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A Feasibility Study of Orbiter Flight Control Experiments

W. H. Geissler

McDonnell Douglas Technical Services Company, Inc.
Houston Astronautics Division
Houston, TX 77058

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Langley Research Center
Hampton, Virginia 23665

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FOREWORD

This report fulfills the final reporting requirements for "A Feasibility Study of Orbiter Flight Control Experiments" performed under the National Aeronautics and Space Administration Contract NAS1-15141. The study was conducted under the direction of M. T. Moul of the Flight Dynamics and Control Division, Langley Research Center.

W. H. Geissler of McDonnell Douglas Technical Services Co., Inc. (MDTSCO), Houston Astronautics Division, was the technical and study manager. D. C. Blanchard, O. R. DeVall, M. E. Fowler, R. E. Speir, and H. W. Stegall of the MDTSCO control group performed some of the control experiment feasibility studies. R. K. Hamilton and R. L. Walsh of the MDTSCO performance group performed the feasibility studies for the aero data extraction experiments.

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ACRONYMS AND ABBREVIATIONS

ACIP	AERODYNAMIC COEFFICIENT IDENTIFICATION PACKAGE
ADE	AERO DATA EXTRACTION
ADS	AIR DATA SYSTEM
AFFDL	AIR FORCE FLIGHT DYNAMICS LABORATORY
AGC	ADAPTIVE GAIN CHANGER
ALT	APPROACH AND LANDING TEST
APU	AUXILIARY POWER UNIT
ARC	AMES RESEARCH CENTER
CAS	CONTROL AUGMENTATION SYSTEM
CDC	CONTROL DATA CORPORATION
CG	CENTER OF GRAVITY
COAS	CREW OPTICAL ALIGHMENT SIGHT
CRT	CATHODE RAY TUBE
DAP	DATA ANALYSIS PACKAGE
DFI	DEVELOPMENTAL FLIGHT INSTRUMENTATION
DFRC	DRYDEN FLIGHT RESEARCH CENTER
DRIMS	DELTA REDUNDANT INERTIAL MEASUREMENT SYSTEM
EAD	ENGINEERING ANALYSIS DIVISION
FCS	FLIGHT CONTROL SYSTEM
FDA	FAULT DETECTION AND ANNUNCIATION
FDAM	FREE DRIFT ATTITUDE MODE
FSSR	FUNCTIONAL SUBSYSTEM SOFTWARE REQUIREMENTS
FTR	FLIGHT TEST REQUIREMENTS
GDQ	GAIN IN PITCH FORWARD LOOP
GG	GRAVITY GRADIENT
GN&C	GUIDANCE, NAVIGATION AND CONTROL
GPC	GENERAL PURPOSE COMPUTER
GSFC	GODDARD SPACE FLIGHT CENTER
HI	HONEYWELL, INC.
HZ	HERTZ
ICD	INTERFACE CONTROL DOCUMENT
IMU	INERTIAL MEASUREMENT UNIT
IUS	INERTIAL UPPER STAGE
JSC	JOHNSON SPACE CENTER
KDF	KALMAN DIAGNOSTIC FILTER
LARC	LANGLEY RESEARCH CENTER
LOS	LINE OF SIGHT
MCAIR	MCDONNELL AIRCRAFT COMPANY
MDAC-HB	MCDONNELL DOUGLAS ASTRONAUTICS CO. - HUNTINGTON BEACH
MDM	MULTIPLEXER/DEMULTIPLEXER

ACRONYMS AND ABBREVIATIONS (Continued)

