | DS-509-3-3

SCALE 1/40 | WBS No. | REV.

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TABLE I

AIRCRAFT DESIGN PARAMETERS

	ALL NEW AIRCRAFT	EXISTING COMPONENT AIRCRAFT
Gross Weight	24392 lbs	25150 lbs
Weight Empty	18753 lbs	19365 lbs
Fuel Weight	3119 lbs	3174 lbs
Vertical Drag, Large Wing Installed	8.5 %	7.5 %
Disc Loading	10 psf	8.33 psf
f, Small Wing Installed	23.4 ft^2	23.6 ft^2
Ultimate Vertical Load Factor	6.0 g	6.0 g
Main Rotor		
Radius	27.86 ft	31 ft
Chord	1.416 ft	1.52 ft
Solidity	.081	.0782
Tip Speed (Hover)	700	686 fps
C_T/σ (Hover @ SLS)	0.115	.091
Twist	-3.0 deg	-3°
Number of Blades	5	5
Aspect Ratio	19.7	20.4
Tail Fan		
Radius	2.19 ft	2.19 ft
Number of Blades	12	12
Tip Speed	850 fps	850 fps
Power Engines		
Number	2	2
Type	GE T-58-16	GE-T58-16
Military Power	1870 HP	1870 HP
Auxiliary Propulsion Engines		
Number	2	2
Type	GE-TF34-2	GE-TF-34-2
Intermediate Installed Static Thrust	7770 lbs	7770 lbs
Intermediate Installed Thrust at Sea Level Standard, 300 knots	5080 lbs	5080 lbs
Drive System Design Power	3700 HP	3700 HP
Performance		
Design Hover	Meets Requirements	Exceeds Requirements
Dash	SLS	SLS
Dash Speed	321 kts	316 kts
One Engine Out Capability	Meets Requirements	Meets Requirements
Horizontal Tail Area	90 ft^2	90 ft^2
Vertical Tail Area	50 ft^2	50 ft^2
Wing Area, Large Wing	322 ft^2	332 ft^2
Wing Area, Small Wing	171 ft^2	176 ft^2
Body Wetted Area	889 ft^2	902 ft^2

AIRCRAFT WEIGHT STATEMENTS

Weight statements for the two RSRA aircraft are tabulated in Table 2 on page 8. Because of the basic similarities in the design, there is not a large difference in their overall weights. The new aircraft has a lighter rotor and drive system since these have been designed strictly for the RSRA and are not existing components from other aircraft. The existing component aircraft has a minor advantage in the body weight because it uses the tandem cockpit. All other subsystem weights are identical, except where they have minor differences to reflect the different design gross weights of the two vehicles.

The all new aircraft also has a higher contingency due to the fact that more of its weight is estimated, whereas many of the weights for the S-67 derivative are actual weights of real hardware.

AIRCRAFT PERFORMANCE

The performance of the RSRA aircraft was calculated for compliance with the Statement of Work requirements. Vertical drags were calculated using the NASA/Army method of the Statement of Work. The equivalent parasite area for the Part II aircraft was estimated primarily using the NASA/Army method as Sikorsky estimates indicate possible lower areas. Engine performances are from manufacturers specifications with SFC's increased by five percent, and forward flight performance was executed using Sikorsky techniques which were shown in Part I to be a more conservative approach than that originally requested in the Statement of Work.

Vertical Drag

Vertical drag was calculated on the basis of the dynamic pressure distribution in the rotor downwash and the vertical position of the centroid of the airframe segment as outlined in Section 6.2.4.1 (g) of the Statement of Work. The vertical drag by the method is 8.5 percent of gross weight for the all new aircraft and 7.5 percent for the existing component aircraft. This difference is a result of the different main rotor diameters being used.

Equivalent Parasite Area

The equivalent parasite area was estimated with the equations supplied and modified by NASA/Army at the end of Part I. Aircraft wetted areas and powers were calculated and the equivalent parasite areas for the rotor hub and mast, plus the wing, fuselage, and empennage were found by the formulas. On top of these values, parasite areas were estimated for the T58-16 and the TF 34 fan installations and an additional five percent was added to account for protuberances and leakage. The equivalent parasite areas for the aircraft (which included zero lift wing drag for the small wing) are 23.4 for the all new design, and 23.6 for the existing component design.

TABLE 2
RSRA AIRCRAFT WEIGHT STATEMENTS

	NEW AIRCRAFT	EXISTING COMPONENT AIRCRAFT
Rotor Group	1576 lbs	2104 lbs
Wing Group, Small Wing	1022	1061
(Large Wing)	(2300)	(2388)
Tail Fan	350	350
Tail Surfaces	514	514
Body Group	3009	2921
Alighting Gear	1022	1050
Flight Controls	1335	1362
Engine Sections	939	939
Engines	3709	3709
Engine Related Items	424	424
Fuel System	276	281
Drive System	1814	2128
Instruments	326	328
Hydraulics	40	40
Electrical	398	398
Avionics	248	248
Furnishings	353	353
Air Conditioning	136	136
Auxiliary Gear	40	40
Vibration Suppression	500	503
Contingency	721	567
Weight Empty	18753	19456
Crew	400	400
Fluids	120	120
Mission Fuel (Including 15 minutes endurance at 300 knots)	3119	3174
Mission Payload	2000	2000
Gross Weight	24392	25150

Hovering Capability

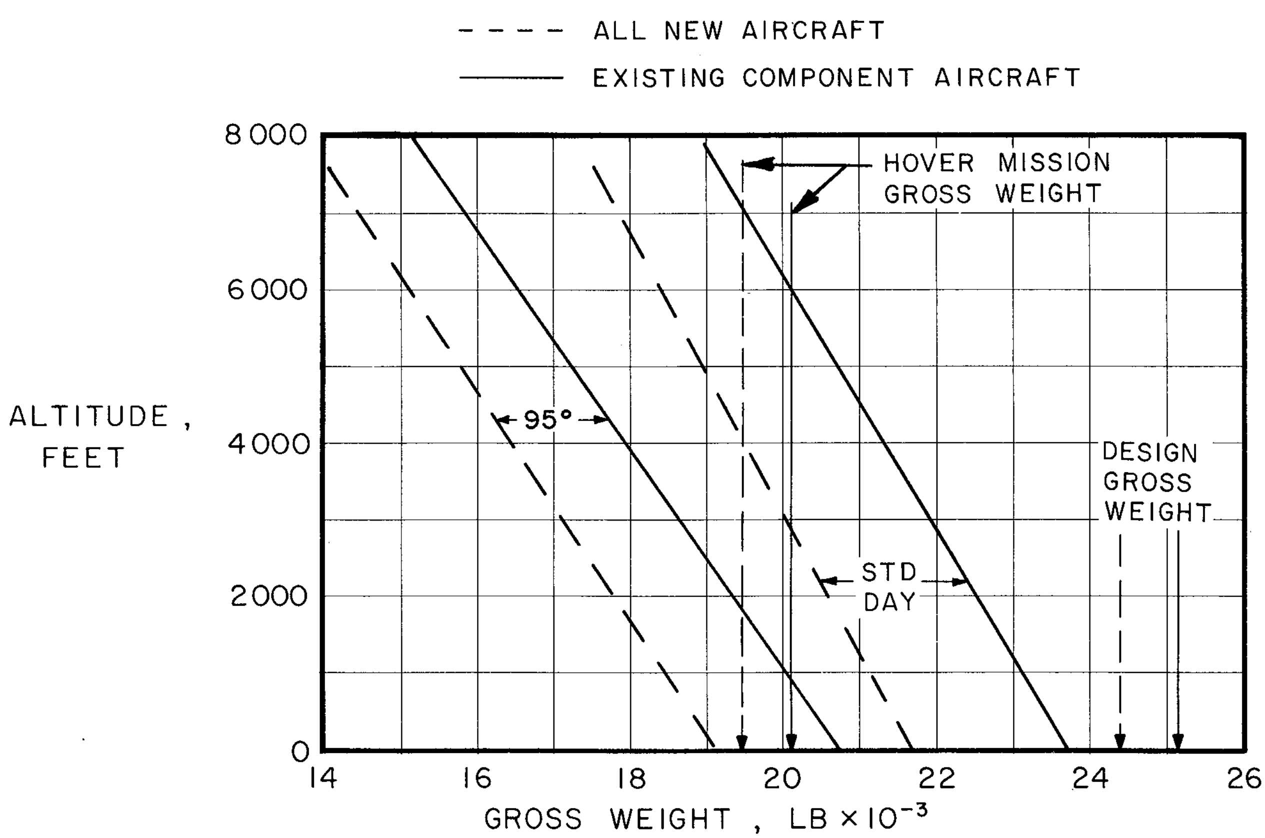
The hovering capability of the RSRA aircraft without the wing was calculated to show compliance with the RSRA hovering missions. The weight-altitude-temperature curve which follows shows the capability of both aircraft with the wings on.

With the fuel for the 32 minutes of hover, 10 nautical miles of cruising and the required fuel reserves, the take-off weights for the hovering missions are 19446 lbs for the all new aircraft and 20107 lbs for the existing component aircraft. The existing component aircraft meets the RSRA design goal. The all new aircraft would be required to remove the wing in addition to the auxiliary propulsion engines to meet the goal, under the sea level, 95°F conditions.

RSRA HOVER PERFORMANCE

FAN-IN-FIN INSTALLED

LARGE WINGS INSTALLED



Speed Capability

High speed thrust requirements were calculated for both aircraft. The thrust required included basic aircraft fuselage drag, wing induced and parasite drag and rotor induced drag and H forces. Rotor forces were calculated using Sikorsky's general rotor performance computer deck with skewed flow effects taken into account as approved by NASA/Army after Part I completion. The following figure shows the high speed end of the thrust required curve with TF34-GE-2 installed available thrust. At the design gross weight, both aircraft are capable of exceeding the 300 knot RSRA speed requirement, at sea level standard and 9500' standard atmospheric conditions.

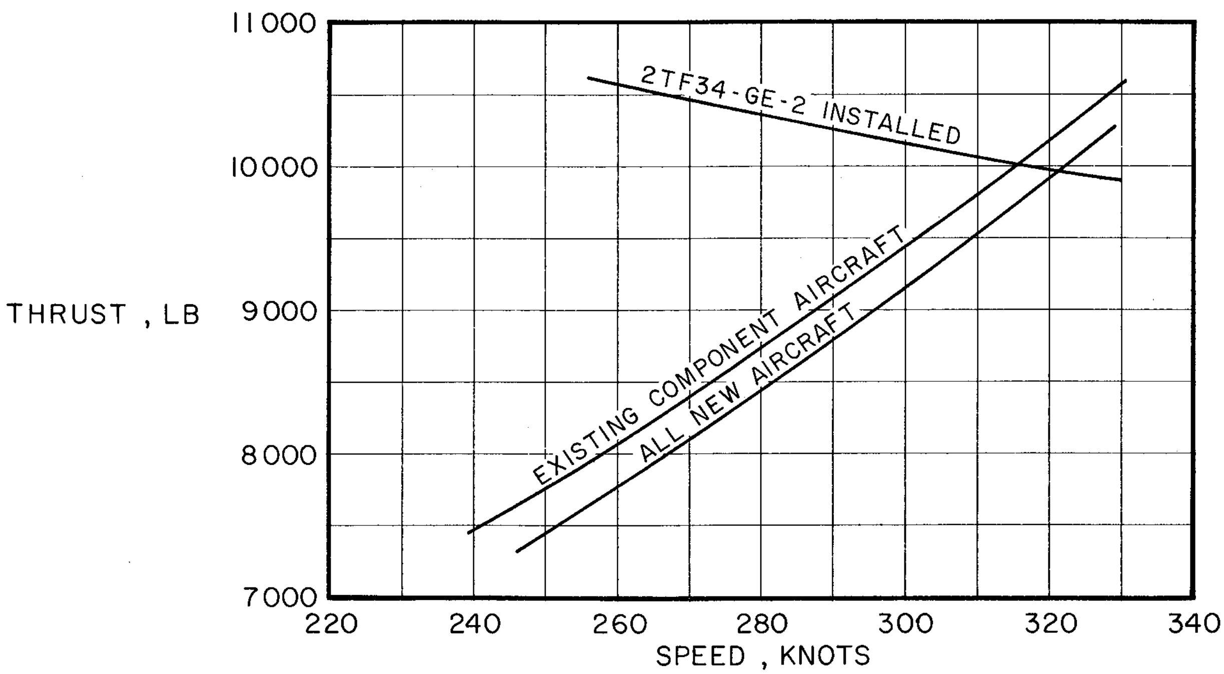
RSRA HIGH SPEED THRUST

S.L. STD.

SMALL WINGS INSTALLED

THRUST AVAILABLE & THRUST REQUIRED VS. SPEED

ALL NEW G.W. = 24 392 LB EXISTING COMPONENT G.W. = 25 150 LB



Mission Analysis

The RSRA mission analysis set up in the compound design model computer program includes fuel flows increased by five percent above the manufacturers engine performance data and all elements required in the Statement of Work. The mission breakdown for the all new aircraft is shown below and has all the elements of the Statement of Work included in the fuel calculation. Twenty minutes of fuel at the airspeed for maximum range is the most critical reserve requirement.

TOGW= 24392.0 LBS., ROTOR RADIUS= 27.86 FT., PARASITE DRAG= 23.4 SQ.FT.

TYPE OF ENGINES- NUMBER 3 (2.)

MODE	GR.WT (LBS)	TEMP (DEG.F)	ALT (FT)	OPTN (ZR/FPM)	SPEED (KTS)	VSTALL (KTS)	DIST (N.MI)	TIME (MIN)	FL.AR. (SQ.FT)	SHP	FUEL (LBS)
WU/TO	24392.	59.	0.	--	--	--	--	2.0	--	10895.0	203.8
HOVER	24188.	59.	0.	1000.000	--	--	--	2.0	--	4563.9	109.2
ACCEL	24079.	59.	0.	.00	150.0	*****	5.0	2.0	23.44	2402.7	81.7
DASH	23997.	59.	0.	.00	300.0	*****	75.0	15.0	23.44	11857.5	1744.5
DECEL	22253.	59.	0.	.00	150.0	*****	5.0	2.0	23.44	2361.3	81.2
HOVER	22172.	59.	0.	1000.000	--	--	--	2.0	--	3888.4	100.1
RESERVE-CRUISE	22072.	59.	0.	.00	146.0	*****	48.7	20.0	23.44	2238.4	798.1

TOTAL MISSION FUEL IS 3118.5 LBS

TOTAL MISSION TIME IS 25.0 MINS

Helicopter Simulation

The negative wing incidence range for both RSRA aircraft was established by the design goal of full gross weight autorotation at 100 knots. With the negative wing angles required by autorotation, the wing has the capability to produce sufficient negative lift to load up conventional rotors to their upper stall limits.

The only restriction on helicopter simulation is on the RSRA design goal of complete rotor unloading down to 100 knots. With the high lift configuration selected for the RSRA aircraft, 100 percent unloading below 120 knots is not possible with a 20 percent stall margin on airspeed. This restriction was imposed with government concurrance when it was found that to get complete unloading, either an even larger wing would be required or the $C_{l\,max}$ would exceed the "state-of-the-art" in high lift design.

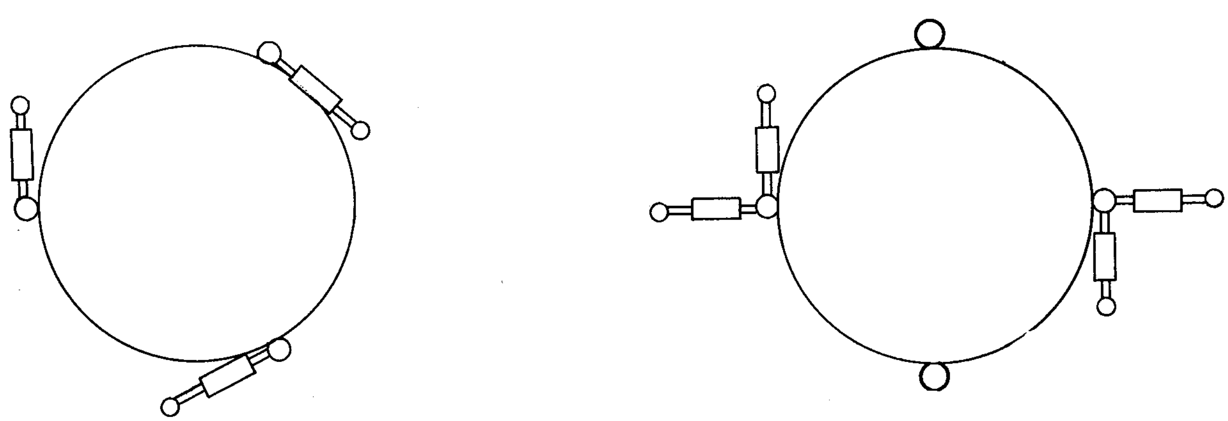
DATA SYSTEMS, INCLUDING INSTRUMENTATION
ACCURACY STUDY

Both RSRA aircraft are configured with onboard load cell instrumentation systems to measure rotor forces and moments, wing forces and moments, auxiliary propulsion thrust and anti-torque system thrust. During Parts I and II, instrument accuracy studies were conducted to show the expected accuracy that could be obtained with these systems in order to establish a level flight test point simulating a pure helicopter.

Rotor Force Measuring System Accuracy

A rotor force accuracy study was conducted with two configurations. Configuration A assumed three horizontal transducers and three vertical transducers at a radial distance of 20 inches from the main rotor shaft. Configuration B consists of four horizontal and three vertical transducers. The two configurations are shown below.

CONFIGURATION A CONFIGURATION B



The effect of configuration by transducer placement is demonstrated in the Table 3. Considerable improvement in longitudinal force and pitching moment accuracy is shown with configuration B. This improvement is due to the fact that no torque loads are felt in the transducers which measure longitudinal force and pitching moment. The torque is now entirely felt in the lateral mode and thus the lateral force and rolling moment accuracy are impaired. This arrangement sacrifices lateral accuracy for an improvement in longitudinal accuracy.

TABLE 3.
EFFECT OF CONFIGURATION
ON ACCURACY
(ROTOR MEASUREMENT SYSTEM)

MAIN ROTOR HUB FORCES	TEST CONDITION	CONFIGURATION A (1σ ACCURACY)	CONFIGURATION B (1σ ACCURACY)
Long.	1380 lbs	± 185 lbs	± 33 lbs
Lat.	0 lbs	± 171 lbs	± 216 lbs
Thrust	18000 lbs	± 105 lbs	± 115 lbs
Roll M.	0 ft-lbs	±1000 ft-lbs	±1296 lbs
Pitch M.	6750 ft-lbs	±1080 ft-lbs	± 300 lbs
Torque	60000 ft-lbs	± 416 ft-lbs	± 432 lbs

Canting of the Vertical Transducers

Canting of the vertical transducers, as is proposed with the optional Active Rotor Balance Vibration Suppression System, will cause a redistribution of load paths particularly in the horizontal direction. Thus, the resulting accuracies would be expected to fall somewhere inbetween that of Configuration A and Configuration B.

Wing Force Measuring Systems Accuracy Study

The wing accuracy study was performed including all force and moment equations of the wing. The transducers were estimated to be accurate to 1% of applied load. The resulting accuracy equations showed that worst accuracy is obtained when wing lift and pitching moment are greatest and the wing angle of attack is large. The resulting accuracy for worst case is seen to be better than 2%. These results indicate that this wing measurement system concept can provide good accuracies. Good alignment and calibration must be made in order to achieve these accuracies. Wing accuracy results are shown on page 14.

Alignment and Calibrations

The instrumentation accuracies are based on the assumed configurations for each concept and the range of anticipated forces acting on the transducers. Further degradation in accuracy will result from misalignment and calibration considerations. This is mainly due to friction and dead band connections of the transducer tie points. Proper alignment of connecting points must be made in order to achieve the accuracies presented here. In addition, physical calibrations of each concept must be made in order to average out small misalignments and to maintain the individual transducer accuracies when connected to form a multiple transducer configuration.

WING ACCURACY RESULTS

CASE 1, V = 300 Knots, Full Wing Loading, $\alpha = 3^{\circ}$

WING FORCE	TEST CONDITION	ACCURACY 1σ
Lift	25000 lbs	± 235 lbs
Drag	2000 lbs	± 17 lbs
Pitch M.	15000 ft-lbs	± 150 ft-lbs
Roll M.	0 ft-lbs	± 448 ft-lbs
Yaw M.	0 ft-lbs	± 34 ft-lbs

CASE 2 (Worst Case) V = 120 Knots, Full Wing Loading, $\alpha = 13^{\circ}$, Flaps Down

WING FORCE	TEST CONDITION	ACCURACY 1σ
Lift	25000 lbs	± 405 lbs
Drag	7600 lbs	± 143 lbs
Pitch M.	75000 ft-lbs	± 750 ft-lbs
Roll M.	0 ft-lbs	± 870 ft-lbs
Yaw M.	0 ft-lbs	± 230 ft-lbs

In addition to the above measurements, auxiliary propulsion system thrust and fan-in-fin thrust and power are measured at or below ± 2% accuracy.