MDTSCO	MCDONNELL DOUGLAS TECHNICAL SERVICES CO., INC. - HOUSTON ASTRONAUTICS DIVISION
MET	METEOROLOGICAL DATA
MLSIP	MAXIMUM LIKELIHOOD SYSTEM IDENTIFICATION PROGRAM
MMLE	MODIFIED MAXIMUM LIKELIHOOD ESTIMATOR
MSFC	MARSHALL SPACE FLIGHT CENTER
N	NEWTON
NASA	NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
NAV	NAVIGATION
NAVDAD	NAVIGATIONALLY DERIVED AIR DATA
OEX	ORBITER EXPERIMENT
OFT	ORBITAL FLIGHT TEST
PIO	PILOT INDUCED OSCILLATION
P/L	PAYLOAD
POP	PERPENDICULAR TO ORBIT PLANE
PTI	PROGRAM TEST INPUT
QI	QUASI-INERTIAL
RCS	REACTION CONTROL SYSTEM
RHC	ROTATIONAL HAND CONTROLLER
RI	ROCKWELL INTERNATIONAL
RM	REDUNDANCY MANAGEMENT
ROM	ROUGH ORDER OF MAGNITUDE
SAIL	SHUTTLE AVIONICS INTEGRATION LABORATORY
SEADS	SHUTTLE ENTRY AID DATA SYSTEM
SFCS	SURVIVABLE FLIGHT CONTROL SYSTEM
SF	SCALE FACTORS
SLRT	SEQUENTIAL LIKELIHOOD RATIO TEST
SMS	SHUTTLE MISSION SIMULATOR
SM	SYSTEMS MANAGEMENT
SOP	SUBSYSTEM OPERATING PROGRAM
SOW	STATEMENT OF WORK
SSFS	SPACE SHUTTLE FUNCTIONAL SIMULATION
SSUS	SPINNING SOLID UPPER STAGE
SUMS	SHUTTLE UPPER ATMOSPHERE MASS SPECTROMETER
S/W	SOFTWARE
TAEM	TERMINAL AREA ENERGY MANAGEMENT
TRS	TELEOPERATOR RETRIEVAL SYSTEM

SYMBOLS

A	AMPLITUDE OF DITHER SIGNAL
α	ANGLE OF ATTACK
α_N	ANGLE OF ATTACK FROM NAV
$\bar{c}$	MEAN AERODYNAMIC CHORD
$C_{\ell_{\delta_A}}$	COEFFICIENT OF ROLLING MOMENT DUE TO AILERONS
$C_{m_{\delta_E}}$	COEFFICIENT OF PITCHING MOMENT DUE TO ELEVATORS
$C_{n_{\delta_R}}$	COEFFICIENT OF YAWING MOMENT DUE TO RUDDER
δ_A	AILERON DEFLECTION
δ_E	ELEVATOR DEFLECTION
δ_{SB}	SPEEDBRAKE DEFLECTION
h	ELEVON HYSTERESIS
Iyy	PITCHING MOMENT OF INERTIA
K	CONTROL SYSTEM GAIN
ℓ_b	VEHICLE BODY LENGTH
L_{δ_A}	ROLLING MOMENT DUE TO AILERONS
M	MACH NUMBER
M_N	MACH NUMBER FROM NAV
M_E	PITCHING MOMENT DUE TO ELEVATORS
n_y, N_Y	LATERAL ACCELERATION
n_z, N_Z	NORMAL ACCELERATION
P, p	ROLL RATE
Q, q	PITCH RATE
$\dot{q}$	PITCH ACCELERATION
$\bar{q}$	DYNAMIC PRESSURE
$\bar{q}_N$	DYNAMIC PRESSURE FROM NAV
R, r	YAW RATE
S	REFERENCE AREA
σ	STANDARD DEVIATION
τ	INTEGRATION INTERVAL
ω_o	DITHER SIGNAL FREQUENCY OR VEHICLE ORBITAL RATE
θ	PITCH EULER ANGLE
ψ	YAW EULER ANGLE
ϕ	ROLL EULER ANGLE
U_{AS}	AIRSPEED

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1.0 SUMMARY

This final report summarizes the results of a feasibility study of Orbiter flight control experiments performed for the Langley Research Center under Contract NAS1-15141. Feasibility studies were performed on a group of 14 experiments selected from a candidate list of 35 submitted to the study contractor by the flight control community. The selected group represented a wide variety of experiments which fall in the general categories called for by the study statement of work. Lack of data or the fact that an Orbiter flight was not required to prove an experiment concept were the major reasons for terminating further study on the other suggested experiments.

Concepts and requirements were developed for the 14 selected experiments and they were ranked on a basis of technical value, feasibility, and cost. It was concluded that all the selected experiments can be considered as potential candidates for the Orbiter Experiment program, which is being formulated for the Orbiter Flight Tests and subsequent operational flights, regardless of the relative ranking established during the study. None of the selected experiments has significant safety implications and the cost of most was estimated to be less than $200K.

2.0 INTRODUCTION

In order to utilize the research potential of the Space Shuttle, a program of Orbiter Experiments (OEX) is being formulated for the Orbiter Flight Tests (OFT). Proposed experiments are directed toward the enhancement of the operational efficiency of the Shuttle or the research and technology base for future spacecraft design. This report presents the results of a study directed to the definition of flight experiments within the flight control discipline which comply with the above guidelines. The fundamental objective of the study was to identify a number of important flight controls experiments which should be conducted in the OEX Program. The contract statement of work (SOW) specified the study objectives to be:

(1) Compile a list of candidate flight control experiments including inputs from the government.

(2) Develop experiment concepts and requirements.

(3) Determine technical value and feasibility of each experiment.

(4) Rank the experiments by criteria developed jointly by the contractor and the Government.

The candidate experiments were to be directed to any of the following Orbiter mission flight regimes: on-orbit, early entry, late entry, terminal area, approach, and landing. Experiments of the following types were to be considered:

(1) Passive, postflight analysis experiments utilizing flight test data. Examples are analyses of aerodynamic parameters, flying qualities, and control system performance.

(2) Experiments which interact with the Orbiter system. Examples are advanced flight control systems that offer improvements in reliability, capability, performance, complexity, cost, size, or power.

Results of the study are presented in the following sections. The study discussion (Section 3.0) describes the our approach employed in obtaining suggestions from the flight control community, the suggested candidate experiments, and the experiments selected for further study. A brief synopsis of each experiment selected for further study is presented in Section 4.0. The detailed discussion of these experiments is contained in Appendix A; experiments not accepted for further study are described in Appendix B. Section 5.0 describes the ranking criteria and presents the experiment rank based on this criteria. Conclusions are summarized in Section 6.0.

As an aid to the reader, original units of measure have been converted to the equivalent value in the International System of Units (SI). The SI units are written first, and the original units are written parenthetically, thereafter.

3.0 STUDY DISCUSSION

A list of candidate experiments within Flight Controls and Flight Dynamics disciplines were solicited from within the Government, the contracting company, and other aerospace organizations. Thirteen responses were received which contained single and multi-experiment suggestions. Suggested experiments were categorized, and a cursory evaluation was made of each one. Experiments were reviewed with the Langley Research Center (LARC) technical monitor and 14 were selected for further study.

3.1 Preliminary Selection Criteria

After receiving the experiment suggestions from the various organizations, they were classified into three general categories:

- Flight Control
- Aero Data Extraction
- Miscellaneous

The latter category included suggestions dealing with redundancy management, displays, payload reference alignment, and others. While these experiments did not, in the strict sense, fall in the flight control category addressed by the study, they have some flight control association and were given due consideration.

Selection of a suggested experiment for more detailed study was coordinated with the LARC technical monitor and was based on fulfilling the following requirements:

(1) The suggestion must fall into the general category of experiments asked for in the SOW.

(2) The experiment must be exercised during an Orbiter flight to prove its feasibility or it must be evaluated post flight from actual flight data.

(3) Adequate data and/or concept must be available to the contractor (MDTSCO) to do an evaluation.

No further study of a suggested experiment was pursued if all of the above requirements weren't fulfilled.

3.2 Candidate Flight Control Experiments

A list of the suggested experiments is presented in Table 1 along with the suggestor(s), a further study indication, and the reason an experiment wasn't selected for further study. This latter reason is designated by the number requirement previously described that wasn't fulfilled. A brief narrative describing each experiment not selected for further study is presented in Appendix B. The 14 experiments selected for further study are described in Section 4.0 and Appendix A.