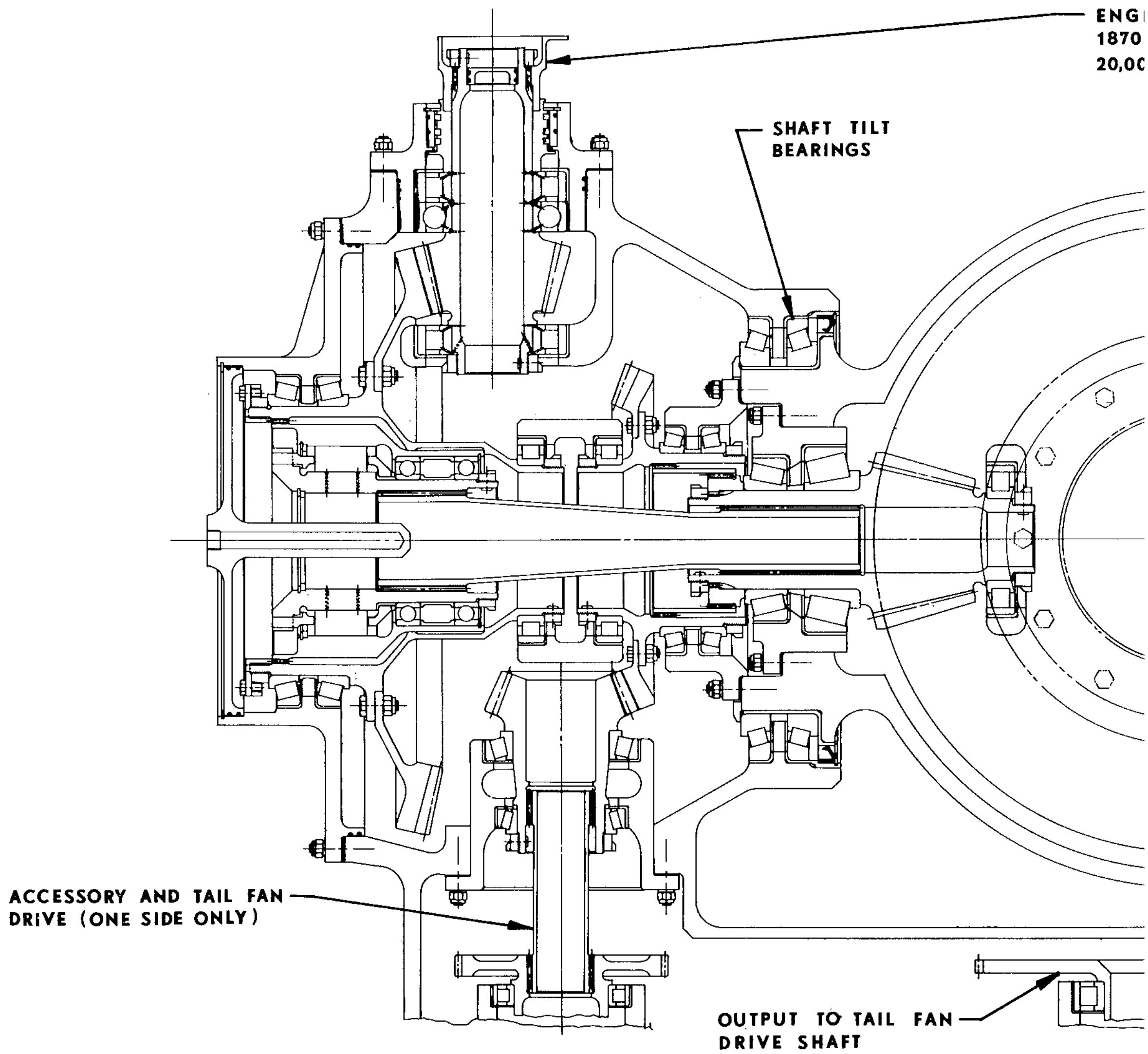
AUXILIARY PROPULSION

The auxiliary propulsion engines for the RSRA aircraft were selected from the range of engines with enough thrust to allow the aircraft to accelerate to and maintain a cruise speed of 300 knots at both sea level and 9500' altitude, standard conditions in the compound mode. The selected engines are two TF34-GE-2 turbofans.

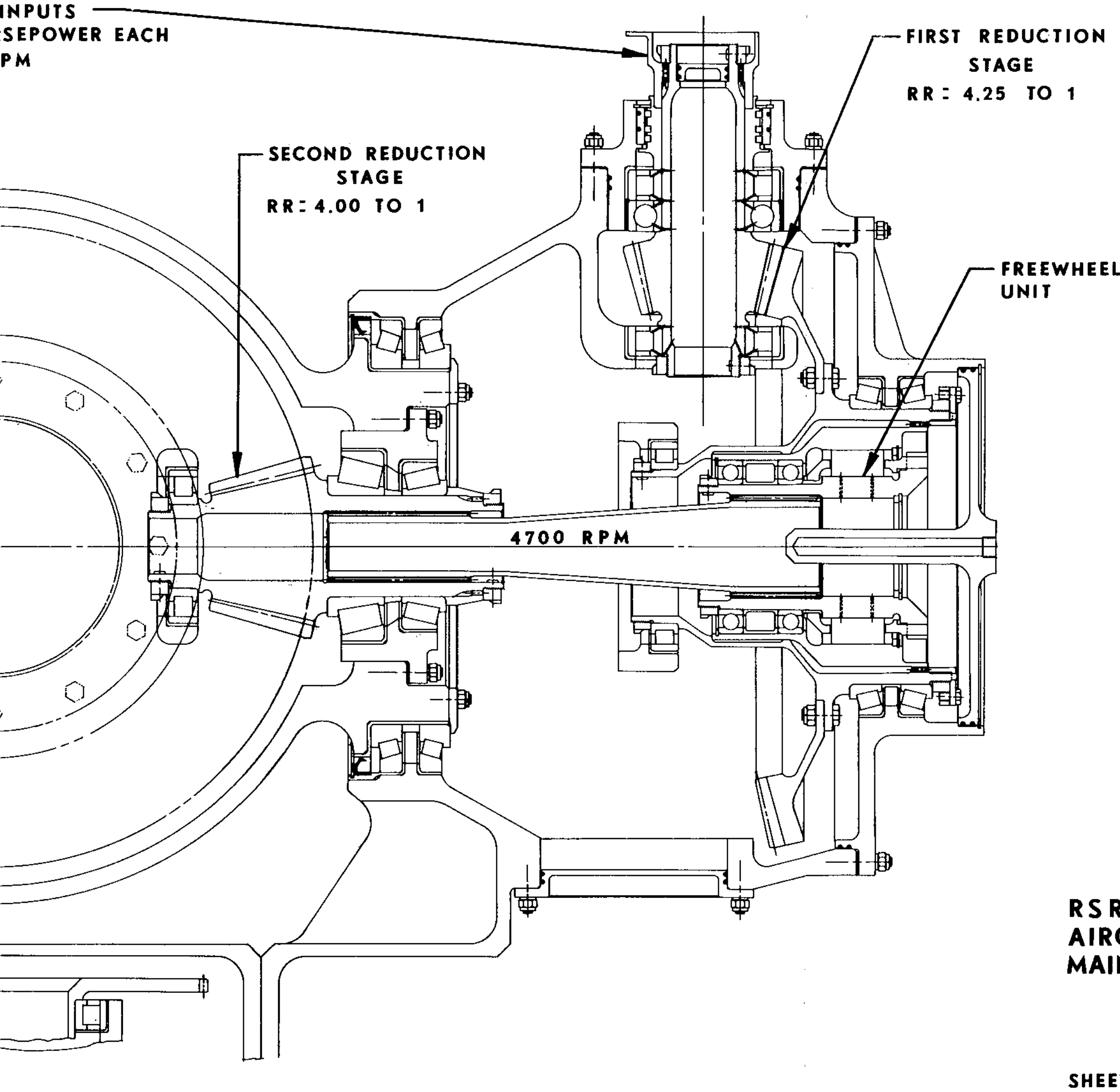
The TF34-GE-2 is a dual-rotor front-fan configuration with a bypass ratio of 6.23. It has a single-stage fan with a pressure ratio of 1.51 to 1, and a 14-stage axial flow compressor with variable stators and nominal pressure ratio of 14.5 to 1. The gas generator turbine has 2 axial stages, both air-cooled. Air is introduced directly to the fan rotor with no fan inlet guide vanes. Performance estimates are based on General Electric model specification E1130, Revision C. The sea level static thrust and sfc values for the uninstalled engines are:

RATING	THRUST (LBS)	(LB/HR/LB)
Maximum	9275	.363
Intermediate	8159	.349
Max. Continuous	7513	.344

Installed losses for the effects of inlet and exhaust pressure losses have been estimated. The inlet and exhaust losses are estimated at 4% on thrust.



ENG
1870
20,0C
SHAFT TILT
BEARINGS
ACCESSORY AND TAIL FAN
DRIVE (ONE SIDE ONLY)
OUTPUT TO TAIL FAN
DRIVE SHAFT



RSRA
AIRCRAFT I
MAIN GEARBOX

SHEET 1

In the compound mode at 300 knots, both aircraft have their maximum thrust required with the rotor in autorotation and the auxiliary thrust engines overcoming the rotor drag in this condition as well as the usual fuselage and wing drags. At the 300 knot condition, the available installed thrust for two TF34-GE-2 engines is 10100 lbs, for the intermediate rating. The required thrust in the compound mode in the worst case is 9440 lbs. Available thrust exceeds required thrust by 7 percent. Both aircraft can achieve 300 knots with the small wings.

The auxiliary thrust engines are mounted on the sides of the fuselage and are easily removed for flight tests in the pure helicopter mode. The TF-34 is in production for the Lockheed S-3A anti-submarine warfare aircraft for the U.S. Navy and is proposed for various other commercial and military aircraft. It is available to the government for the RSRA program.

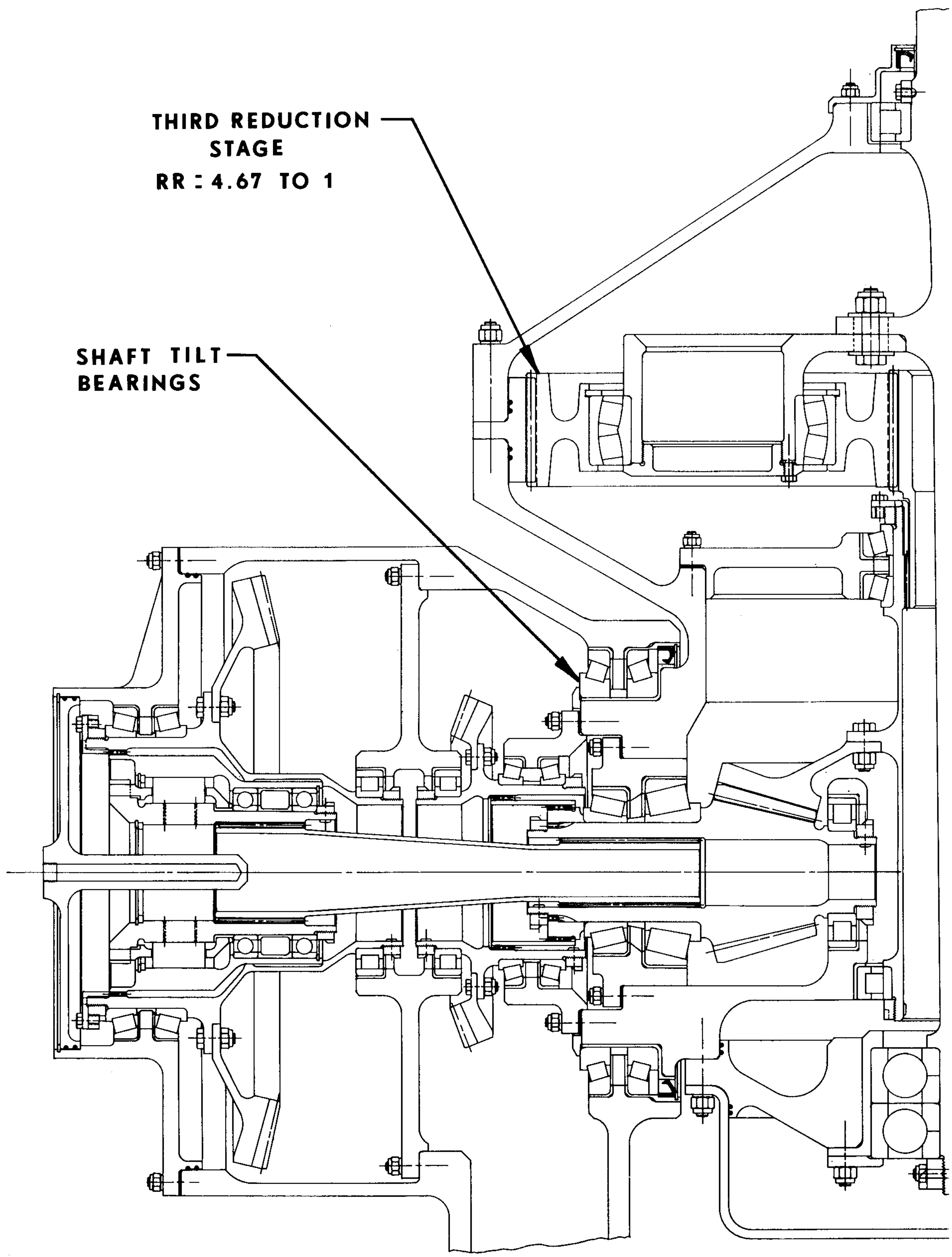
DESCRIPTION OF THE MAIN ROTORS, DRIVE SYSTEM AND SHAFT TILT

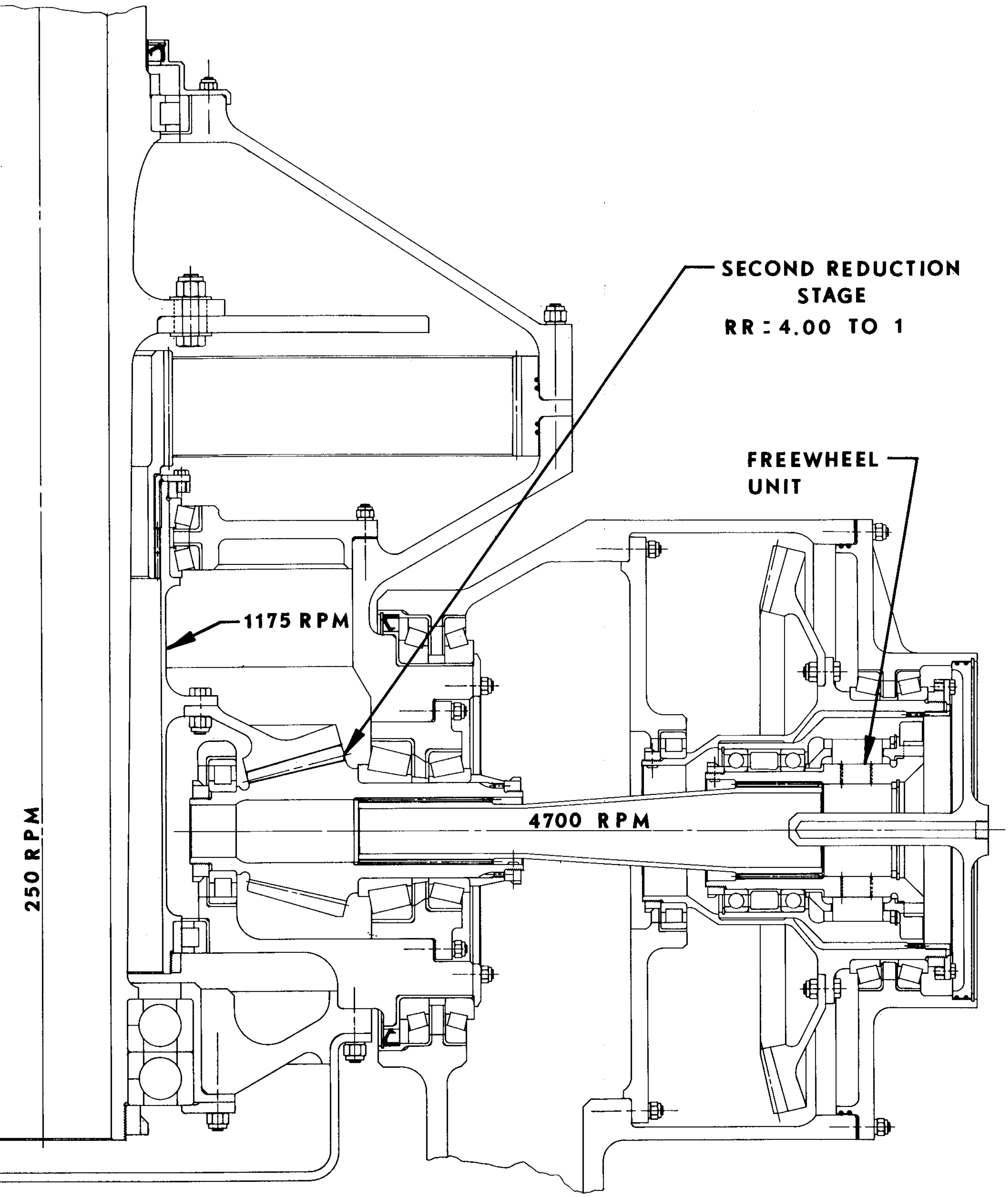
The main rotor, drive system and the shaft tilt arrangements are different for the two aircraft. The all new aircraft has an all new rotor and drive system and incorporates inflight variable shaft tilt. The existing component aircraft has existing components and has only ground adjustable shaft tilt, over a minimum range.

All New Aircraft Main Rotor and Drive System

The all new aircraft has a five-bladed 55.7 ft. diameter rotor with a chord of 1.42 ft. The twist of the blade is -3° and the last seven percent of the blade is swept aft 30 degrees relative to the span axis. The blade section used for this study is 0012. Aft tip sweep is selected to obtain low vibratory control loads, low blade stresses at the high RSRA forward flight speeds, and improved hovering efficiency through aerodynamic compressibility relief.

The drive system consists of a new gearbox capable of transmitting 3700 horsepower. A three stage reduction is used with two bevel stages and one planetary stage. Power is transmitted between the first and second reduction stage by horizontal transverse shafts which provide the axis for the inflight adjustable main rotor shaft tilt. The center section of the gearbox is mounted on bearings to the two input sections, so that it can be rotated through 10 degrees (±5) by hydraulic actuators to provide the inflight adjustable shaft tilt feature. The power take-off to the tail and accessories is from one of the input sections so that it is not affected by shaft tilt. It is sized to accept the powers required by the fan-in-fin anti-torque device. Drawings of this gearbox are shown on pages 15 and 17.


THIRD REDUCTION STAGE
RR = 4.67 TO 1
SHAFT TILT BEARINGS



RSRA
AIRCRAFT I
MAIN GEARBOX

SHEET 2

Existing Component Aircraft Main Rotor and Drive System

The S61 rotor system is the basic rotor for the existing component aircraft. The rotor is a five-bladed, 62 ft. diameter with a 1.52 ft. chord. The twist of the blades is -3^{o} and the section and tip sweep are the same as the all new aircraft.

The drive system uses the roller gearbox now under a U.S. Army development program. It is rated at 3700 horsepower. The gearbox is mounted such that it can be shimmed to allow ground adjustable shaft tilt of ±2 degrees. Thomas couplings in the tail drive shaft allow this deflection. The power required of the fan-in-fin can be accommodated by the roller gearbox and beefed up components in the tail shaft drive. A drawing of this system is shown on page 19. Also illustrated on this drawing is the load cell rotor force measuring system.