TABLE 1

EXPERIMENT SUMMARY AND STATUS

EXPERIMENT	SUGGESTOR	ACCEPTED FOR FURTHER STUDY	REASON DROPPED
1. MODIFIED ROTATIONAL HAND CONTROLLER	MCAIR	YES	-
2. ADAPTIVE GAIN CHANGER	MCAIR	YES	-
3. OPTIMAL CONTROL BLENDING TO EXPAND ENVELOPE	BOEING	MP	3
4. BLENDING USE OF AILERONS AND RUDDER FOR IMPROVED LAT/DIR CONTROL	MDTSCO/BOEING LEC/SYSTEMS CONTROL	NO	2
5. DECREASE FLIGHT CONTROL SAMPLE RATE	GD/MDTSCO	YES	-
6. BENDING MODE SUPPRESSION	SYSTEMS CONTROL	NO	3
7. FREE DRIFT ATTITUDE MODES	RI/MDTSCO	YES	-
8. ORBITER FLYING QUALITIES AND FCS PERFORMANCE	ARC/LARC	YES	-
9. CONTROL OF LARGE SPACE STRUCTURES	HI/LEC	NO	1
10. CRITERIA FOR FCS RCS/AERO WORK LOAD	MDTSCO	NO	3
11. SHUTTLE POINTING WITH CMG'S	HI	NO	2
12. RM & CMG CONTROL BLENDING	HI	NO	3
13. CLOSED LOOP ARM CONTROL	HI	NO	3
14. COMPARISON OF AERO DATA EXTRACTION (ADE) TECHNIQUES	MDTSCO	YES	-
15. EVALUATION OF ADE MANEUVER FORMATS	MDTSCO	YES	-
16. INSTRUMENTATION QUALITY IMPACT ON ADE	MDTSCO	YES	-
17. AERO DATA EXTRACTION	ARC/DFRC/MCAIR/MDTSCO	YES	-

TABLE 1 (CONTINUED)

EXPERIMENT SUMMARY AND STATUS

EXPERIMENT	SUGGESTOR	ACCEPTED FOR FURTHER STUDY	REASON DROPPED
18. INVESTIGATION OF HYPERSONIC CHARACTERISTICS DUE TO VISCOUS INTERACTION AND REAL GAS EFFECTS	MDTSCO/RI-SD	YES	-
19. INFLUENCE OF JET FIRINGS ON ORBITER FLIGHT CHARACTERISTICS	MDTSCO/HI/DFRC	YES	-
20. STRAKE VORTEX VISUALIZATION	DFRC	NO	1
21. ESTIMATION OF ORBITER INERTIAL PROPERTIES WITH RM DEPLOYED	SYSTEMS CONTROLS	NO	1
22. SYNCHRONIZED MID-VALUE SELECT AVERAGING	MCAIR/SYSTEMS CONTROL	NO	2
23. VOTING WITH LRU NOT IN COMMON LOCATION	MCAIR	NO	3
24. RCS/FDI USING ONBOARD VEHICLE STATE ESTIMATES AND/OR PHASE PLANE SWITCHING LINES	LEC	NO	2
25. ANALYTICAL REDUNDANCY	HI	YES	-
26. REAL-TIME TRAJECTORY GENERATION	MCAIR	NO	3
27. FLAT SURFACE DISPLAY TECHNOLOGY	SYSTEMS CONTROL	NO	2
28. ADVANCE DISPLAY DESIGN	HI	NO	2
29. SYSTEM MONITOR DISPLAY	HI	NO	3
30. HELMET SIGHT, DISPLAY AND POINTING IN ZERO G	HI	NO	2
31. ESTIMATION TECHNIQUES FOR DATA SMOOTHING	HI	NO	2
32. TERMINAL AREA SENSING	HI	NO	2

TABLE 1 (CONTINUED)

EXPERIMENT SUMMARY AND STATUS

EXPERIMENT	SUGGESTOR	ACCEPTED FOR FURTHER STUDY	REASON DROPPED
33. ALIGNMENT TRANSFER FROM SHUTTLE TO PL	MSFC/MDAC-HB	YES	-
34. ALIGNMENT VARIATION - REFERENCE TO CARGO BAY	RI-SD	YES	-
35. WIND ESTIMATION	SYSTEMS CONTROL/HI	NO	2

4.0 SUMMARY OF SELECTED EXPERIMENT STUDY RESULTS

A list of the experiments selected for further study is presented in Table 2. This group represents a wide variety of types which fall in the general experiment categories called for by the study SOW. The first five (A-E) pertain directly to vehicle attitude control and contribute the following:

- The "Modified RHC" and "Adaptive Gain Changer" experiments potentially provide aid for Orbiter entry FCS problem areas.

TABLE 2

EXPERIMENTS SELECTED FOR FURTHER STUDY

A. MODIFIED ROTATIONAL HAND CONTROLLER (RHC)

B. ADAPTIVE GAIN CHANGER (AGC)

C. DECREASED FLIGHT CONTROL SAMPLE RATE

D. FREE DRIFT ATTITUDE MODES (FDAM'S)

E. ORBITER FLYING QUALITIES AND FCS PERFORMANCE

F. COMPARISON OF AERO DATA EXTRACTION (ADE) TECHNIQUES

G. EVALUATION OF ADE MANEUVER FORMATS

H. INSTRUMENTATION QUALITY IMPACT ON ADE

I. AERO DATA EXTRACTION

J. INVESTIGATION OF HYPERSONIC CHARACTERISTICS DUE TO VISCOUS INTERACTION AND REAL GAS EFFECTS

K. INFLUENCE OF JET FIRINGS ON ORBITER FLIGHT CHARACTERISTICS

L. ANALYTICAL REDUNDANCY

M. ALIGNMENT TRANSFER FROM SHUTTLE TO PAYLOAD

N. ALIGNMENT VARIATION - REFERENCE TO CARGO BAY

- Decreasing the flight control sample rate is basically a demonstration for future digital FCS designers.
- Successful demonstration of the FDAM's will add to the Shuttle orbital operation capabilities.
- The "Orbiter Flying Qualities and FCS Performance" experiment combines in one document all of the Orbiter control characteristics for historical and future reference.

The next six experiments listed in Table 2 (F-K) pertain to aero data extraction (ADE). They represent experiments which are a natural follow-on to the present mainline effort. Figure 1 presents the general relationship