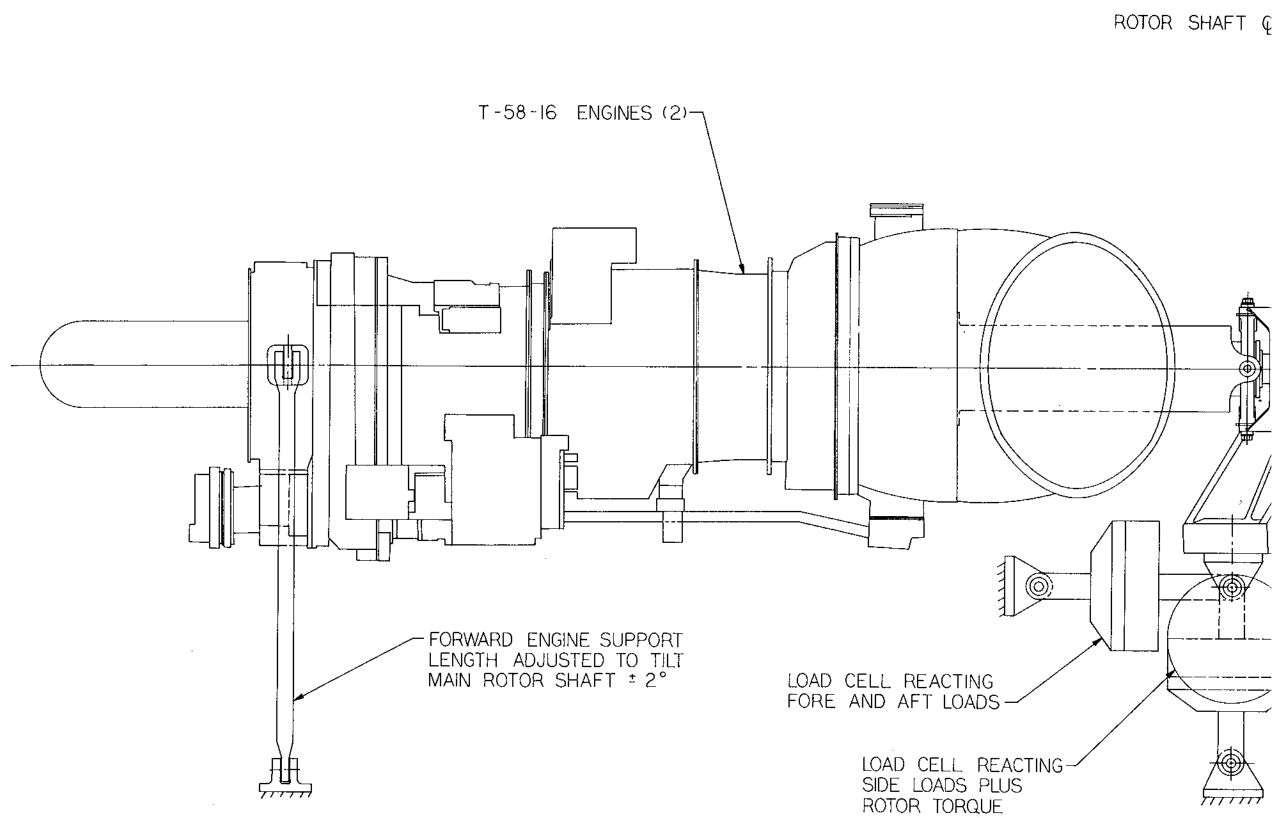
Active Rotor Balance/Vibration Suppression System

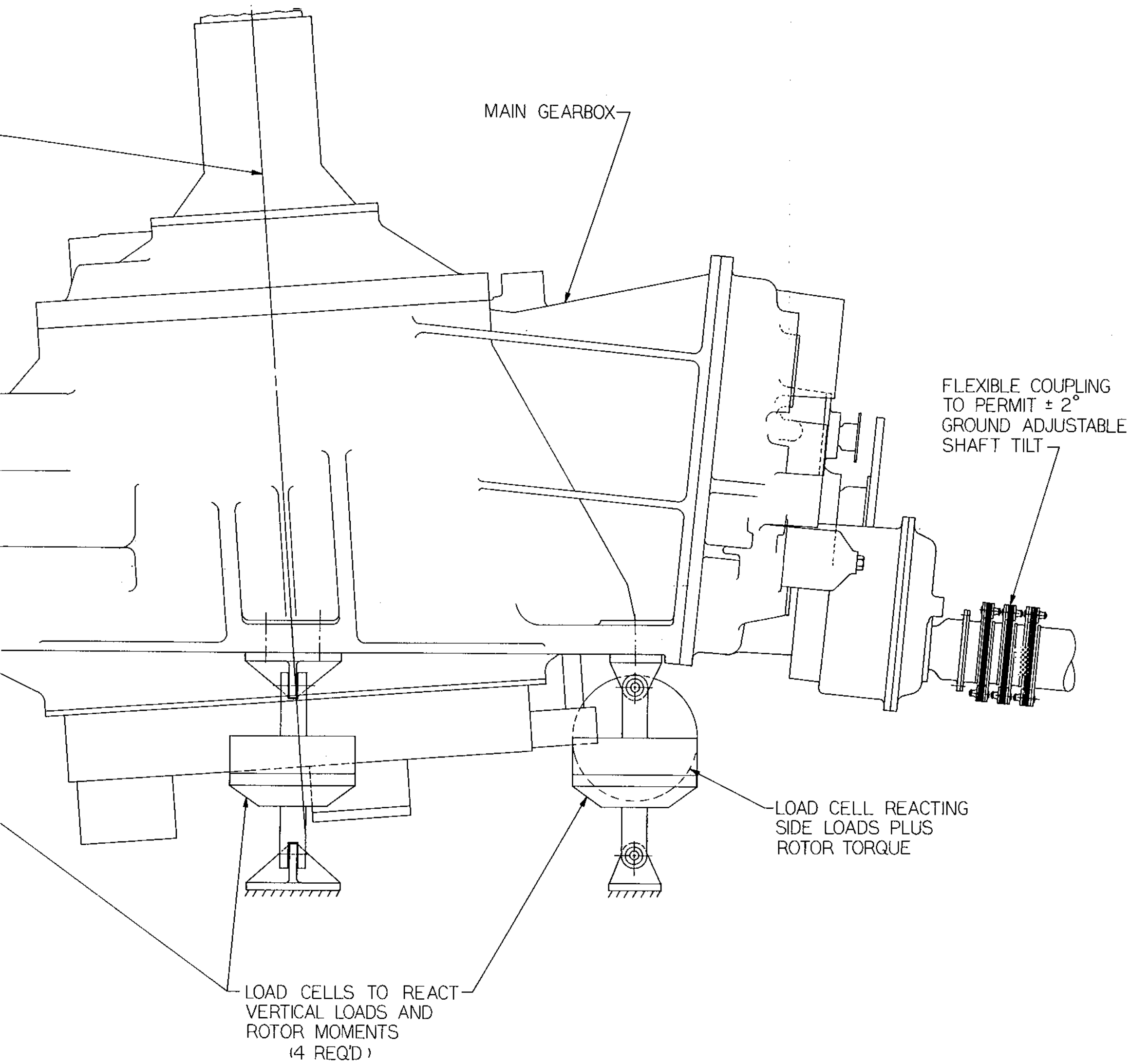
The range of rotors which may utilize the RSRA as a flying test bed combined with variations of tip speed and blade number will produce a wide band spectrum of vibratory excitation frequencies (blade passage frequencies). It is impossible to structurally detune the airframe modes so that they will never be resonant with any vibratory excitation frequencies. Transmission isolation can produce this effect. It must be noted that the term isolation in this context defines a method of vibration suppression. An isolation system for the RSRA is not intended to totally eliminate vibration nor can such a system be designed from a practical standpoint.

Passive isolation systems are limited in that there is a practical lower limit to their flexibility due to control system and engine drive shaft displacement tolerance, thus requiring the use of stops. Wide band passive isolation does not appear practical for an RSRA vehicle since the spectrum of rotors and wide variation of steady rotor forces would tend to bottom the isolation system too often.

Active isolation can provide all of the required isolation characteristics in addition to simultaneous isolation of forces and moments. This is accomplished by active control of static and transient displacements while providing low spring rates for wide band vibration isolation. In addition, the Sikorsky self contained hydropneumatic isolation unit has been shown to act as an accurate load sensing device during a recent NASA supported test program. The total system can therefore serve as a rotor balance by providing a defined load path for measuring steady, transient and vibratory rotor loads.

The proposed configuration of the Sikorsky Active Rotor Balance/Vibration Suppression System is illustrated on page 21. Analyses have been performed which substantiate the ability of this system to isolate the airframe from the simultaneous effects of six rotor forces and moments while providing accurate measurement of principal rotor forces. Full scale ground tests of existing hardware under a current NASA/Army contract will substantiate the systems ability


ROTOR SHAFT ℄
T-58-16 ENGINES (2)
FORWARD ENGINE SUPPORT LENGTH ADJUSTED TO TILT MAIN ROTOR SHAFT ± 2°
LOAD CELL REACTING FORE AND AFT LOADS
LOAD CELL REACTING SIDE LOADS PLUS ROTOR TORQUE



MAIN GEARBOX
FLEXIBLE COUPLING
TO PERMIT ± 2°
GROUND ADJUSTABLE
SHAFT TILT
LOAD CELL REACTING
SIDE LOADS PLUS
ROTOR TORQUE
LOAD CELLS TO REACT
VERTICAL LOADS AND
ROTOR MOMENTS
(4 REQ'D)

to measure rotor forces. The configuration shown contains four canted isolators and three inplane units. The redundancy in the number of canted units is provided so as to decouple the vertical, pitch and roll modes of the isolation system while permitting independent pitch and roll focusing to make maximum advantage of the systems ability to suppress vibrations. The variation in focusing of the canted isolators, without modification of the isolator units themselves, is provided by the adjustable circular mounting plates on the airframe.

The following low risk approach is recommended to achieve the objective of the RSRA without comprising vehicle delivery schedules.

a. Initiate design analyses and development of the Active Rotor Balance/Vibration Suppression System as a side by side effort to RSRA and include installation provisions in the RSRA design.

b. Structurally tune the RSRA so that all airframe modes are outside the N/Rev bands of the following rotors:

   i. Five bladed compound rotor with rpm variations up to 30 percent for the 300 knot mission

   ii. Six bladed variable geometry rotor up to 200 knots

   iii. Four bladed variable diameter rotors.

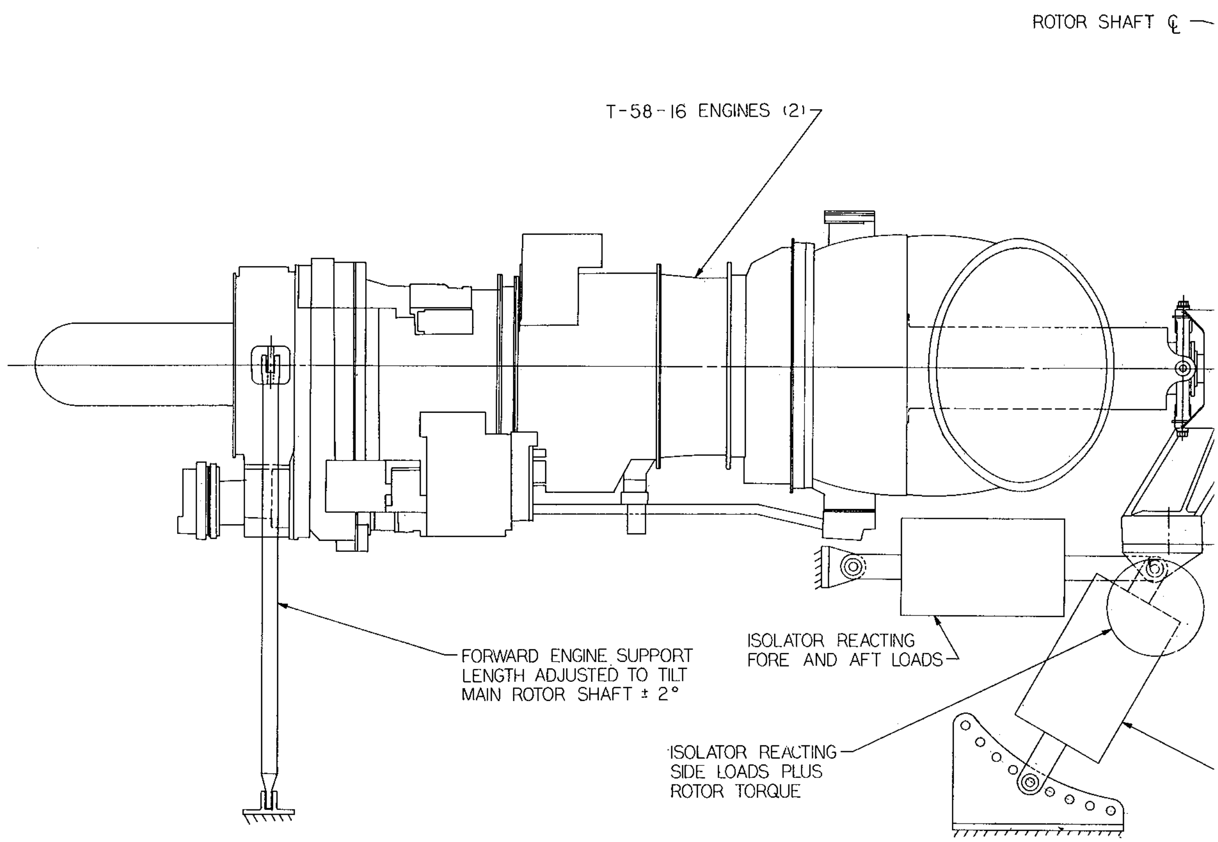
c. In the event that the Active Rotor Balance/Vibration Suppression System development slips from the targeted completion date, the following intermediate plan can be implemented.

   i. Utilize Active Rotor Balance installation hardware to hardmount the RSRA transmission on load cells for testing of rotors defined in (b) and any other rotors whose N/Rev excitation frequencies are in acceptable bands.

   ii. Utilize Active Rotor Balance installation provisions to install a transmission support stiffness control which will permit limited airframe mode shifts so as to extend the operational envelope of the vehicle.

   iii. Install passive isolation if practical for limited applications using Active Rotor Balance installation provisions.

d. Install Active Rotor Balance/Vibration Suppression System in RSRA upon completion of design, fabrication and flight tests on second vehicle. This will expand compatability of RSRA to all rotors including slowed rotors.



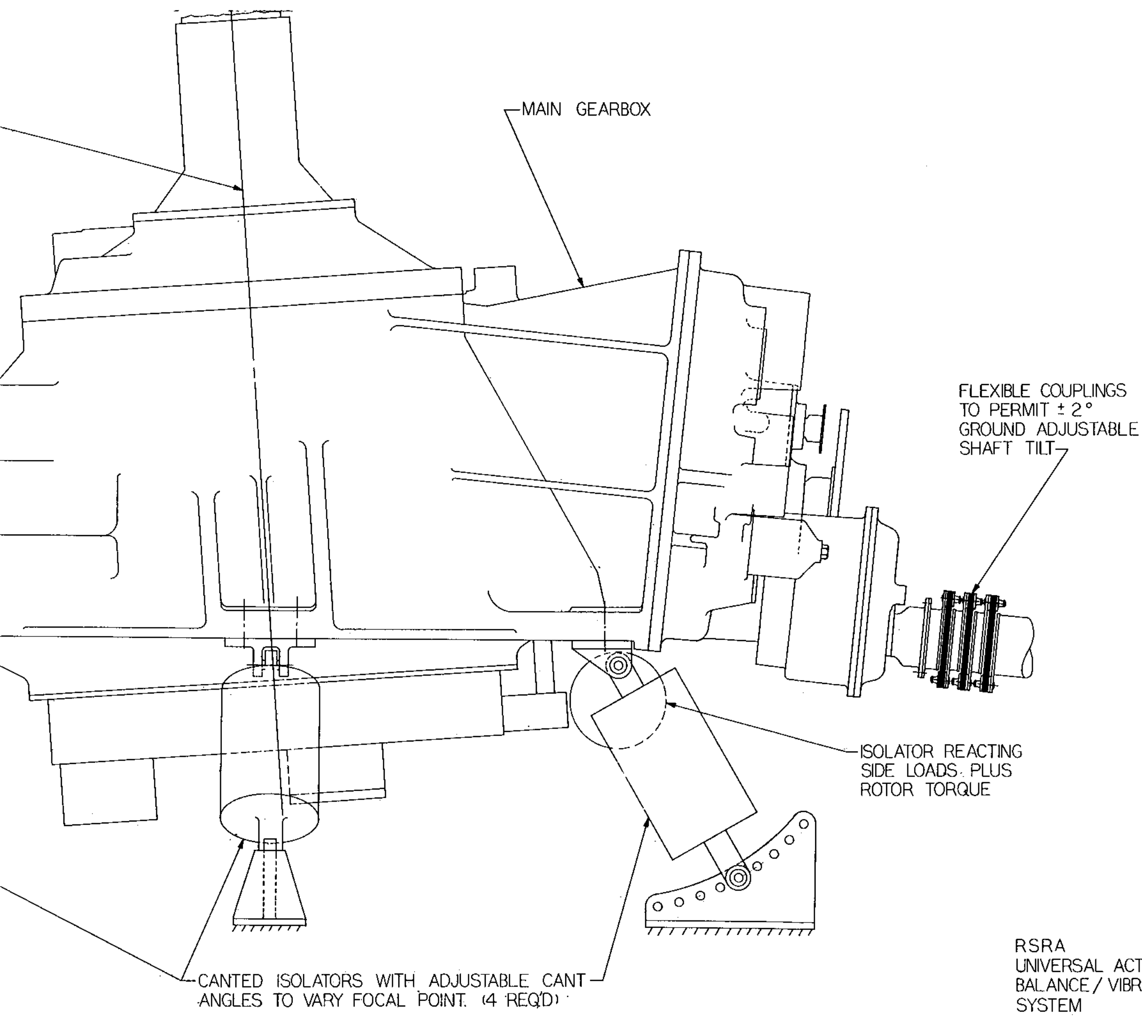
ROTOR SHAFT ℄
T-58-16 ENGINES (2)
FORWARD ENGINE SUPPORT
LENGTH ADJUSTED TO TILT
MAIN ROTOR SHAFT ± 2°
ISOLATOR REACTING
FORE AND AFT LOADS
ISOLATOR REACTING
SIDE LOADS PLUS
ROTOR TORQUE

The Variable Geometry Rotor

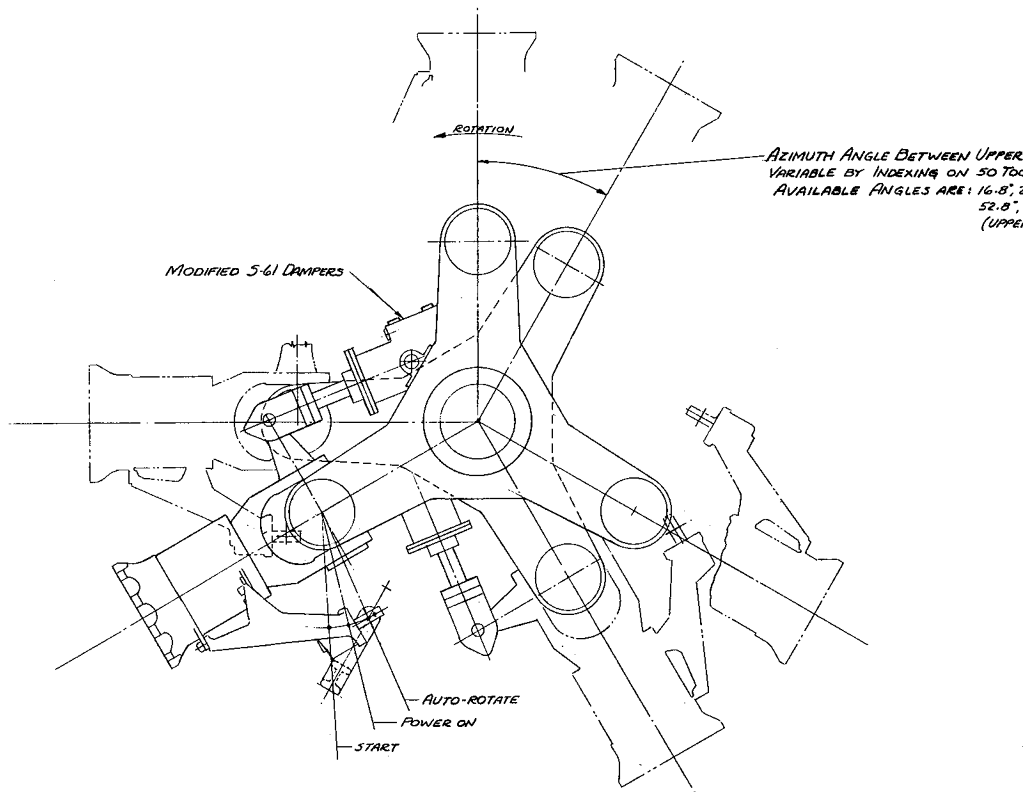
A variable geometry rotor head has been designed to allow features beyond changing blade shapes and sizes such as radius, tips, chords, planforms, etc. The hub has been designed with two three-bladed hubs mounted one on top of the other to facilitate the capability to test blade vertical spacing and azimuthal spacing. Spacers are incorporated to allow three different spacings between the rotor hubs. The azimuth angle between the upper and lower blade sets is changed by indexing on a 50 tooth shaft spline. Both the azimuth and vertical spacing are ground adjustable. Both sets of blades rotate in the same direction and the upper blade set leads. Eccentric pushrods have been designed to allow azimuth angles beyond 45.6 degrees. The rotating swashplate is designed such that the blade control rods can be repositioned with the blades.

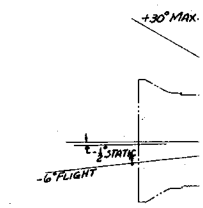
The design is based on making maximum use of existing S-61 components, tooling and inspection gages. The only new parts required are three-bladed hub plates, upper hub shaft and spacers, plus the rotating swashplate and pushrods. The shaft splines, threads, bores, tapers, etc. for the new parts are the same as on S-61 standard parts except that grease lubrication instead of oil will be used throughout the hub assemblies.

The complete assembly will consist of two hub assemblies and three sets of pushrods and spacers to accommodate all three upper hub positions. A drawing of the system is shown on the opposite page.

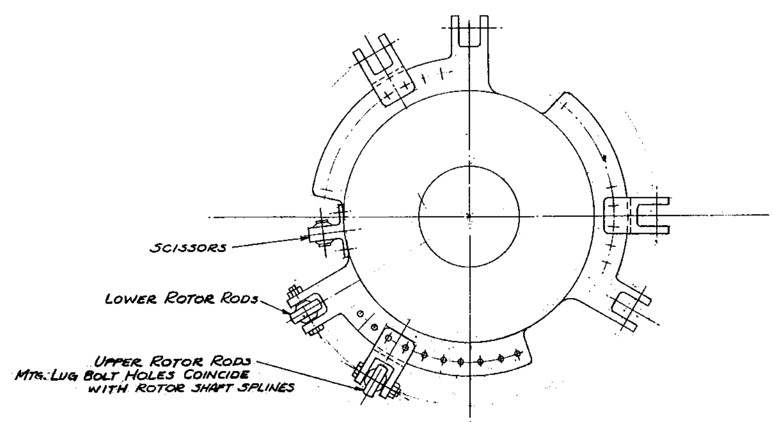


RSRA
UNIVERSAL ACTIVE ROTOR
BALANCE / VIBRATION SUPPRESSION
SYSTEM





S-61 COMPONE



ROTATING SWASHPLATE

DESCRIPTION OF WING AND HIGH LIFT DEVICES

Two wings are used on these aircraft. The first is a large wing for helicopter simulation from 100 to 200 knots. The second is a smaller wing for compound flight at speeds up to 300 knots.