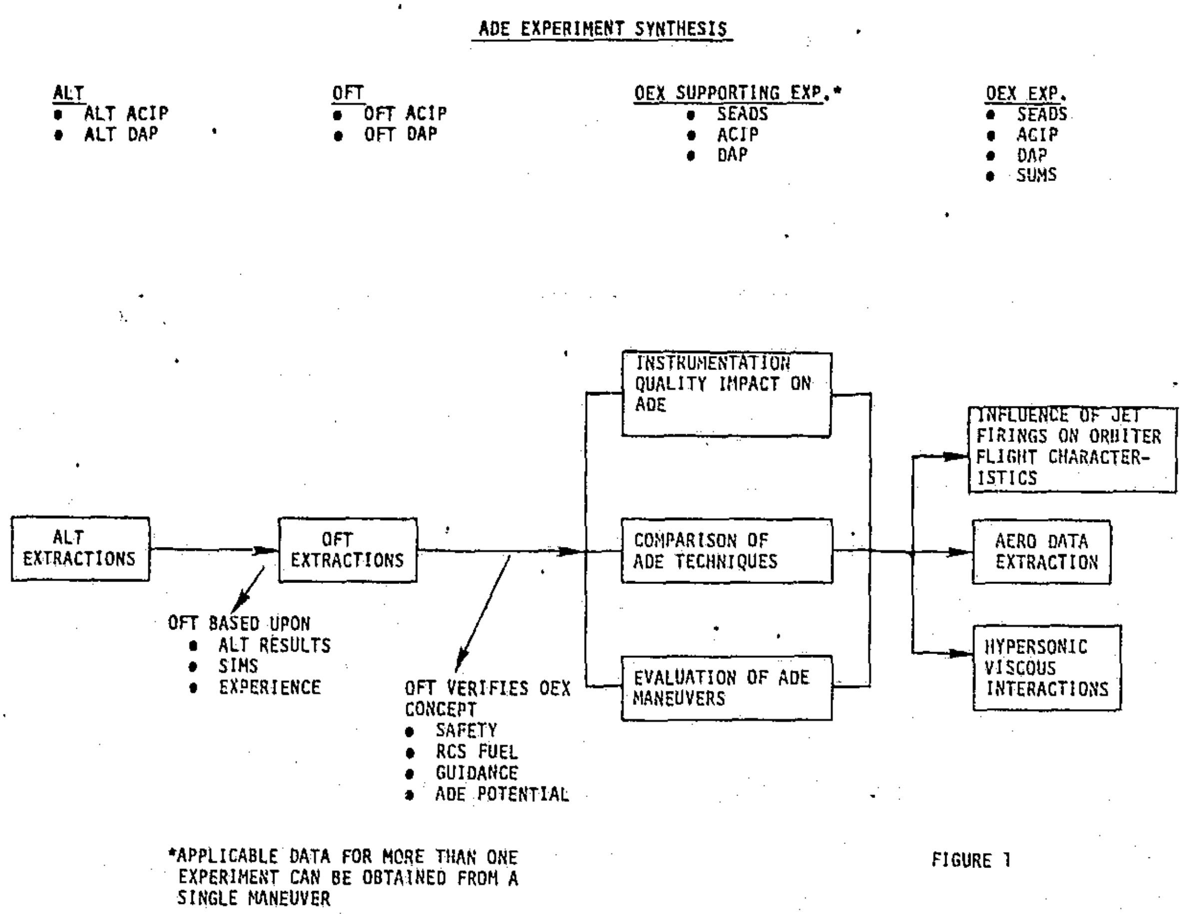

FIGURE 1

to the mainline effort. The OEX ADE experiments are separated into two groups, one group being support for the others. Results obtained from the supporting experiments will enhance the achievement of good results from the others. The systems listed above the experiment blocks are required to obtain good results throughout the entry flight regime.

The last three experiments listed in Table 2 (L-N) are in the "miscellaneous" category discussed in Section 3.1. The analytical redundancy scheme of the proposed experiment offers potential to significantly reduce the number of redundant sensors required in the flight control systems of fly-by-wire aircraft. Results of the "Alignment Transfer from Shuttle to Payload" experiment would be of great interest to the Inertial Upper Stage (IUS) program or any other payload that needs its inertial reference accurately aligned. Measurement of the structural deformation between the Orbiter Nav base and various mount points in the payload bay (last experiment in Table 2) would be of interest to any user that relies on the Orbiter for pointing the payload with an accuracy of less than two degrees. In addition it would help establish feasibility of the previous experiment.

In order to maintain the main body of this report as concise as possible, only a summary of the study results for these experiments is presented in this Section. A more detailed description is presented in Appendix A.

Feasibility of each experiment was established or substantiated by our studies and reviews (Appendix A). Experiment objectives, requirements, mission impact and costs are included in the summaries at the end of this section. In order to establish a cost commonality between experiments, the following was assumed:

- Engineering support - $30/hour

- Software development - $300/word (Based on present rough order of magnitude (ROM) IBM costs for Orbiter GPC)

- Shuttle Avionics Integration Laboratory (SAIL) and Shuttle Mission Simulator (SMS) time not charged to experiments (These facilities are assumed to be maintained by institutional support or the Shuttle Program. Present plans have the SAIL maintained through 1984; the SMS beyond this.).

- It was assumed that Government computational facilities with no direct charge to the NASA would be used for analysis and data reduction.

Unique cost assumptions relating to certain experiments are discussed in Appendix A.

A. ROTATIONAL HAND CONTROLLER (RHC) EXPERIMENT

SUGGESTOR: McDonnell Aircraft Company (MCAIR)

TYPE: Flight Control

OBJECTIVES: Employ a RHC in the Orbiter cockpit which provides a linear rate command vs. stick force gradient (roll and pitch) to 1) improve Orbiter handling qualities, 2) reduce pilot-induced oscillation (PIO) tendency, and 3) provide more desirable on-orbit RHC spring forces.

REQUIREMENTS:

- Modified RHC for Simulator Studies
- Pre-flight evaluation on Shuttle Mission Simulator
- Flight-qualified modified RHC .
- Modification to RHC System Operating Program (SOP) Software
- Installation of modified RHC in Orbiter

MISSION IMPACT: Small

COST $90K - 120K

B. ADAPTIVE GAIN CHANGER EXPERIMENT

SUGGESTOR: MCAIR

TYPE: Flight Control

OBJECTIVES: Demonstrate the ability to vary forward loop FCS gains using in-flight measurement of aerosurface effectiveness.

REQUIREMENTS:

- 1100 words of software in each of four primary GPCs.
- One switch dedicated for entry to engage/disengage function.
- 4 to 6 man-months of off-line analysis.
- Limited verification at systems integration levels desirable.