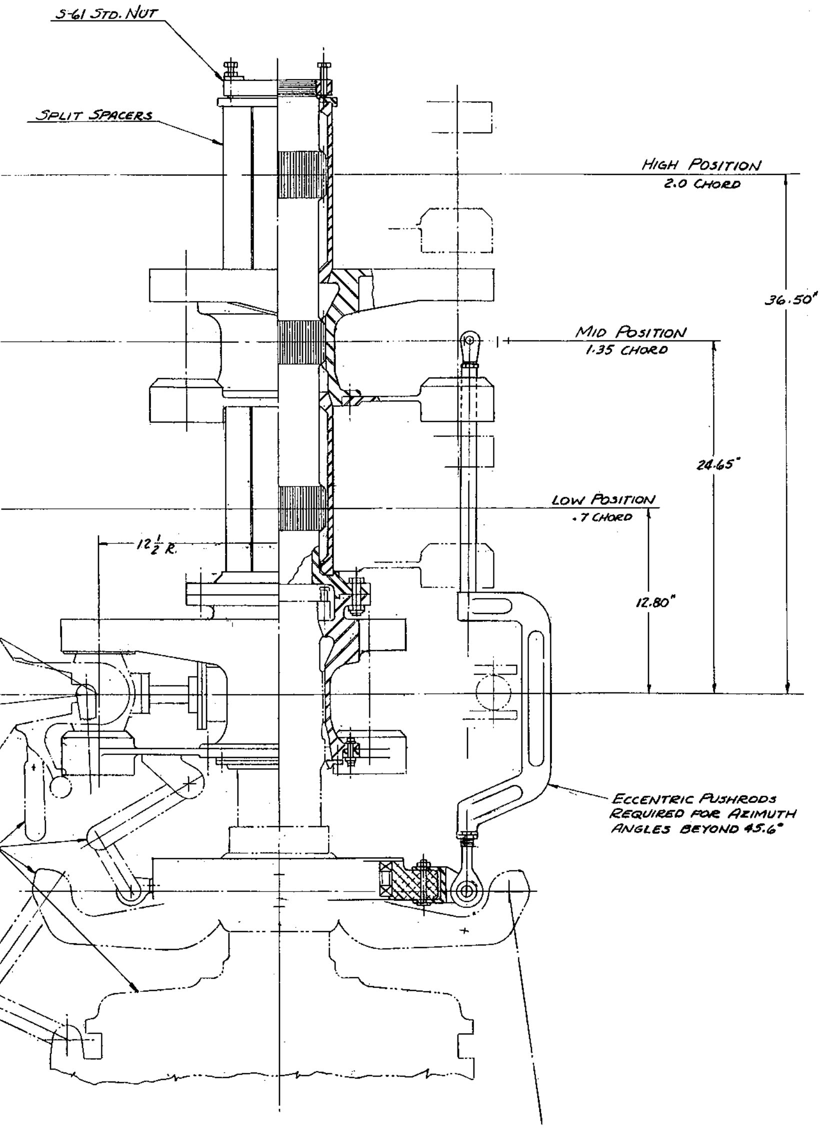
The design of the large wings of both aircraft fulfills the requirement to support the gross weight of the aircraft at 150 knots, sea level, standard conditions, in a clean, unflapped configuration. The stall margin is 20 percent. An aspect ratio of six, zero sweep angle, and a 0.6 taper ratio were selected for the wings to provide the maximum lift at the design condition and also yield the best lift performance with flaps down. The unflapped wing loading is 75.8 lbs/ft^2.

Both large wings are equipped with double slotted trailing edge flaps and leading edge slats. This high lift system provides the capability to unload conventional main rotors to a C_L/σ of approximately .03 at 100 knots with a 20 percent stall margin and complete unloading of the main rotor above 120 knots.

The small wings for both RSRA aircraft were designed to lower the design gross weight required for the 300 knot mission. They were designed to support 100 percent of the aircraft gross weight at 200 knots with flaps down. Split flaps were selected to keep wing complexity and weight to a minimum. The small wings for the high speed compound testing and the large wings for helicopter simulation meet the RSRA requirements as modified for Part II for 300 knot compound testing and helicopter simulation between 100 and 200 knots. With two TF34-GE-2 turbofan engines installed, speeds of both aircraft exceed the 300 knot requirement with either the large or small wing installed. Drawings of the large and small wings of the existing component aircraft are shown on page 25 .

All of the various wing configurations of the two RSRA aircraft have inflight variable incidence. The incidence is varied by three hydraulic actuators which are controlled by a lever in the cockpit. The actuator range is designed to provide the full incidence range required by the wing, in addition to achieving ±10 degrees of effective rotor shaft tilt by varying fuselage incidence. For the all new aircraft which has ±5 degrees of inflight variable shaft tilt, the range of wing tilt necessary to provide enough fuselage angle to yield the ±10 degrees under all operating conditions is 32 degrees. For the existing component aircraft which does not have inflight variable shaft tilt, the actuator range is 42 degrees. The actuator requirements are within the capability of three CH-53A tail pylon fold actuators. The drawing of the wing tilt actuators is shown on page 27.

The capability for performance mapping between 100 and 200 knots is shown on page 26.



PROGRAM	DRAWN BY	G. HOWARD	4-21-72	FUNCTIONAL	AC & SS		
	DESIGNER	"			AD & D		
	SYS DESIGN				STRUCT & MATL		
	SYS ENGINEER				AERO MECHANICS		
	ACFT DESIGN				SD & E		
	CHIEF DESIGN				CHIEF TEST ENGR		
	TASK MANAGER				CHIEF ENGINEER		
					ENGRG MANAGER		
					MFG/MFG ENGRG		
		NAME	DATE			NAME	DATE

REPORT NO. | PAGE NO. | FIGURE NO.

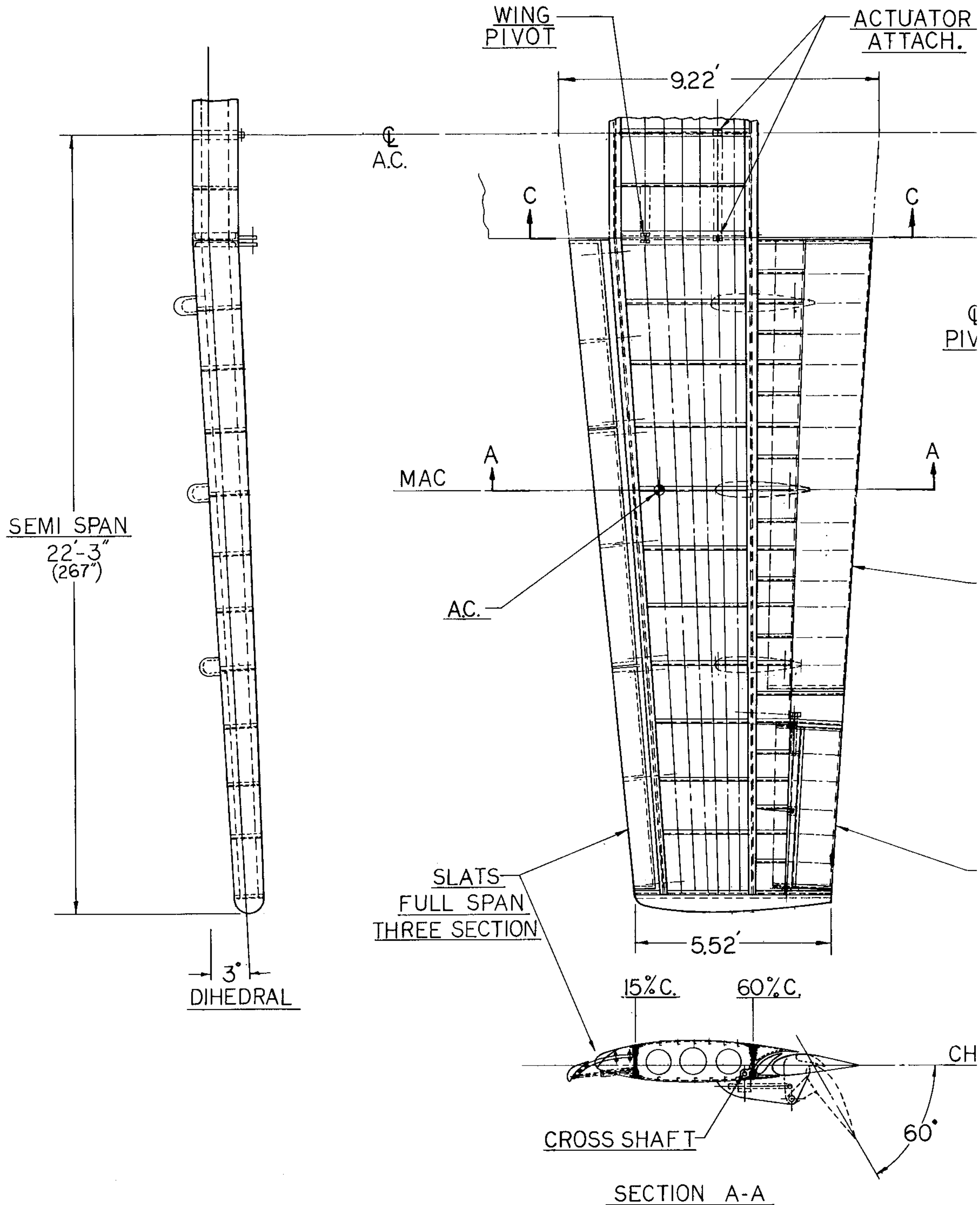
VARIABLE GEOMETRY ROTOR FOR RSRA
(2) 3-BLADED ROTORS - S-61 COMPONENTS
NASA / ARMY

Sikorsky Aircraft | CODE IDENT. NUMBER 78286 | DS-509-3-5 | SCALE 1/4 | WBS No. | SHEET OF | REV.

3692 REV. A.

LARGE WING

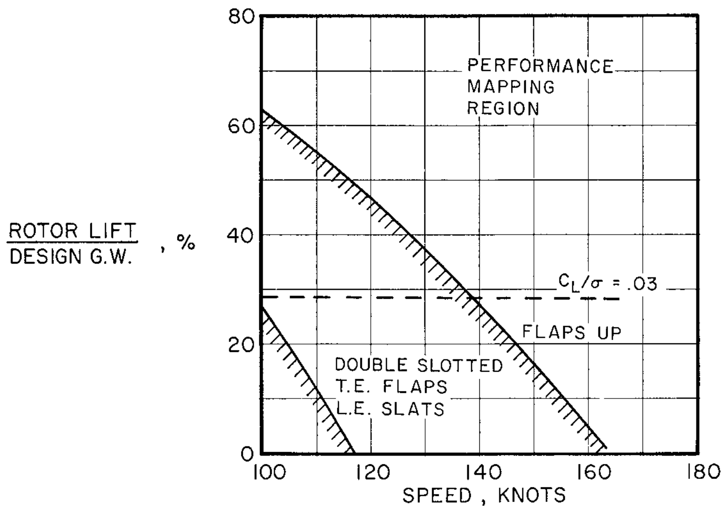
AREA = 332 FT^2



RSRA CAPABILITY FOR PERFORMANCE MAPPING
SEA LEVEL STANDARD

PERCENT LIFT - ROTOR OF DESIGN GROSS WEIGHT
VS. SPEED

LEVEL BODY ATTITUDE



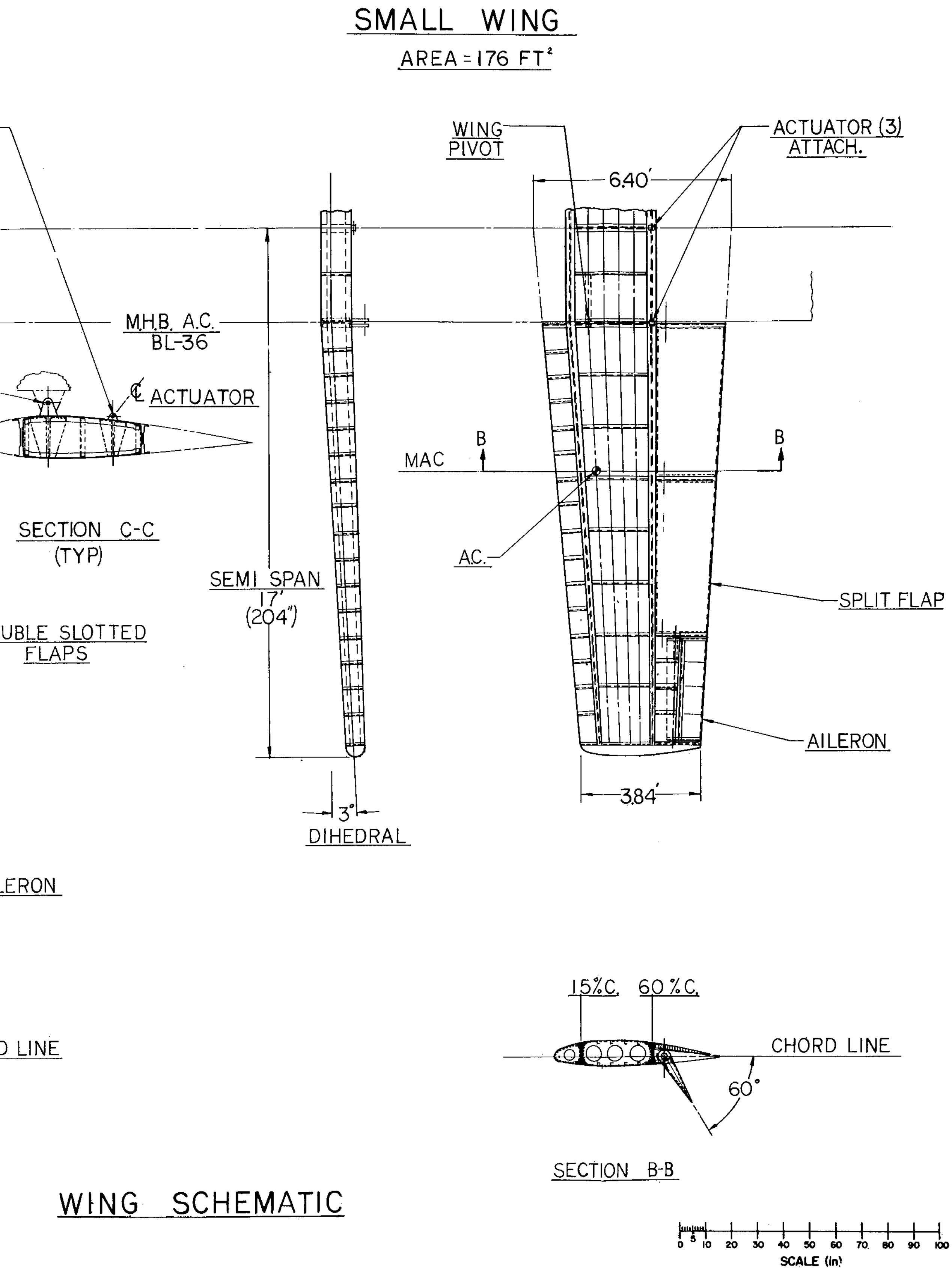
FLIGHT CONTROL SYSTEMS

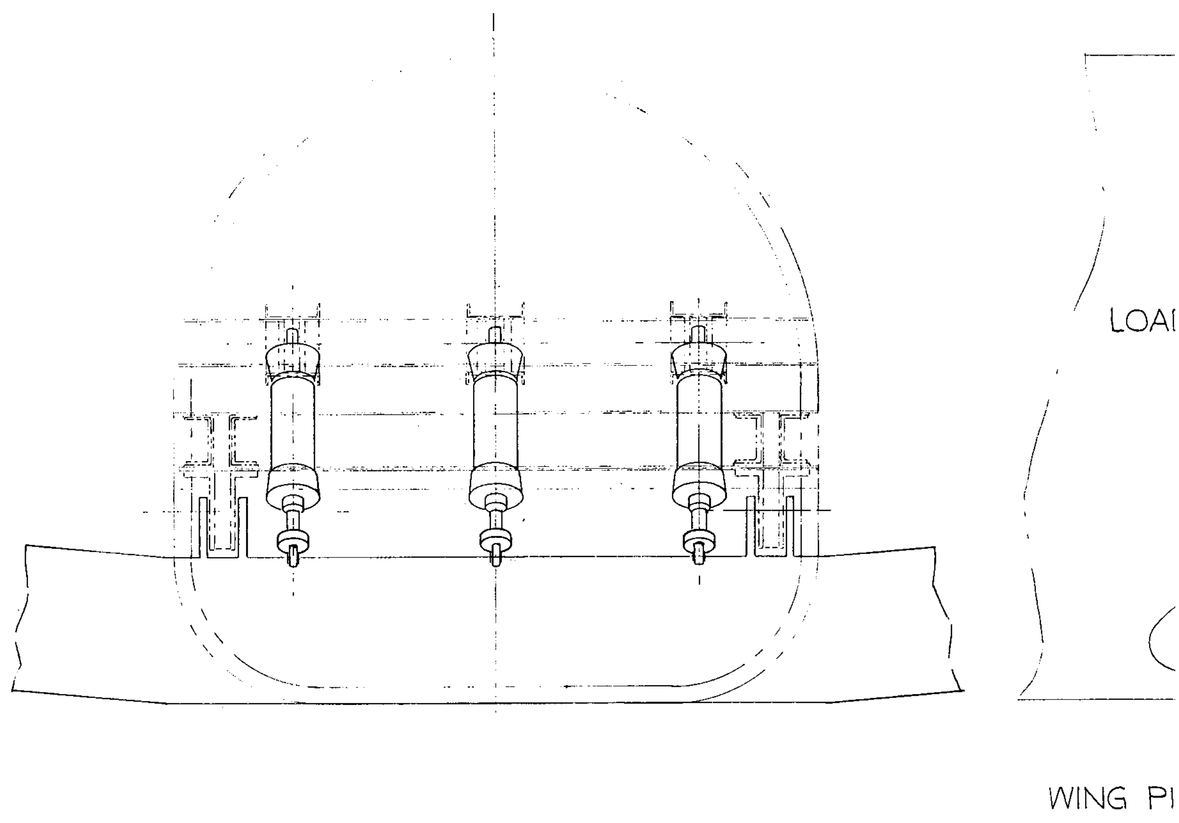
System I Electrical/Mechanical Control System

The baseline flight control system for the RSRA is a "pseudo fly-by-wire" system. This system is a hybrid electrical-mechanical system which has some of the flexibility of a fly-by-wire system with the same reliability of a conventional mechanical system. This system is shown on page 29. The pilot's and copilot's cyclic and collective controls are separated, with the copilot retaining the mechanical link to the control surfaces and the pilot being provided with an electrical control system. The pilot is therefore the evaluation pilot, and the copilot performs as a safety pilot.

Copilot Controls

The copilot's control station is of conventional design. His control inputs are transmitted directly to the surfaces via mechanical links. The cyclic and collective controls can be flown conventionally or can be driven by the Force Augmentation System (FAS). This system also provides the control force gradients and maneuvering feel desirable for precise control of a high speed helicopter. The FAS provides the necessary interface between the pilot and copilot controls. In the normal flight mode, the pilot's control motions are transmitted to the copilot's FAS which then positions his controls accordingly, thus providing mechanical inputs to the control surfaces. The copilot's inputs in the normal mode are sensed electrically and sent to the pilot's FAS in a similar manner to allow for control transfer between the crew with no ambiguity in control position.


SMALL WING
AREA = 176 FT²
WING PIVOT
ACTUATOR (3) ATTACH.
6.40'
M.H.B. A.C.
BL-36
℄ ACTUATOR
B
MAC
B
SECTION C-C
(TYP)
A.C.
SEMI SPAN
17'
(204")
SPLIT FLAP
UBLE SLOTTED
FLAPS
AILERON
3.84'
3°
DIHEDRAL
LERON
15% C.
60% C.
CHORD LINE
D LINE
60°
SECTION B-B
WING SCHEMATIC
0 5 10 20 30 40 50 60 70 80 90 100
SCALE (in)



LOAI
WING PI

Pilot's Controls

The pilot's control station is similar to the copilot's with the exception that his controls are connected electrically to the computer and the copilot's FAS. In any of the test modes, his control motions are sent to the computer for processing and distribution to the various actuators.

Auxiliary Controls

The pilot and the copilot are provided with a set of auxiliary controls with which they may trim the aircraft to a desired test condition. These trim controls may also be commanded by the computer to achieve a computer programmed test condition.

Control Integration

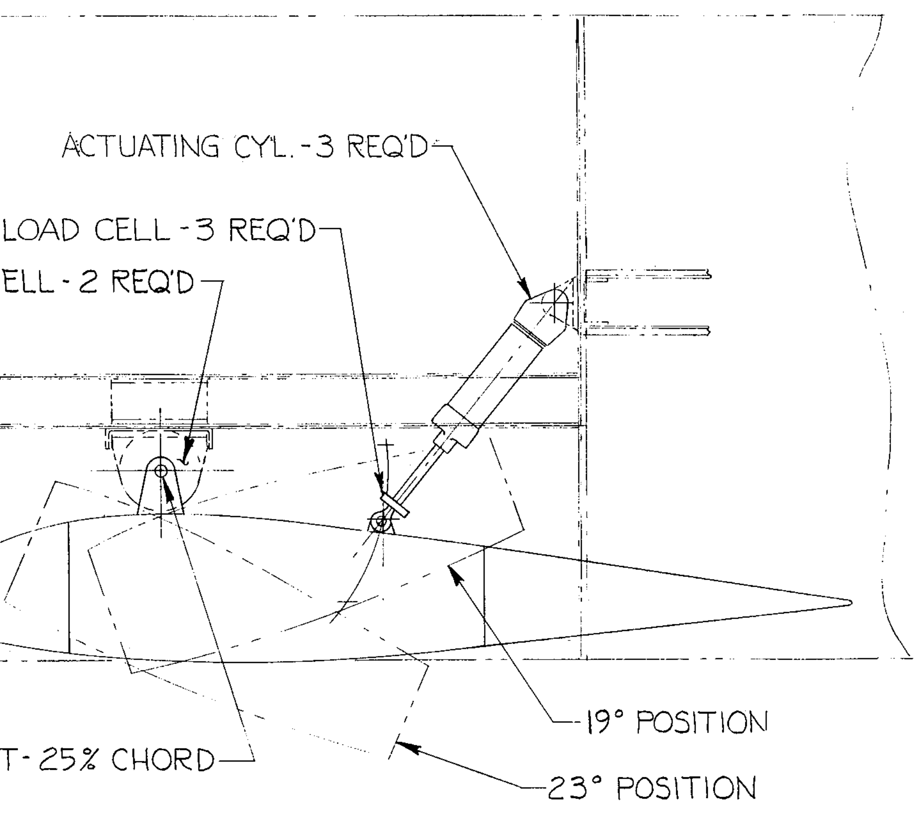
The sensitivity of the rotor and fixed wing surfaces to pilot inputs must be varied to provide a proper control response and complete testing capability throughout the wide speed range. This control integration is accomplished mechanically. The amount of control apportioned to the rotor and flying surfaces is determined by the position of the control integration actuator which is controlled by the crew or the computer. The control integration mechanism is designed to allow the control sensitivity to either the rotor or the fixed wing surfaced to be reduced to near zero, but not simultaneously.

Rotor Controls

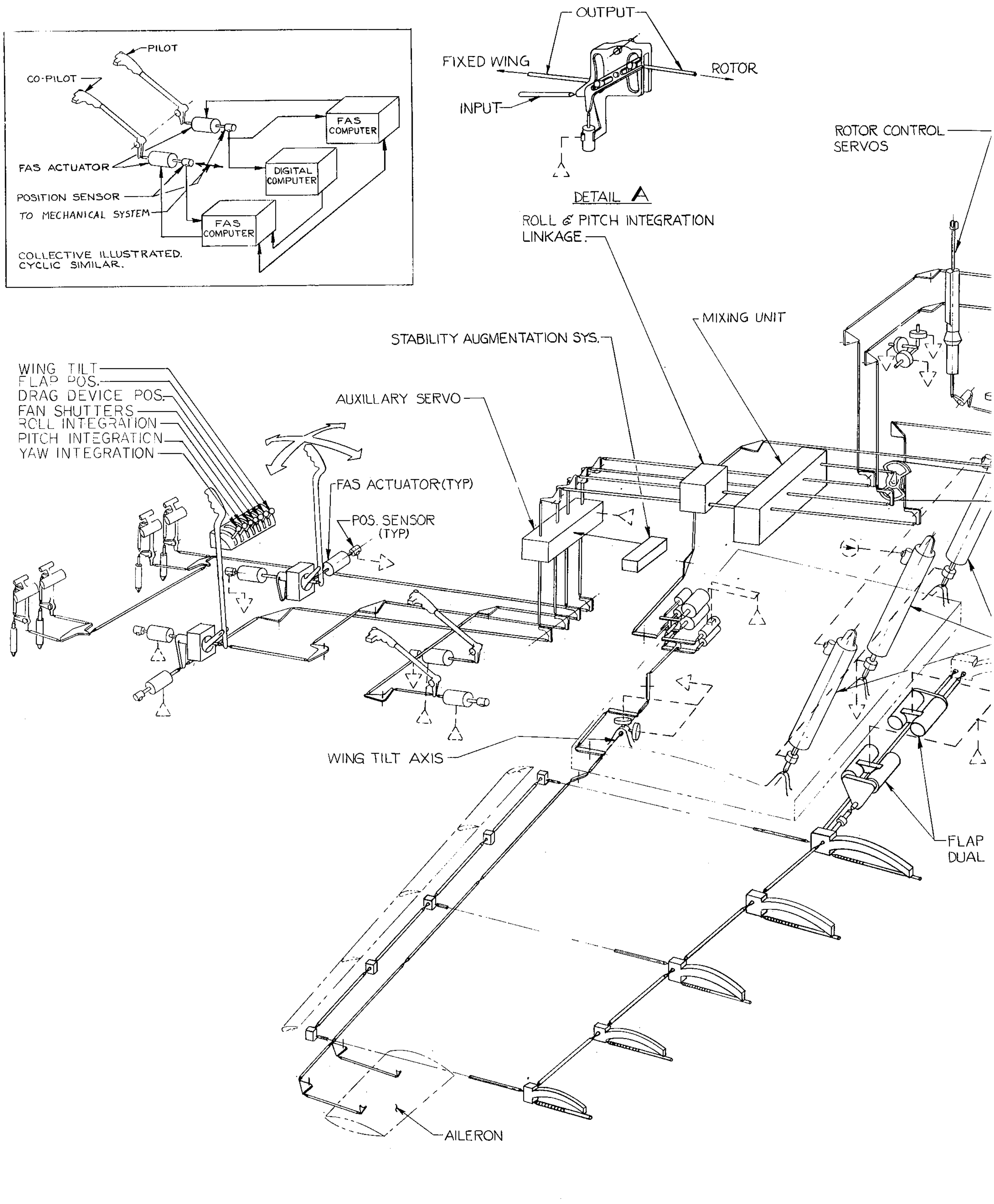
The main rotor is controlled through the standard mechanical control system in the normal flight mode. In the test flight modes, the computer commands the rotor through a limited authority electrical input to the auxiliary servos and the full authority FAS actuators. The high frequency control signals, which are usually small in amplitude are sent to the auxiliary servo. The low frequency trim commands, which may be large in amplitude, are sent to the FAS actuators. These FAS actuators will move the copilot's controls, allowing the copilot to monitor the inputs to the main rotor. The implementation of this scheme is shown on page 30.

Conventional Control Surfaces

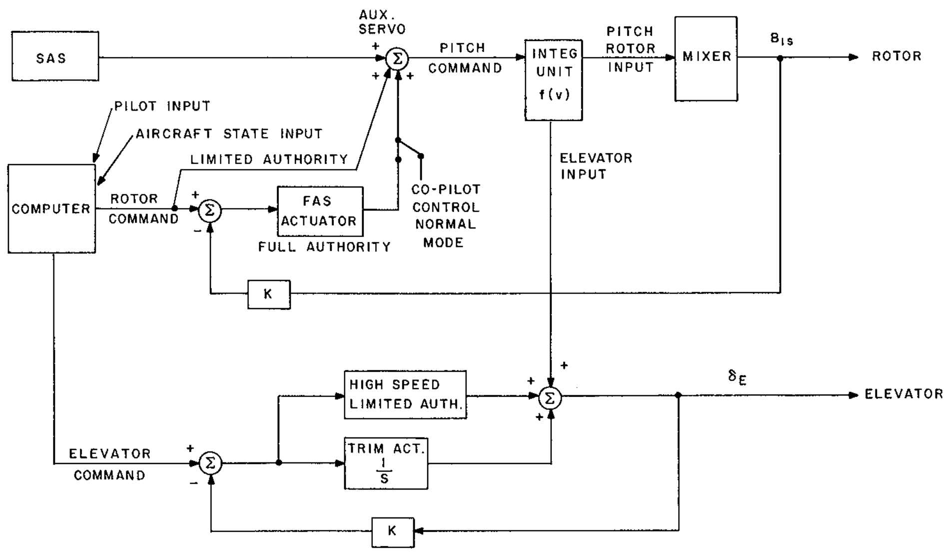
The conventional control surfaces are controlled through the mechanical control system in the normal flight mode. In the test flight modes, the computer commands the surfaces through dual, full authority, trim actuators and single limited authority high speed actuators. The high and low frequency computer commands are apportioned to these actuators in a manner similar to the rotor control described above. The control scheme is also shown on page 30. Control surface position is fed back to the control actuators to provide accurate positioning of the surface and to eliminate the mechanical inputs to the conventional controls caused by motion of the copilot's controls in response to computer generated rotor commands.



RSRA
WING TILT
MECHANISM


PILOT
CO-PILOT
FAS COMPUTER
DIGITAL COMPUTER
FAS ACTUATOR
POSITION SENSOR
TO MECHANICAL SYSTEM
FAS COMPUTER
COLLECTIVE ILLUSTRATED.
CYCLIC SIMILAR.
OUTPUT
FIXED WING
ROTOR
INPUT
DETAIL A
ROTOR CONTROL SERVOS
ROLL & PITCH INTEGRATION LINKAGE
MIXING UNIT
STABILITY AUGMENTATION SYS.
AUXILLARY SERVO
WING TILT
FLAP POS.
DRAG DEVICE POS.
FAN SHUTTERS
ROLL INTEGRATION
PITCH INTEGRATION
YAW INTEGRATION
FAS ACTUATOR (TYP)
POS. SENSOR (TYP)
WING TILT AXIS
FLAP
DUAL
AILERON

RSRA PITCH CONTROL AXIS BLOCK DIAGRAM



Stability Augmentation System (SAS)

A SAS is provided for use during the normal flight mode. The SAS will provide basic aircraft stability through the limited authority auxiliary servo and will move both the rotor and conventional controls as determined by the integration unit. In the test modes, the SAS is put into standby and control is provided by the computer.

Override Capability

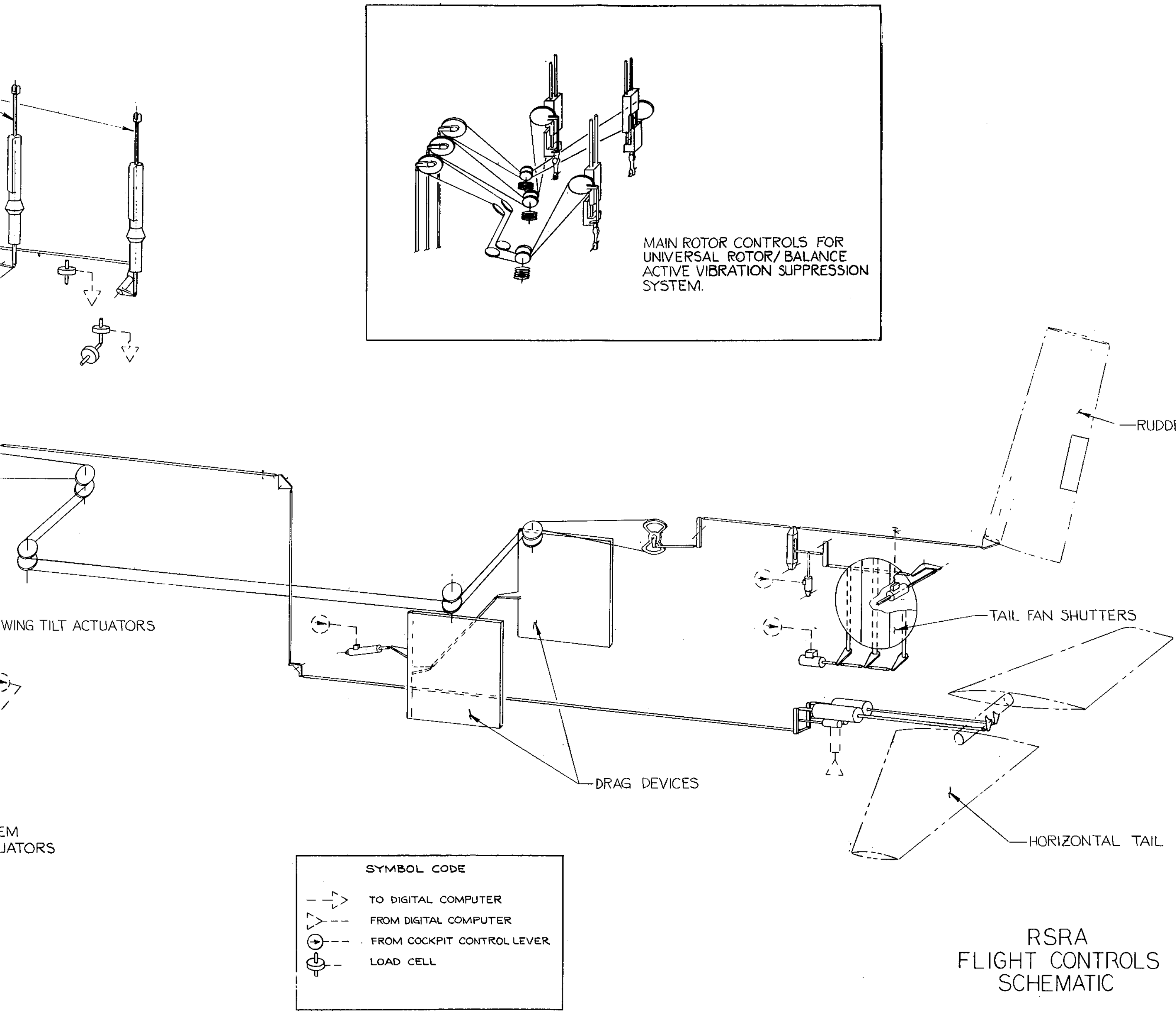
The copilot may override any of the controls at any time by exerting a force on the control to overcome the command force or detent, except for the conventional control trim actuator. Since these are full authority series actuators, they have been made dual to allow monitoring and allow shutdown to prevent remaining runaway time.

System II Fly-by-Wire Flight Control System

The Fly-by-Wire Control System is a quadruple system with built in test and voting logic to provide the required system reliability.

Crew Controls

The crew controls are identical to those of the Base Line Control System.



RSRA
FLIGHT CONTROLS
SCHEMATIC

Control Integration

The control integration is performed in the electronic control unit. The function is automatic and variable with airspeed. The integration functions may be varied with relatively minor hardware changes within the control unit.

Rotor Control Mixing

The mixing of the pitch, roll and collective rotor commands is performed in the control unit. The mixing may be changed to accommodate various rotor configurations with relatively minor changes within the control unit.

Rotor Controls

The rotor control signals are sent from the control unit to quadruple hydraulic actuators which are again monitored to provide the necessary reliability. The actuator stroke is within the standard primary main rotor servos.

Conventional Controls

The conventional control surfaces are controlled in an identical manner to the rotor controls.

Computer and Trim Controls

The computer inputs and the crew trim commands are sent to the control unit for operation of the controls.

Auxilliary Controls

Control of the auxilliary control devices, flaps, drag, thrust and wing tilt, are controlled identically to the Baseline Control System.

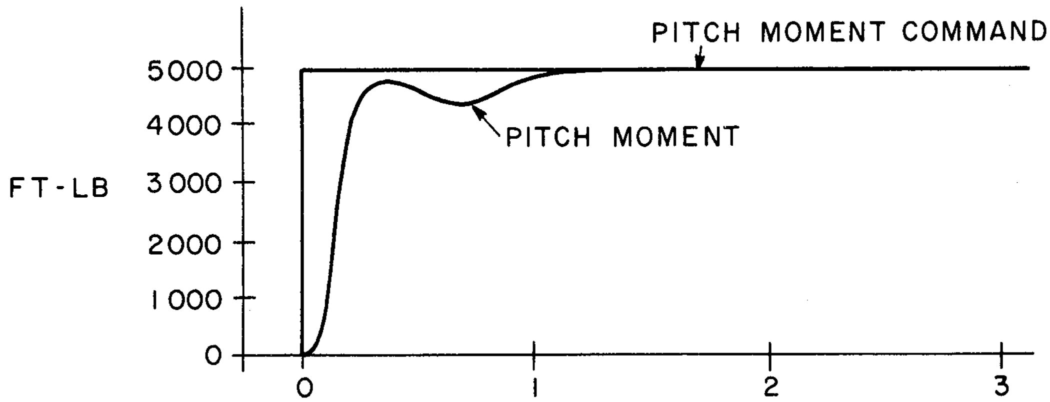
ROTOR RESPONSE TO COMMAND INPUTS

A hybrid computer simulation of a five-bladed articulated rotor revealed that the rotor control response was more than adequate at speeds up to 200 knots. As speeds approached 300 knots, a marked deterioration in rotor stability was noted. The figure on page 32 shows the rotor response to a pitch input at 100, 200, and 300 knots. The 300 knot response shows two distinct dynamic characteristics, a long period aperiodic response and a short period oscillatory response.

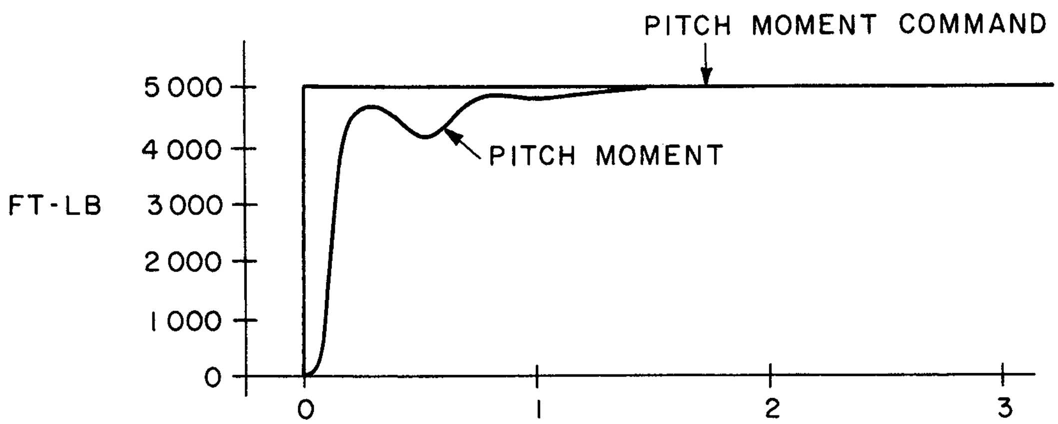
It should be noted that the simulated rotor has no pitch flap (delta three) coupling. This would improve the rotor response at higher speed.

ROTOR RESPONSE TO PITCH MOMENT COMMAND

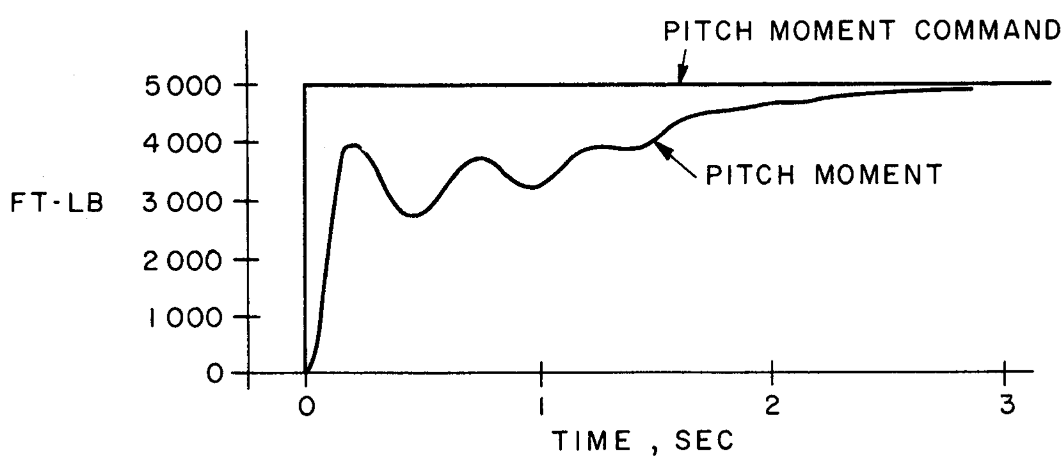
V = 100 KNOTS



V = 200 KNOTS



V = 300 KNOTS



EFFECTS OF SYSTEM COMPONENTS ON PERFORMANCE

The hybrid simulation was used to determine the effects of component characteristics on system stability and accuracy. The characteristics which were studied were time response, hysteresis, and computer solution rate.

Actuator Time Lags

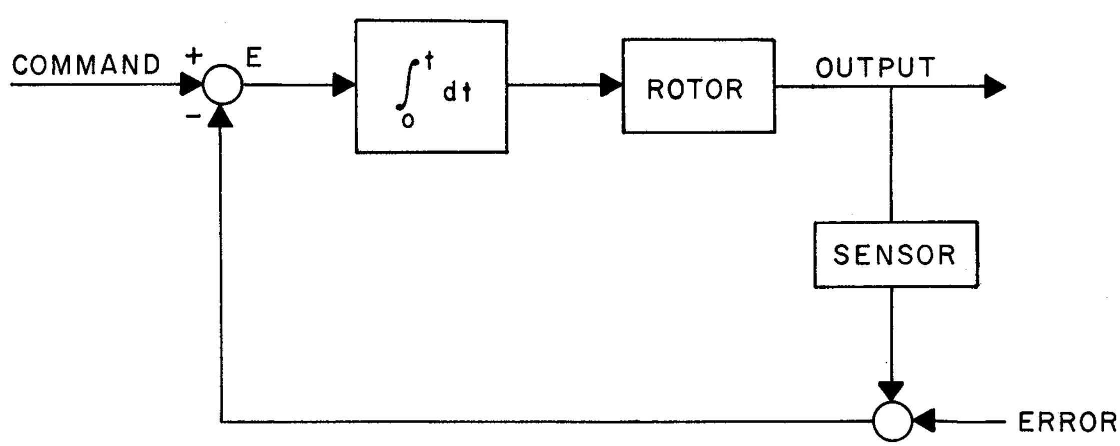
The actuators were modeled by a system of the form $\frac{K}{(.03S + 1)\ (.03S + 1)}$ which is typical of the current helicopter actuator. System performance was satisfactory within 50% of the nominal time constant and the nominal was used throughout the remainder of the study.

Actuator Hysteresis and Deadband

The actuator hysteresis and deadband were increased until performance degradation was noted. This value (4% of total control) was well above the actuator accuracy requirements of current control systems (.5 - 1%).

Sensor Hysteresis and Deadband

The sensor hysteresis and deadband produced one for one error in system accuracy. This was expected since the output of the sensor is used as an error signal to an integrator, and the error must be zero in the steady state. The inaccuracy is then a direct function of sensor inaccuracy. This relationship is shown below.

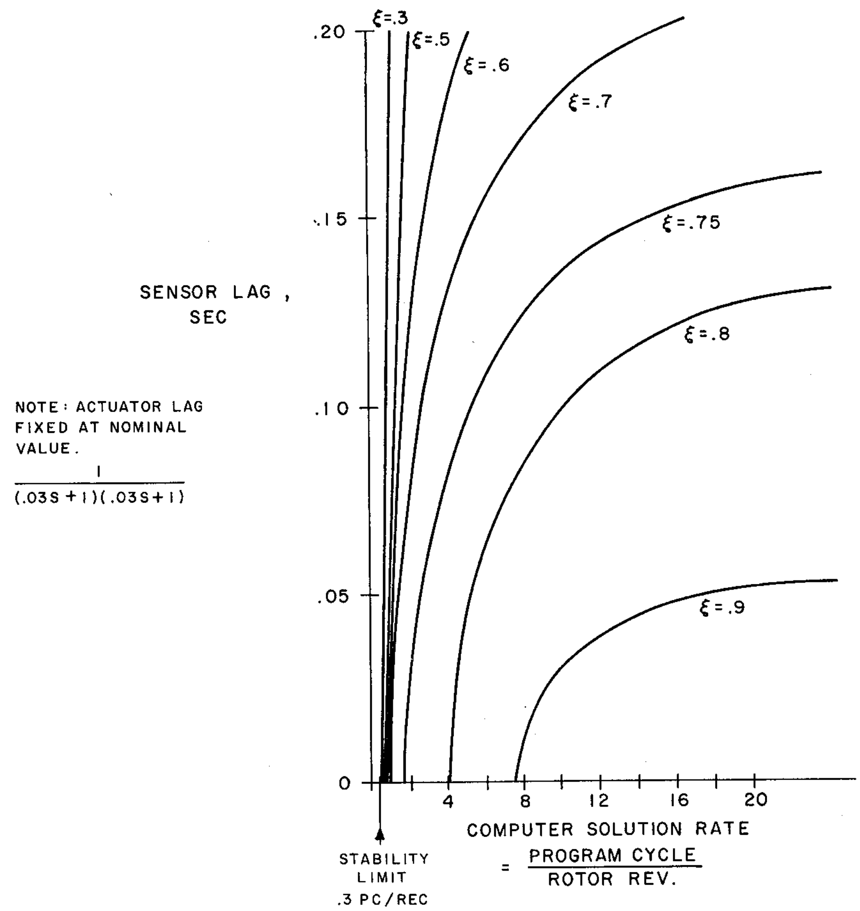


E = O IN STEADY STATE

Sensor Time Lags

The sensor time lags were increased until stability degradation was noted. The figure below shows the change in system damping ratio (ξ) with sensor lag at constant computer solution rates.

SENSOR LAG & COMPUTER SOLUTION RATE EFFECTS ON SYSTEM STABILITY



Computer Solution Rates

The time necessary for the digital computer to solve the control equations was varied to define the lower limit of computer speed. The illustration above also shows the performance degradation with long computer solution times. (low component rotating rate)

The damping ratio (ξ) stabilizes, for a constant sensor time lag, at about 20 solutions per rotor revolution. This coincides with sampled data theory which states that the optimum data rate is 20 times the highest system frequency, which is, in this case 1/rev. This figure shows that the minimum practical solution rate is about 5/rev.

TAIL SIZE AND LOCATION

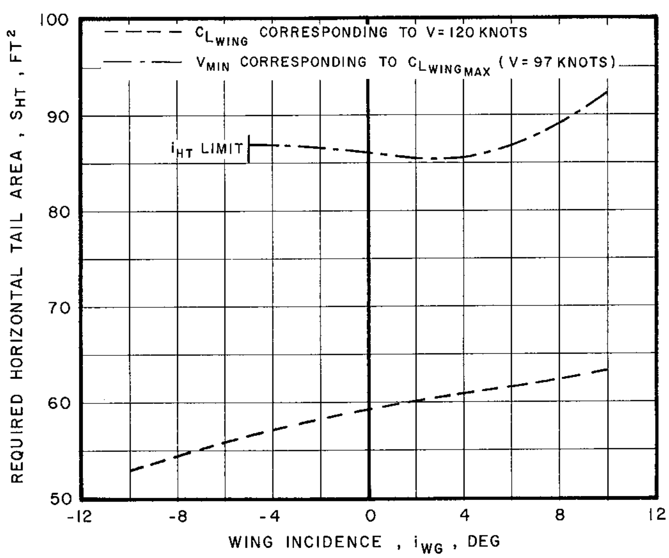
The vertical tail size requirement for these aircraft was based on neutral static directional stability. The vertical tail location of the basic configuration was used, and the analysis was done about the aft cg location. The lift properties of the vertical fin were determined analytically and an estimated correction accounting for the presence of the fan was included. The yawing moment derivative with sideslip for the fuselage was calculated; and this quantity was balanced by the vertical tail. Sidewash and dynamic pressure losses at the vertical tail were included in the analysis. The area needed to obtain neutral stability was found to be 83% of the actual area designated for the aircraft.

Typically, helicopter vertical tail size is determined based on neutral stability. Some positive stability margin is desired, and this is usually provided by the tail rotor. The RSRA design employs a fan rather than a tail rotor, and the positive margin should be available from this device. Presently, little empirical data exists describing the effect of the fan thrust on the lift curve slope of the fin. It is known that the slope decreases as the fan thrust is reduced. Therefore, the present vertical tail size should be considered adequate at most unless the fan duct is closed with a shutter mechanism. The necessity of either increased area or covering the fan openings to provide for neutral stability in event of a failure of the fan should be further investigated when more data become available on the effect of fan thrust on fin lift. For the present, a fan shutter mechanism is assumed to be included in both aircraft designs.

The design condition for sizing the horizontal tail was the ability to land the RSRA at design gross weight in the pure conventional aircraft mode. For this condition, it was assumed that the main rotor produced only drag. The most critical configuration selected was the forward cg with full flaps. Two speeds were studies; 120 knots and 95 knots. The latter is a minimum speed corresponding to the maximum obtainable lift coefficient of the flapped wing. The resulting horizontal tail size requirement as a function of wing incidence appears on page 36. Plots are shown for the tail operating at its maximum lift capability, and at lower lift coefficients which allow for control and stall margins. Horizontal tail incidence limits of +20 deg to -25 deg restrict the lift producing capability of the tail in some instances; these conditions are indicated in the figure.

The data presented were gathered by determining the lift at the horizontal tail needed to counter the pitching moment produced by the wing, fuselage, and rotor. Thus, these data represent trim criteria. The RSRA exhibits positive pitching moment with angle of attack stability for the landing cases studied. Neutral stability about the aft cg for the unflapped wing and thrusting main rotor condition was also investigated and found to yield a horizontal tail size requirement of 43.5 ft^2. Thus, the landing condition is the most critical for tail design. Dynamic pressures losses, fuselage downwash,and induced flow at the tail due to the bound and shed vortices of the wing were all considered in the horizontal tail analysis.

RSRA HORIZONTAL TAIL SIZING BASED ON LANDINGS WITH FULL FLAPS IN PURE FIXED WING MODE (INCL. DRAG FROM ROTOR)



ANTI-TORQUE DEVICE

The anti-torque device considered for the RSRA in this study is the fan-in-fin configuration presently being developed for the S-67 helicopter under a U.S. Army funded program. The fan must deliver a thrust made up of two basic components: an anti-torque requirement, and a yaw response requirement. The hovering gross weight configuration of the RSRA was studied, and the total fan thrust requirement for this case was compared to the fan thrust requirement of the S-67. The directional characteristics specified in MIL-H-8501A were used to determine desirable yaw response handling qualities of both the RSRA and S-67.

Based on the yaw response and anti-torque requirements, it was found that the total fan thrust required for the RSRA was slightly greater than the fan thrust required for the S-67. Since the RSRA hover requirement is at sea level, 95°F compared to the S-67 requirement at 4000 ft, 95°F, the operating C_T/σ of the fan for the RSRA will be virtually the same as the S-67. Thus the fan-in-fin as developed for the S-67 should provide sufficient thrust to meet the RSRA requirements considered in this analysis.

LANDING GEAR DESIGN

Adaptation of Existing Main Landing Gear to RSRA

Preliminary investigation has been made to evaluate the feasibility of adapting an existing aircraft main landing gear to the RSRA. The following table compares the RSRA main gear with potential candidates. All gears considered are fuselage mounted.

	GROSS WEIGHT lbs	V_v (fps) (a)	V_1 (kts)	Braking (ft/sec^2) (b)	Tread (ft) (c)	Weight on MLG (%) (c)
RSRA	25000	8	150	8	10	89
LTV A-7A	32500 - 42000	10	150	10	9.6	81
LTV RF-8G	29000	10	150	10	9.6	84
Lockheed F-104	29500	10	150	10	8.65	92

Notes:
a. MIL-A-8862 specification value used for candidate aircraft
b. MIL-W-5013 specification value used for candidate aircraft
c. Candidate aircraft data estimated

Installation and weight data on candidate aircraft are presently being reviewed. Because of the unique requirements of RSRA, which require the landing gear to have both helicopter and fixed wing characteristics, it presently appears doubtful if an existing gear can be used. The Part II designs are using an all new landing gear. Final selection of main landing gear (whether new or existing) will be made after a further review of available data.

Landing Gear Configuration

A tail gear configuration has been selected for the RSRA based on its having the following advantages over a nose gear configuration:

- minimum weight solution
- protects tail fan-in-fin
- improved braking
- easier structural integration

The tail gear disadvantages of

- tendency to ground loop in cross wind landing or uneven braking
- Potential for nose-over

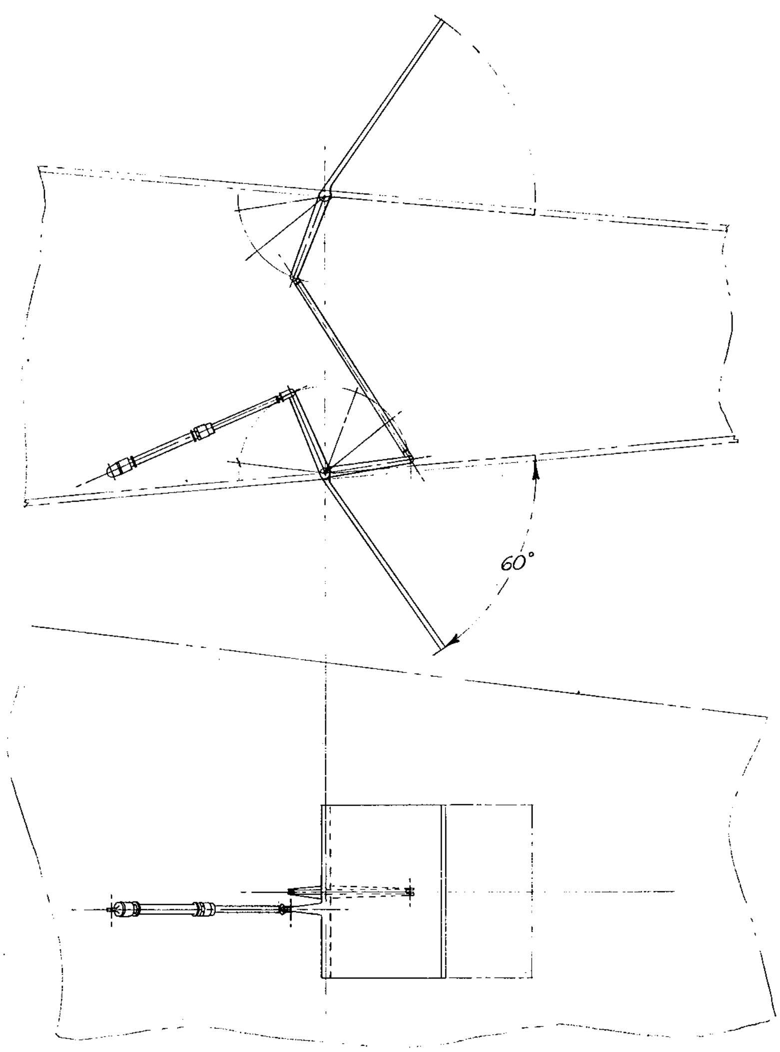
are considered to be manageable for the RSRA and do not outweigh its advantages.

RSRA DRAG BRAKES

The split plate drag brake, located on the sides of the aft fuselage was selected because in this position there was enough brake area available, and the design yielded good test flexibility, a minimum of undesireable moments and comparatively easy structural integration. The installation of the brake and its actuation is shown below.

The brakes on each side of the aircraft are extended by a single actuator. A CH-53A upper ramp door actuator can be used for this task. The brake position is set by the pilot.

With this brake, both RSRA aircraft will be capable of simulating any historic helicopter of gross weights up to 30,000 lbs.



RSRA
DRAG DEVICE
MECHANISM

BALLAST REQUIREMENTS

Both versions of the RSRA are designed with two ballast bays; one under the cockpit and the other in the forward section of the tailcone. Each bay has a 1000 lb ballast capacity. This configuration provides the following approximate total center of gravity shift.

GROSS WEIGHT	TOTAL CG SHIFT - INCHES	GROSS WEIGHT	TOTAL CG SHIFT - INCHES
18000	15.1	26000	10.4
20000	13.6	28000	9.7
22000	12.3	30000	9.0
24000	11.3		

AIRCRAFT RELIABILITY

Existing Component Aircraft

Rotor System

Experience with the S-61 rotor and blades is extensive on SH-3A, CH-3C, HH-3E, S-61N, SH-3D configurations performing varied missions including air rescue, anti-submarine warfare, cargo, commercial passenger carriers, and Apollo recovery. Historical data and established reliability values verify the high reliability of these assemblies. Exceptions to this are the S-67 Fairings and the Blade Severence System, both of which have not received a detailed reliability analyses. In particular, the severence system would require a complete safety-reliability analysis during the aircraft design program.

Wing Group

The small wing does not present significant reliability problems. The large wing with all the added controllable surfaces will require a detailed reliability analysis of each control relative to failure modes and redundancy. This system is similar to conventional fixed wing aircraft, and no unusual problems are anticipated.

Anti-Torque System

Reliability trade-off analysis was part of early studies of several fan designs and they are currently being made on the S-67 fan-in-fin program. Full failure mode and effect analysis will be completed and reliability values on this portion of the aircraft system should have been established in time to be included in the RSRA detail design,construction and testing.

Tail Surfaces

These assemblies should pose no problems in defining reliability criteria. Reliability of fixed tail surfaces on S-61 helicopters has been excellent. S-67 fan-in-fin tail design will have been fully analyzed and documented prior to RSRA.

Body Groups

Reliability studies completed for the S-67 indicates no unusual problems in the basic airframe. Detailed analysis will have to be done during the aircraft design phase on the additional instrumentation, controls, and escape system.

Alighting Gear

Sikorsky has been designing and building retractable helicopter landing gear longer than any other helicopter manufacturer with experience beginning on production S-56 helicopters in the mid-1950's. Most Sikorsky helicopters designed and produced from then on have had retraction or kneeing alighting gear systems. Historical and reliability criteria is established, and no unusual problems are anticipated.

Flight Controls

Reliability failure mode and effect analysis and trade-off studies are most important in evaluating the flight control systems and complete analyses of all control systems will be required. The S-61 control systems are well proven, however the additions of fixed wing controls, fly-by-wire and computer increases the complexity of the system requiring greater emphasis on reliability analysis.

Drive System

Roller Gearbox development included extensive detailed reliability analysis and is expected to be fully matured and have proven reliability. Tail drive shaft is standard with extensive historical data to prove high reliability. Tail Gearbox will be analyzed with anti-torque fan.

Onboard Data System

The importance of this system to the mission of the RSRA justifies a reliability program during aircraft design with emphasis on redundancy. The design is straightforward and no unusual reliability problems are anticipated if reliability is addressed from the start of the program.

Hydraulics

Current S-61 hydraulic systems have been purged of reliability sensitive parts and are proven systems.

<u>All New Aircraft</u>

Systems from S-61 or S-67 helicopters have been exposed to reliability analyses and operational experiences that insures high reliability. New systems and new applications of currently evaluated systems will require extensive reliability studies during the design phase to insure adequate operational availability to perform the required tasks to be imposed on the RSRA. If reliability is addressed from the beginning of the program, no unusual reliability problems should be encountered.

AIRCRAFT SAFETY

The Part II aircraft were reviewed for safety by a preliminary hazard analysis. The results were qualitative statements on the pros and cons of the two aircraft when compared, and also where both could be improved in Part III.

The direct comparison of the two aircraft resulted in the existing component aircraft being "safer" as it was felt that this aircraft has proven components and fewer innovations (in-flight variable shaft tilt, "T" tail).

The safety review also suggests that both aircraft could be improved in Part III by increasing fuselage crashworthiness in the cockpit area by increasing the structural depth below the cockpit.

From a safety standpoint, the mechanical backup for the fly-by-wire system was preferred over the "pure" fly-by-wire system. It was also felt that the blade severance/ejection system was a high risk item until it is proven.

CREW ESCAPE SYSTEM

Rotor blade severing plus the yankee escape system of Stanley Aviation Corporation has been selected from several escape system design concepts for both aircraft. Other escape systems considered for their feasibility, weight and availability were downward or sideward escape to avoid the main rotor, capsule ejection, and manual bail out.

The rotor blade severing system is designed such that each blade is severed just out board of its cuff by a flexible linear shaped charge which is attached externally without blade modification. The charge is detonated by pulling a handle in the cockpit which starts a confined detonation stimulus. This signal is transferred to the rotor through intermediate lines and a rotating transfer unit. As presently planned, the blades are severed simultaneously by a primary system with a redundant backup system designed to fire after a delay of one rotor revolution (0.3 seconds).

The yankee escape system provides escape by using a rocket, attached to a parachute type harness, which is launched out of the vehicle. As the rocket is fired and the canopy section over the crew removed, the seats travel up rails to the edge of the aircraft. The seat pan drops to a vertical position and the rocket pulls the men out of the vehicle. After the men are clear of the vehicle, the escape system deploys a parachute. The yankee system has made 40 successful escapes to date.

The rotor blade severing plus yankee escape system provides both pilots and the third crewman with a zero altitude - zero to 300 knot escape envelope. The basic technology for rotor blade severing has been demonstrated and the yankee escape system is operational.

RSRA - ACOUSTICS

The noise signature of the RSRA vehicle was calculated using a combination of the simplified calculation procedure developed by Lowson and Ollerhead[1] and the broadband noise equation and spectrum presented by Schlegel, et al.[2] This program has been found to correlate well with multibladed rotor systems in the thrust range 15,000 lb to approximately 60,000 lb. The program calculates main rotor rotational and broadband noise. Another program, based on current work being done on low noise fan propulsors, was used to calculate the noise generated by the fan-in-fin anti-torque device. The procedure is not exact, however calculated and measured Perceived Noise levels have shown reasonable agreement. The main rotor dominates the spectrum, however the fan blade passage harmonics will be dominant in the higher frequency bands.

Since the harmonics are tone components, they can be separated from the main rotor spectrum during data analysis thereby permitting an accurate assessment of main rotor noise.

Engine noise was calculated using a semi-empirical method developed at Sikorsky Aircraft with the aid of Pratt and Whitney Aircraft. The procedure was derived from data measured on several engine types and manufacturers, including G.E., Allison, and Pratt & Whitney. The calculated levels have been adjusted to account for sound attenuation resulting from nacelle acoustic treatment of the TF-34 propulsion package; the resulting levels are well below main rotor noise. Sideline PNL is 90 PNdB and forward radiated PNL is 85 PNdB at 500 feet from the aircraft during IGE hover.

The tone corrected perceived noise level (PNLT) as a function of time during take off was calculated at several points on the ground. The points are on a 500 foot equal distance ground contour; that is, a contour that is the locus of points on the ground which are 500 feet <u>from the aircraft</u> as it preceeds along its flight path. The point on the ground where X = 900 feet from the take-off point and Y = 354 feet to the side of the flight track is the critical point since calculations have shown the maximum PNLT (94.8 PNdBT) occurs here. Applying the FAA Effective Perceived Noise Level (EPNL) calculation method[3] at this point results in the value of 93.0 EPNdB, 2.0 EPNdB below the desired limit of 95 EPNdB for the all new aircraft with the fan-in-fin installed. The existing aircraft with a lower rotor tip speed, and larger radius will have a lower EPNdB.

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[1] M.V. Lowson and J.B. Ollerhead; "Studies of Helicopter Rotor Noise;" USAAVLABS TR-68-60, January 1969.

[2] R. G. Schlegel, R.J. King; H.R. Mull; "Helicopter Rotor Noise Generation and Propagation;" USAAVLABS TR-66-4, October 1966.

[3] W.C. Sperry; "Aircraft Noise Evaluation; "FAA Report No. FAA-NO-68-34; September 1968

MODIFICATIONS FOR ALTERNATE ROTORS

A summary of the aircraft modifications required to accommodate certain types of test rotors are listed on Page 44. Topics considered are drive system modifications, engine modifications, control system modifications, and whether an RPM variation of greater than 30 percent is required.

All of the single rotor shaft driven concepts could use the same main gearbox and engine installation. The rigid counterrotating coaxial rotor would require a new gearbox, but possibly could use the same engine installation, if the gearbox were designed to be compatible with it. The jet flap rotor would require a new engine installation with the shaft engines being replaced by gas generators. The mechanical drive system would not be needed, but a structure would have to be designed to support the rotor and provide the required gas flow.

All of the concepts will require some modification to the rotor control system. The variable geometry rotor requires modifications to the rotating swashplate to provide the variable blade azimuth position, and various length pushrods to provide the variable vertical positioning of the upper rotor hub. The rigid coaxial requires major modifications to the control system to provide control for two rotors. The variable diameter rotor, the variable twist rotor, and the slowed rotor can all use the baseline system with minor modifications. With the variable diameter rotor, a separate control must be provided for the rotor diameter. For the variable twist rotor, separate control of twist is required; this may include a second swashplate assembly or other similar device. The jet flap rotor will require a complete new control system, the details of which will depend upon what specific control concepts are being considered.

The final item on the chart considers whether an RPM variation of over 30 percent is required in the operation of the rotors being considered. If such an RPM variation is required, some type of active vibration suppression system will be needed to generate power over a wide range of RPM's.

SUMMARY AIRCRAFT COMPARISON

The chart on page 45 summarizes the design differences between the two aircraft. During Part II the two designs have become quite similar, using the same propulsion systems, wing, anti-torque fan and basic subsystems. The main differences that remain are in the cockpit arrangement and the main gearbox. The existing component aircraft uses a tandem cockpit since its fuselage is based on the existing Sikorsky S-67 Blackhawk. The all new aircraft uses a side-by-side cockpit seating arrangement.

The existing component aircraft does not include the inflight adjustable main rotor shaft tilt since it uses an existing gearbox which is difficult to adapt to that arrangement. With the all new aircraft, a new gearbox configuration is used specifically to provide that feature.

ROTOR CONFIGURATION	AIRCRAFT MODIFICATIONS REQUIRED: NEW MAIN GEARBOX	NEW ENGINE INSTALLATION	CONTROL SYSTEM	RPM VARIATION GREATER THAN 30% *
VARIABLE GEOMETRY ROTOR	NO	NO	MINOR	NO
RIGID COUNTERROTATING COAXIAL ROTOR	YES	POSSIBLY	MAJOR	NO
VARIABLE DIAMETER ROTOR	NO	NO	MINOR	NO
JET FLAP ROTOR	YES	YES	MAJOR	NO
VARIABLE TWIST ROTOR	NO	NO	MINOR	NO
SLOWED ROTOR	NO	NO	MINOR	YES

* WILL REQUIRE ACTIVE VIBRATION SUPPRESSION PLUS FURTHER MODIFICATIONS TO AIRCRAFT ELECTRICAL POWER GENERATION SYSTEMS

The only remaining difference of any significance is the difference in main rotor diameters. The existing component aircraft uses the 62' diameter of the Sikorsky S-61 series, while the new aircraft uses a smaller rotor, which is sized to give a higher hovering disc loading. Because they both have the same installed rotor power, the aircraft with the larger rotor can hover at higher gross weights, and carry higher useful loads. The useful loads quoted on page 45 are with the auxiliary propulsion systems removed, but with the large wing installed. These values could be increased by approximately 2200 pounds if the large wing were also removed.

SUMMARY AIRCRAFT COMPARISON

	EXISTING COMPONENT AIRCRAFT	ALL NEW AIRCRAFT
COCKPIT ARRANGEMENT	TANDEM	SIDE - BY - SIDE
NEW DYNAMIC SYSTEM	NO	YES
INFLIGHT VARIABLE MAIN ROTOR SHAFT TILT	NO	YES
ROTOR DIAMETER	62.0 FT	56.8 FT
AVAILABLE USEFUL LOAD*		
SEA LEVEL STD. HOVER	6358 LB	4358 LB
SEA LEVEL 95° HOVER	3358 LB	2469 LB

*LARGE WING ON , AUXILIARY PROPULSION SYSTEM REMOVED

SIKORSKY RECOMMENDED CONCEPT

As a result of the Part II studies, Sikorsky recommended that the Part III RSRA design consist of the existing S61/S67 dynamic components and an all new airframe specifically designed to meet the RSRA requirements. In addition, Sikorsky recommended returning to a single wing design.

The dynamic systems for both aircraft were designed to transmit the same total power, 3700 horsepower. They both used the same engines mounted in approximately the same position on the aircraft. The only advantage of the all new gearbox is that it was designed to provide inflight variable main rotor shaft tilt. However, this becomes a questionable feature with an aircraft such as this which has full adjustable incidence on its wing and horizontal tail. Main rotor shaft angle with respect to the flight path can be varied by trimming the fuselage itself with the horizontal tail. The wing can then be set at its required angle of attack. The only remaining concern would be variations in airframe parasite drag as the body incidence is varied. However, this is a minor factor over the angle ranges considered for RSRA. Because of the higher cost of developing a new dynamic system, Sikorsky recommended that the existing dynamic system be used.

For the airframe on the existing component aircraft, Sikorsky had made every attempt to use the existing S-67 Blackhawk with modifications as required for RSRA. However, these modifications were extensive and it was concluded that little is gained by this approach. An extensive redesign is required for almost all airframe components and the Part II cost estimate showed that there is virtually no difference in airframe cost between the two aircraft designs. Because of this, it seemed reasonable to design the airframe specifically for the RSRA, with the desired side-by-side seating.

At the end of Part I, Sikorsky recommended using two wings for these aircraft -- one large wing for helicopter simulation from 100 to 200 knots, and a second for compound flight investigations to 300 knots. This was done to reduce power required at 300 knots and also to reduce aircraft design gross weight. However, with the more detailed Part II design both aircraft used two TF-34 engines for auxiliary propulsion. These have enough thrust to provide speeds of 300 knots with even the large wing installed, and therefore the small wing cannot be justified on a drag basis. The other advantage of the small wing is that it can reduce the required aircraft design gross weight. This was important when we were trying to use the existing S-67 fuselage, but is not as important when an all new airframe is being designed for the RSRA. Finally, the small wing does add an additional cost to the total program. Because of all these considerations, Sikorsky suggested returning to the single wing design.

APPENDIX I

SUMMARY AND CONCLUSIONS OF PART 1 OF

THE ROTOR SYSTEM RESEARCH AIRCRAFT (RSRA) STUDY

Part 1 of the RSRA study was concerned with determining the overall feasibility of the RSRA technical requirements and concepts envisioned by NASA/Army as specified in the contract work statement for a Rotor Test Vehicle. The contract stated both aircraft requirements and aircraft design goals. Two potential aircraft designs were developed during part I. One was an all new aircraft specifically designed as an RSRA vehicle, and the other was a design which used existing aircraft components wherever feasible to reduce aircraft cost.

The overall feasibility of the technical requirements and concepts and the associated costs were assessed by engineering trade-off studies on a series of all new aircraft. These aircraft were defined by parametric equations within Sikorsky's computerized Helicopter Design Model (HDM) such that each aircraft subsystem was scaled based on the values of it's subsystem design parameters. As the gross weight varied, reflecting a change in inputted requirements, all basic aircraft subsystem parameters were varied to reflect that requirement. With a requirement change, at a particular disc loading, rotor diameter varied with the gross weight, fuselage size and aircraft equivalent parasite area were resized and reestimated respectively, and the mission critical point specified the power installed and sized the engine. Cost equations are an integral part of HDM and were also sensitized to subsystem parameters to assess disireability of the cost of various requirements and features.

THE ALL NEW AIRCRAFT DESIGNS

Three all new aircraft designs were presented to show the effect of designing the RSRA aircraft for minimal capability (below the design requirements and goals), for the design requirements only, and the design goals in addition to design requirements. The minimal capability design, called the basic aircraft, was sized with the features shown in Table AI, and is representative of a 300 knot compound helicopter with the RSRA required wing planform. Few RSRA special features are included.

TABLE AI

BASIC AIRCRAFT

Items Included:

- Propulsion system for 300 knots
- Full wing with full span flaps
- A useful load containing
  - A crew of two
  - 2000 pounds payload
  - Fuel for 15 minutes at 300 knots
- Ultimate load factor 4.5

Items Not Included:

- Variable wing incidence
- Wing instrumentation
- High lift devices, for full wing lift at 100 knots
- Drag devices
- Rotor isolation
- Rotor instrumentation
- Rotor shaft tilt
- Special provisions for gearbox/rotor interface
- Special provisions for overdesign of main gearbox
- Special provisions for overdesign of control system
- Tail rotor instrumentation
- Noise suppression
- Crew escape system
- Full landing gear/braking requirements
- Ballast system
- Third crewman
- Air conditioning, anti-ice
- Drag equal to or greater than NASA/Army minimums
- Auxiliary propulsion instrumentation

The basic aircraft design was used to show the sensitivity of the design to the research payload, dash time, and dash speed. The results are shown in figures A1 and A2. The desireability of keeping the mission payload and time to a minimum for a less costly program was demonstrated. The 2000 lb payload, 15 min dash time point was selected as the requirement for the all new aircraft based on minimum cost for a reasonable payload and testing time.

The list of features included in the aircraft with full required items only and the aircraft with full required and desired items is shown in Table AII. The solution aircraft gross weights of the three aircraft with 2000 lb payload and 15 minutes @ 300 knots were:

Basic aircraft	21447 lbs
Aircraft with Required Items	34741 lbs
Aircraft with Required & Desired Items	36000 lbs

The NASA/Army statement of work asked for a desired payload of 3000 lbs and a desired endurance of 30 minutes at 300 knots. The design gross weight of a vehicle with this capability and full required and desired items was 45,420 lbs.

Table AIII shows the impact of each of the special RSRA requirements on aircraft weight. Each requirement is applied independently to the basic 21447 lb gross weight aircraft. The change in weight empty at constant gross weight and the change in gross weight for constant payload is assessed for each item. The summation of these values for two or more requirements will not produce a true aircraft redefinition since aircraft growth factor varies with gross weight level, but does provide an estimate of relative impact on the aircraft design.

Factors considered in these weight estimates are summarized as follows:

1. Includes a third crewman in the cabin and his Yankee type extraction seat installation.

2. Includes a wing tilting mechanism, structural penalties to both wing and fuselage, and a wing position indicating system.

3. Includes the above wing tilt penalty plus four load transducers, wiring, and wing penalties for modified attachment fittings.

4. Two extendable 7.5 sq. ft. panels mounted on the tail cone are used to provide aerodynamic drag. The weight estimate includes drag surfaces, hinge fittings, actuating mechanism and controls, and structural penalties for cutouts and higher loads.

5. Includes a Yankee upward extraction system for pilot and co-pilot, a main rotor blade severance system and a canopy separation system.

6. Includes standard extraction seats for pilot and co-pilot adapted to downward ejection. Large penalties are assessed for the rerouting of flight controls and heavy structural members directly under the cockpit.

7. Includes direct structure to support and restrain depleted uranium ballast, and structural penalties for increased loads in both the cockpit and tailcone areas of the aircraft.

8. Includes a load transducer, mount modifications, and wiring for each auxiliary propulsion module.

GROSS WEIGHT VS. PAYLOAD & DASH TIME

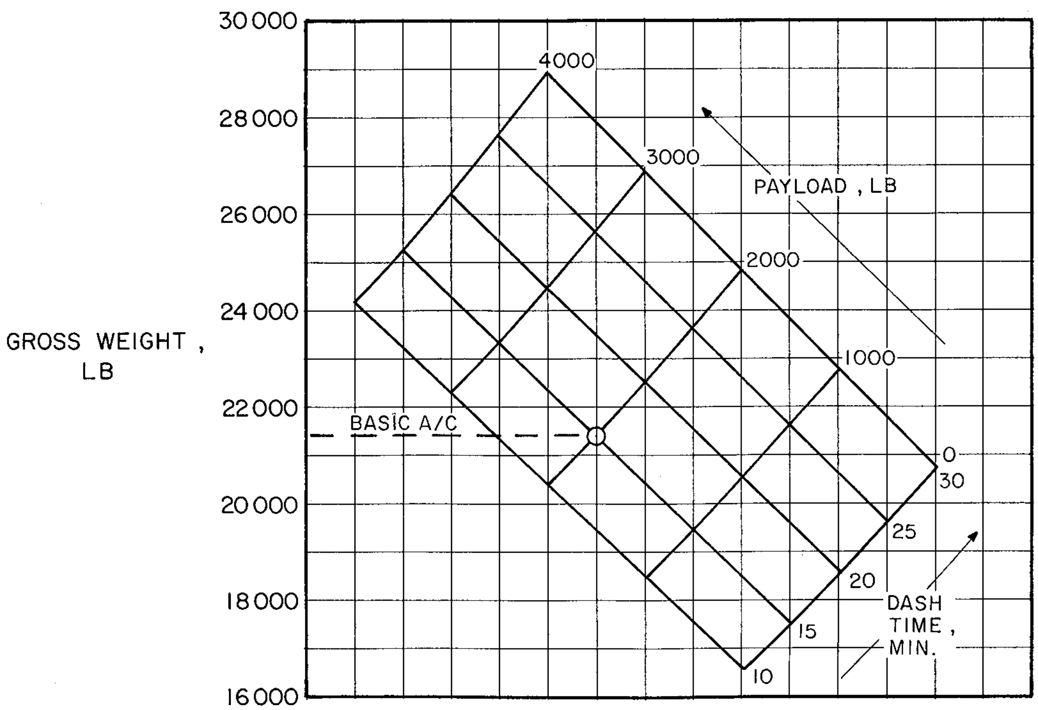


Figure A1

GROSS WEIGHT VS. DASH SPEED & ENDURANCE

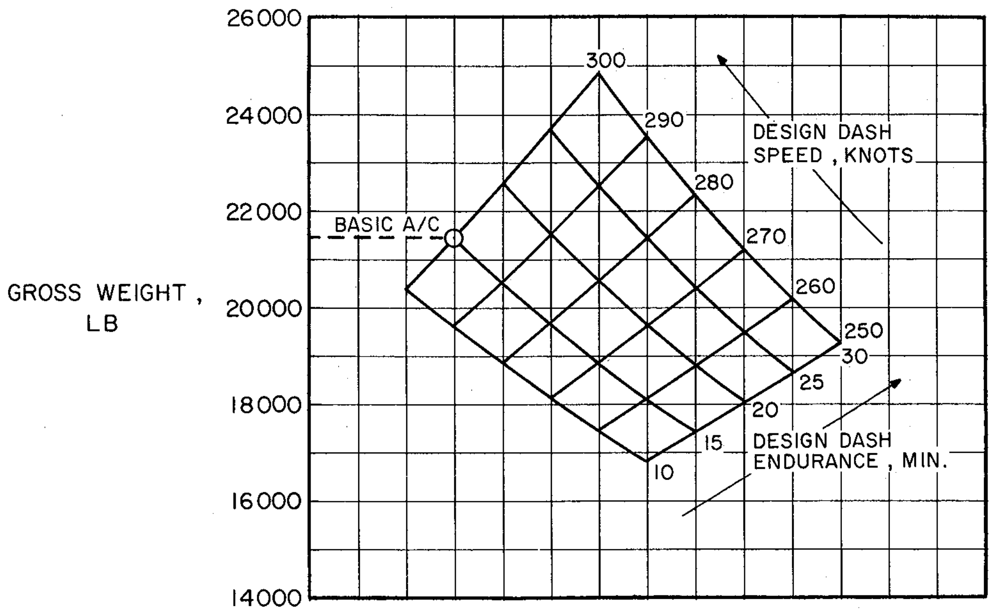


Figure A2

9. Includes load transducers mounted between the tail gearbox and airframe structure, and associated wiring.

10. Rotor load instrumentation is placed at the main gearbox/fuselage interface. The system includes six load transducers, special mounting provisions on the main gearbox and airframe, and wiring.

11. The rotor isolation system is placed at the main gearbox/fuselage interface. The system includes six hydraulic active isolators, special mounting provisions on the main gearbox and airframe, and wiring. The isolators are modified units which act as transducers to measure rotor loads.

12. The rotor instrumentation/isolation system described above can provide main rotor tilt with the addition of isolator extension rods and hydraulic modifications and control.

13. The basic aircraft wing includes simple hinged flaps, supports, and controls. A weight reduction is taken for the conversion to a no flap design.

14. A weight delta is added to provide a rotor shaft/rotor head coupling for each new rotor system. It may therefore be possible to mount new rotor heads without redesigning the rotor shaft.

15. Components of the Part I basic aircraft met the desired noise signature level of 95 db except for the tail rotor. The criterion can be met by constraining the tail rotor to 5 blades and 525 fps tip speed in hover. The resulting compromised design produces a weight penalty. (Part II designs replaced the tail rotor with an anti-torque/yaw control fan.)

16. The basic aircraft landing gear is assumed to be of conventional helicopter design, and is designed to a limit sink speed from hover of 8 fps and a 40 knot conventional landing speed. A penalty is assessed for the criteria of 15 fps limit sink speed from hover and a conventional landing speed of over 100 knots.

17. The basic aircraft is designed to an ultimate load factor of 4.5. A penalty is assessed for increasing the ultimate load factor to 6. Load factor has a pronounced effect on airframe weight; particularly the fuselage and wing.

18. A penalty is assessed for a change from simple hinged wing flaps to leading edge slats and double slotted flaps. The impact on supports and controls is included.

19. A penalty is assessed for 20% increase in main gearbox design power over the basic aircraft design hover power.

TABLE AII

AIRCRAFT WITH FULL REQUIRED ITEMS

Required items:

- 2000 lbs payload
- Fuel for 15 min at 300 knots
- Variable wing incidence
- High lift devices for full lift at 100 kts
- Drag device
- Rotor isolation and instrumentation
- Main rotor shaft tilt
- Special provisions for gearbox/rotor interface
- Special provisions for overdesigning main gearbox
- Special provisions for overdesigning control system
- Tail rotor noise suppression
- Upward escape system
- Full landing gear/braking requirements
- Ballast system
- Full control system requirements
- Air conditioning/anti-icing
- Ultimate load factor of 6.0
- Drag = NASA/Army minimums

Desired Items (Not Included):

- Wing instrumentation
- Anti-torque system instrumentation
- Third crewman
- Aux. propulsion instrumentation

AIRCRAFT WITH FULL REQUIRED AND DESIRED ITEMS

Items included the required item listed above plus:

- Wing instrumentation
- Anti-torque system instrumentation
- Third crewman
- Aux. propulsion instrumentation

TABLE AIII

WEIGHT INCREMENTS FOR ADDITIONAL REQUIREMENTS

TO BASIC AIRCRAFT (GW = 21447 lb.)

	ITEM	Δ WE @ Const. GW	Δ GW @ Const. PL.
1.	Third Crewman	83*	548
2.	Wing Tilt w/o Instr.	346	691
3.	Wing Tilt with Instr.	418	830
4.	Drag Devices	229	443
5.	Upward Eject	146	281
6.	Downward Eject	470	911
7.	Ballast System	226	442
8.	Aux. Prop. Instr.	31	59
9.	Anti-torque Instr.	21	39
10.	Rotor Instr. w/o Isol. w/o Shaft Tilt	179	351
11.	Rotor Instr. with Isol. w/o Shaft Tilt	335	668
12.	Rotor Instr. with Isol. with Shaft Tilt	368	737
13.	Wing with No High Lift Devices	-119	-229
14.	Gearbox/Rotor Interface	44	86
15.	Acoustics	197	393
16.	Landing Gear Full Requirements	698	1383
17.	Load Factor of 6.0	413	826
18.	High Lift Devices For Full Wing Lift At 100 Knots	408	829
19.	Overdesign of MGB	231	457
20.	Overdesign of Control System	310	632
21.	Allowance for Heavier Rotors	1163	2532
22.	Air Cond. & Anti-Ice	101	195

*Add 200 lb to useful load

20. The basic aircraft flight control system is designed for the rotor loads associated with compound helicopter designs. A penalty is assessed for a system capable of taking 120% of the rotor loads of a typical conventional helicopter.

21. The basic aircraft rotor system is designed to meet specified performance requirements, and represents 6.8% of gross weight. The penalty for installation of an alternate rotor system representing 12% of gross weight is assessed. This penalty is severe and it is suggested that the weight for heavier rotors be subtracted from aircraft useful load.

22. The basic aircraft has a simple ventilation system. A penalty is assessed for a full capability ventilating, heating and air cooling system.

At the end of Part I, Sikorsky recommended a modified list of features for the all new aircraft. This included:

Variable wing incidence
Drag device
Rotor instrumentation
Special provisions for gearbox/rotor interface
Special provisions for overdesigning main gearbox
Special provisions for overdesigning control system
Upward crew escape system
Tail rotor noise suppression
Full control system requirements
Air conditioning, anti-icing
Payload = 2000 lb
Tandem seating
Fuel for 15 mins. at 300 knots
Ultimate load factor of 4.5

Not included in the design were:

Wing instrumentation
High lift devices for full lift or 100 knots
Main rotor shaft tilt
Anti-torque system instrumentation
Full landing gear/braking requirements
Ballast system
Third crewman
Drag = NASA/ARMY minimums
Aux. propulsion instrumentation
Rotor isolation system

A three view drawing of the aircraft is shown in figure A-3, page 57.

THE PART I EXISTING COMPONENT AIRCRAFT

The Part I existing component aircraft was configured after surveys were made of flightworthy components which would be readily available to the U.S. Government. Main rotor - tail rotor dynamic systems, wings, fuselages, and engines were reviewed for the applicability to the RSRA program.

The S-61 five bladed rotor system was selected for the rotor system as the best compromise between aircraft size and program cost. A modified existing helicopter fuselage or all new fuselage was required as the structural requirements for escape, wing incidence change, and gearbox installation eliminated fixed wing fuselages from consideration. The survey of wings showed that existing wings in the RSRA size category were designed, in general, with too light a wing loading and a new wing was called for. The engines chosen for the RSRA aircraft were the General Electric T58-16 Turboshaft for main rotor power and the TF 34 turbofan for auxiliary propulsion.

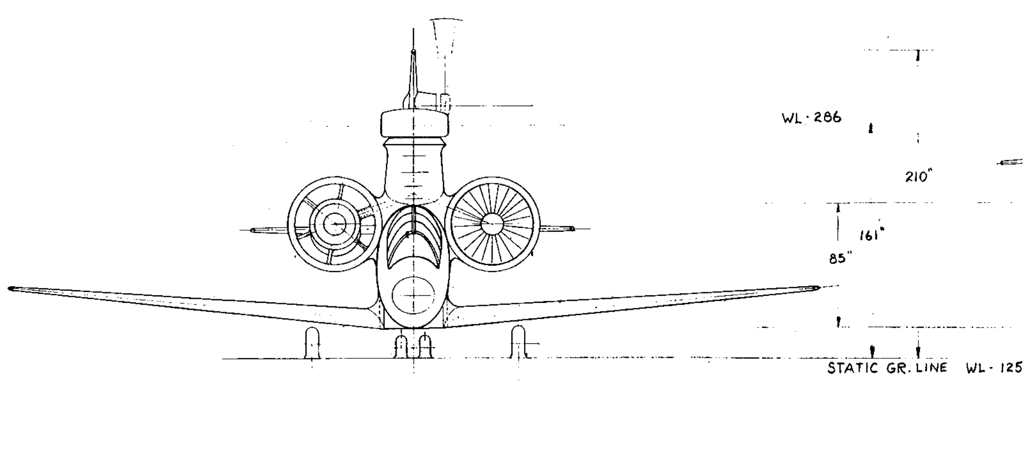
The Sikorsky S-67 airframe was selected for the existing component aircraft as it has been designed for low equivalent parasite area. The new components and modifications are shown below:

New Components

(2) T58-GE-16 engines were substituted for the existing S-67 engines.
(2) TF34-GE-2 Turbofan thrust engines were added
A new 217 sq. ft. wing was substituted for the S-67 wing
A new alighting gear was designed for 24000 lb gross weight, 8 fps helicopter sink speed, and 40 knots conventional landing criteria
The instruments, electrical, avionics, furnishings, and auxiliary gear specified for the basic all-new version were also used for this aircraft.

Basic Modifications

An uprated 3700 design horsepower drive system currently under development for the S-61 type aircraft was incorporated.
The airframe was strengthened to accomodate a 24000 lb design gross weight.
Flight controls were uprated for the increased gross weight and fixed wing controls were integrated.
The five-bladed tail rotor was modified to a six-bladed configuration.
A rudder was added to the vertical tail surface (under development).
The S-67 rotor vibration isolator was removed, and replaced with a full active isolation system.


WL · 286
210"
161"
85"
STATIC GR. LINE WL · 125

A list of the RSRA features which were included in the existing component aircraft at the end of part I is shown below.

Existing Component Aircraft

Included

Variable Wing incidence
Drag device
Rotor instrumentation system
Upward crew escape system
Air conditioning and anti-icing
Ballast system
Auxiliary propulsion thrust instrumentation

Not Included

Wing instrumentation
High lift devices, for full lift at 100 knots
Main rotor shaft tilt
Anti-torque system instrumentation
Tail rotor noise suppression
Full landing gear/braking requirements
Third crewman
Ultimate load factor of 6.0
Drag = Government minimums

A three-view of the Part 1 existing component aircraft is shown as figure A4.

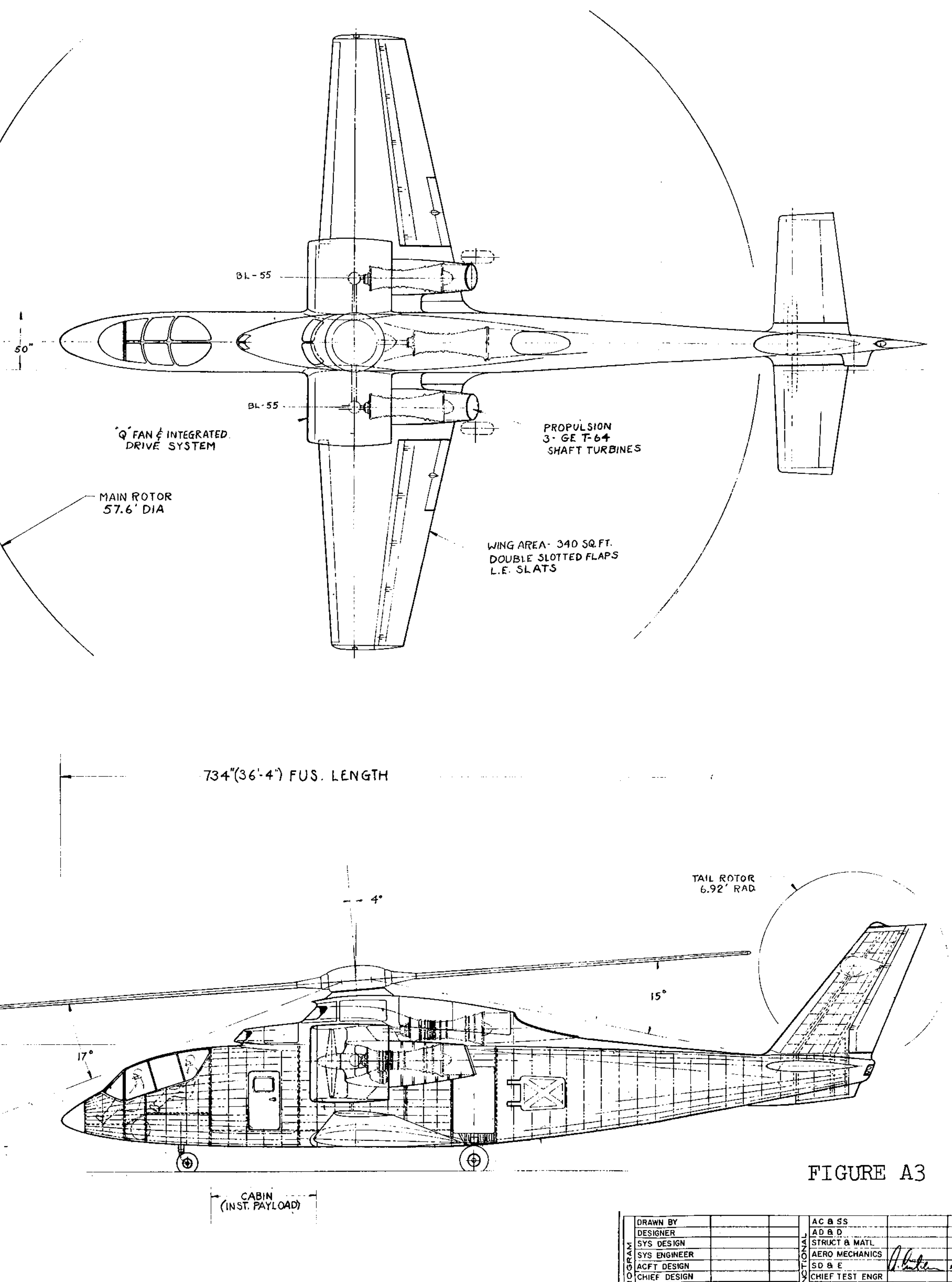


FIGURE A3

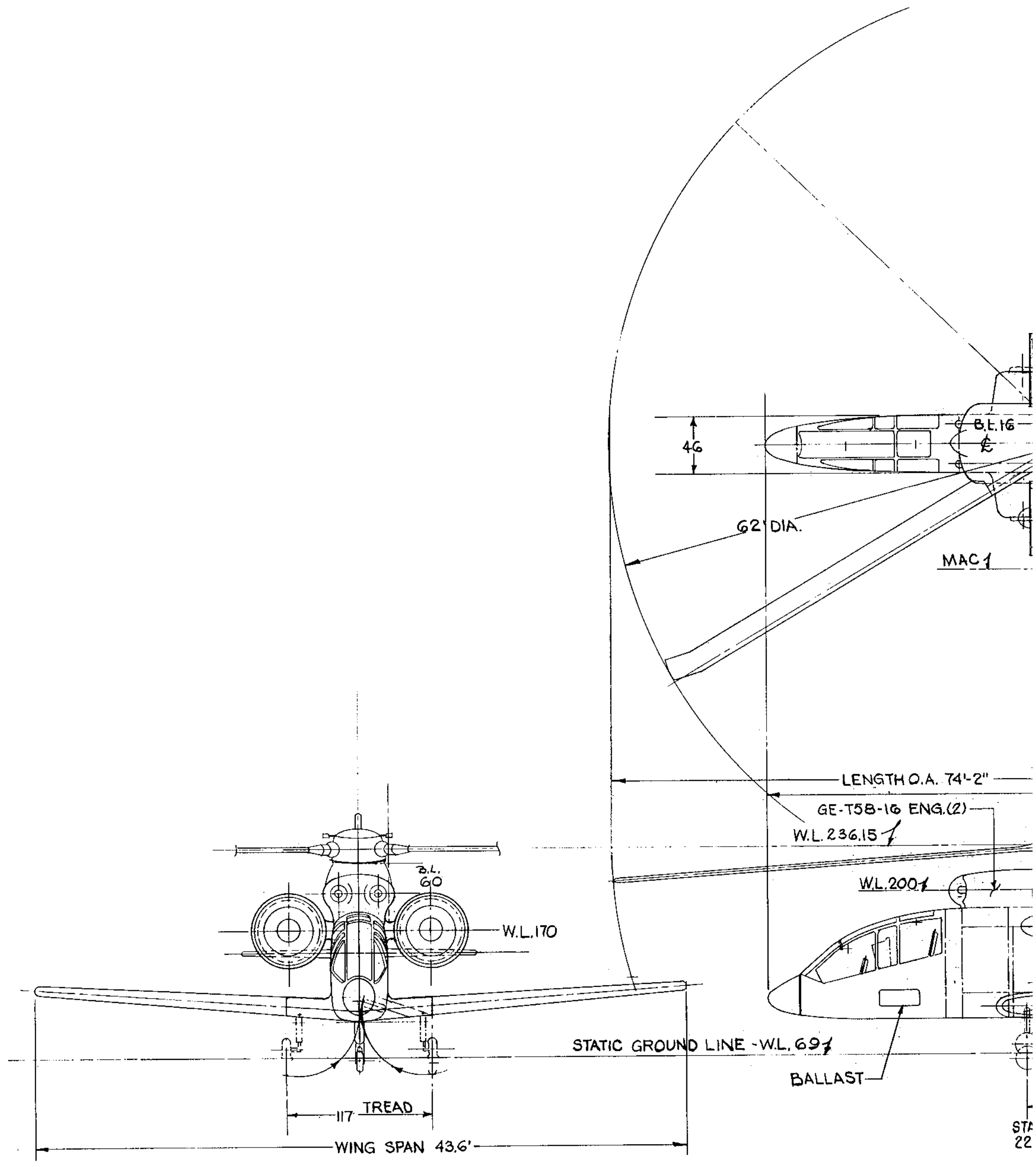
DRAWN BY			AC & SS		
DESIGNER			AD & D		
SYS DESIGN			STRUCT & MATL		
SYS ENGINEER			AERO MECHANICS		
ACFT DESIGN			SD & E		
CHIEF DESIGN			CHIEF TEST ENGR		
TASK MANAGER			CHIEF ENGINEER		
			ENGRG MANAGER		
			MFG/MFG ENGRG		
PROGRAM	NAME	DATE	FUNCTIONAL	NAME	DATE

REPORT NO. | PAGE NO. | FIGURE NO.

ROTOR SYSTEM RESEARCH AIRCRAFT
NEW DESIGN GENERAL ARRANGEMENT
(NASA)

Sikorsky Aircraft | CODE IDENT. NUMBER 78286 | DS-509-3-2
SCALE 1/40 | WBS No. | REV.
SHEET OF

3052 REV. A



B.L.16
46
62' DIA.
MAC
LENGTH O.A. 74'-2"
GE-T58-16 ENG.(2)
W.L. 236.15
W.L. 200
B.L.
60
W.L.170
STATIC GROUND LINE - W.L. 69
BALLAST
117 TREAD
WING SPAN 43.6'
STA
22

PART 1 SUBSYSTEM STUDIES

Specific studies of features of particular importance as to feasibility of their use and effect on the RSRA were conducted during part 1. A synopsis of each is given below:

Crew Extraction System

Main rotor blade severance plus the Yankee escape system for upward extraction of the pilot was selected from several alternate methods because it provided both pilots with a zero to 300 knot escape envelope for the least system weight with readily available components.

Range of Rotor Sizes

A study was conducted to see if a range of disc loadings from 5 psf to 20 psf could be achieved with various rotors on the RSRA aircraft. The study showed that a disc loading of 20 could be tested on the all new aircraft by either a 43.7 ft. diameter rotor by overloading the aircraft to 30,000 lbs gross weight or by a 36.4 ft. diameter rotor at the minimum gross weight of 20,790 lbs. A disc loading of 5.96 could be achieved with a 62 foot rotor at the minimum gross weight of 17,900 lbs. The above rotors are hypothetical and gearboxes are sized to absorb only their hovering power at the sea level standard condition.

Rotor Wing Interference

Wing interference was found to have an increasingly significant effect on rotor lift and flapping as the rotor was unloaded and the wing loaded. This effect diminished with increased forward speed. At "normal" rotor thrust levels the effect of interference appears minimal. Limited experimental studies show small effect of the wing on rotor blade stress levels but this information cannot be generalized.

Expected Accuracy of Rotor Measurements

Main Rotor accuracies for the work statement test condition (Section 4.1.2c) were shown to be the following. These were further analyzed during Part II.

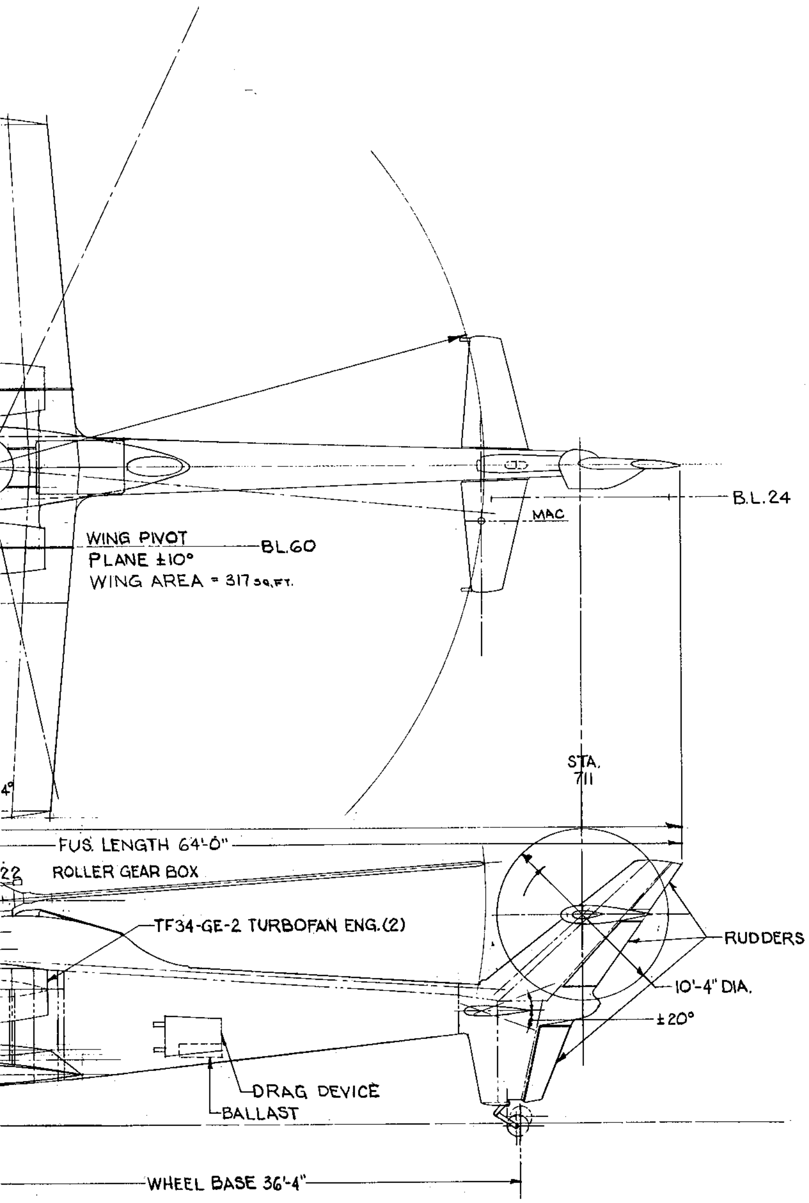


FIGURE A4

PROGRAM	NAME	DATE	FUNCTIONAL	NAME	DATE
DRAWN BY	[illegible]	[illegible]	AC & SS		
DESIGNER			AD & D		
SYS DESIGN			STRUCT & MATL		
SYS ENGINEER			AERO MECHANICS		
ACFT DESIGN			SD & E		
CHIEF DESIGN			CHIEF TEST ENGR		
TASK MANAGER			CHIEF ENGINEER		
			ENGRG MANAGER		
			MFG/MFG ENGRG		
REPORT NO.			PAGE NO.	FIGURE NO. 31	

ROTOR SYSTEM RESEARH AIRCRAFT
EXISTING COMPONENT AIRCRAFT
PART-I

Sikorsky Aircraft	CODE IDENT. NUMBER 78286	DS-507-8-7	
		SCALE 1/40	WBS No.	REV. A
		SHEET OF	

SD92 REV. A

TEST POINT STUDY (150 kts)

Main Rotor Hub Forces (Shaft Axis)	Test Condition	Accuracy of Measurement (1σ)
Longitudinal	-1380 lbs	±185 lbs
Lateral	0 lbs	±171 lbs
Thrust	18000 lbs	±105 lbs
Rolling Moment	0 ft-lbs	±1000 ft-lbs
Pitching Moment	6750 ft-lbs	±1080 ft-lbs
Torque	60,000 ft-lbs	± 416 ft-lbs

Expected Accuracies of the Auxiliary Propulsion, Wing and Anti-Torque Device

Accuracies on the order of 2% were projected for auxiliary propulsion and anti-torque system thrust and torque. Wing load accuracy was projected to be within 4%, however, part II analysis has shown wing accuracies to within 2%.

Performance Analysis

A engineering study conducted of Sikorsky's rotor prediction method, similar but more sophisticated than that of NASA CR-114, showed generally more conservative rotor performance.

Wing Incidence Requirements

The table below shows the maximum and minimum wing incidence required to provide rotor unloading (maximums) and full gross weight autorotation (minimums) at the most critical speed points. The range required is the included angle between the maximum and the minimum.

WING INCIDENCE RANGE REQUIRED
WING INCIDENCE WITH RESPECT TO THE FUSELAGE

	Variable Shaft Angle -10° to $+10^\circ$	Fixed Shaft Angle 0°
Maximum Wing Incidence	13°	23°
Minimum Wing Incidence	-9°	-19°
Range Required	22°	42°

Alternate Anti-Torque System

A fan-in-fin was designed for the basic aircraft as an alternative to the tail rotor. The study results on the basic aircraft are shown below. Due to the problems at higher cruise speeds with a tail rotor, the fan-in-fin appears to be a desireable item for compound testing.

	Tail Rotor	Fan-in-Fin
Gross Weight, lb	21447	21634
Weight Empty, lb	16232	16402
Δf, ft^2	--	-.8
% Main GB input power	10	25
Radius, ft	5.75	2.34
Steady hover disc loading psf	20	75

Rotor Vibration Suppression

A vibration suppression device was configured which will be applicable to all foreseeable rotor systems to be tested on the RSRA aircraft. This system would require development.

Feasibility of Building Two Different Aircraft

Feasibility of building two different aircraft, one with full requirements and one minimum size resulted in the smaller aircraft having a very short test time at 300 knots (7 min.). Building two similar aircraft with different size wings, one for high speed and one for rotor simulation between 100 and 200 knots, allowed lower design gross weight without either aircraft impacting on the RSRA requirements.

Control System

A feedback control system was designed to provide accurate control of the rotor thrust and moments during inflight data gathering. The control system was shown to have sufficient gain and phase margin to make control of the rotor appear feasible given sensors of sufficient accuracy.

Government Modifications to Aircraft Requirements

After Sikorsky's oral presentation of the part one results at Langley on 28 February, 1972, the government made certain changes in the aircraft technical requirements to be used during Parts II and III of the study. These are summarized as follows:

1. The method of calculating rotor performance was modified slightly to allow the contractor to use methods other than CR-114.

2. The parasite drag minimums for the main rotor hub and mast were reduced.

3. The desired mission payload and endurance was dropped from 3000 pounds/30 minutes to a required mission of 2000 pounds/15 minutes. Fuel tankage capacity remained at that required for 30 minutes of flight at 300 knots.

4. The helicopter flight simulation boundary was modified to delete the requirement for zero rotor lift at speeds from 100 to 200 knots. The new requirement was 150 to 200 knots.

5. Requirements on the anti-torque device were modified so that the requirements of MIL Spec 8501A will be met.

6. The wing requirements were modified to allow the use of two wings; a large wing for the helicopter simulation from 100 to 200 knots, and a smaller wing for the 300 knot dash speed requirement.

7. The requirement on braking capacity was reduced slightly.

8. Landing gear design limit sinking speed in hover was reduced from 15 feet per second to 8 feet per second. For fixed wing landings, the requirement was reduced from 12 feet per second to 8 feet per second.

Other government suggestions included reducing the capability of the rotor isolation system and the ballast system. An anti-torque fan was preferred to a tail rotor, mainly because of the 300 knot cruise speed condition and the problems that the tail rotor might have operating at that point. A side by side cockpit seating arrangement was preferred, even though Sikorsky showed that this results in an aircraft weight penalty of approximately 600 pounds over a tandem design. Finally, an auxiliary propulsion system completely separate from the rotor propulsion system was prefered to the convertible propulsion scheme shown on Sikorsky's all new aircraft design.