MISSION IMPACT:

- Crew will engage at Mach 2.5 (80 KFT)
- Crew will disengage before touchdown.
- Small elevator dither signal required.
- Crew alertness to non-standard vehicle motion.

COST:

- $360K

C. DECREASED FLIGHT CONTROL SAMPLE RATE EXPERIMENT

SUGGESTIONS: McDonnell Douglas Technical Services Company (MDTSCO) and General Dynamics

TYPE: Flight Control

OBJECTIVES: Show the feasibility and desirability of designing digital control systems with sample-size as one of the design parameters. Demonstrate in-flight operation of the control system at a selected reduced FCS sample rate.

REQUIREMENTS:

- Off-line analysis and simulation studies to design filters and gain schedules.
- Onboard software to duplicate those modules of the FCS that depend upon sample period or require initialization.

MISSION IMPACT: Small

COST: $185K

D. FREE DRIFT ATTITUDE MODE (FDAM) EXPERIMENT

SUGGESTOR: Rockwell International/Space Division

TYPE: Flight Control

OBJECTIVES: To provide confidence in the use of FDAM's to provide extended time periods without RCS firings and to provide better estimates of uncertain parameters such as principal axis location variations, crew motion disturbances, earth magnetic disturbance torques, etc., which are required to achieve desired performance levels in an operational system implementation.

REQUIREMENTS:

- Off-line analysis to obtain disturbance torque sensitivities

MISSION IMPACT:

- Ground real-time support during mission.
- Twenty to 90 minutes of Orbiter drift during several orbits.
- Restrained and prescribed crew motion
- Scheduled venting

COST: $60K

E. ORBITER FLYING QUALITIES AND FLIGHT CONTROL SYSTEM PERFORMANCE EXPERIMENT

SUGGESTORS: NASA/LARC and NASA/ARC

TYPE: Flight Control

OBJECTIVES: Orbiter flying qualities and flight control system performance throughout the entry-to-touchdown flight regime will be obtained. Astronaut subjective data, and Orbiter transfer functions desired from postflight analysis of flight data will be combined in one flying qualities document.

REQUIREMENTS:
- Shuttle Entry Air Data System (SEADS)
- Aerodynamic Coefficient Identification Package (ACIP)
- Programmed Test Inputs (PTI's) for inflight maneuvers
- Flight measured to ground analysis data processing (DAP)
- Transfer function determination program

MISSION IMPACT:
- Reaction Control System (RCS) Propellant
- Auxiliary Power Unit (APU) Fuel
- Normal and established crew activity
- Safety impact assessment of downmode option

COST: $175,000

F. COMPARISION OF AERO DATA EXTRACTION TECHNIQUES EXPERIMENT

SUGGESTORS: MDTSCO

TYPE: Aero Data Extraction

OBJECTIVES: Evaluation of capabilities of alternate advanced Aero Data Extraction Programs using flight test data from specified Orbiter maneuvers in order to determine which program is the more appropriate within the Shuttle-related spectrum of conditions and constraints as well as to highlight areas where effort should be expended in future developments

REQUIREMENTS:
- Shuttle Entry Air Data System (SEADS)
- Aerodynamic Coefficient Identification Package (ACIP)
- Programmed Test Inputs (PTI's) for in-flight maneuvers
- Aero Data Extraction Programs
- Flight Measured to Ground Analysis Data Processing (DAP)

MISSION IMPACT:
- Reaction Control System (RCS) Propellant
- Auxiliary Power Unit (APU) Fuel
- Minimal and established crew activity

COST: $144,000

G. EVALUATION OF AERO DATA EXTRACTION MANEUVER FORMATS EXPERIMENT

SUGGESTORS: MDTSCO

TYPE: Aero Data Extraction

OBJECTIVES: Evaluation of flight test data obtained from selected aero data extraction maneuver formats in order to determine preferable formats, define the deterioration in accuracy with less than optimum formats, and to compare with conclusions from studies performed solely with ground based simulations.

REQUIREMENTS:
- Shuttle Entry Air Data System (SEADS)
- Aerodynamic Coefficient Identification Package (ACIP)
- Programmed Test Inputs (PTI's) for in-flight maneuvers
- Aero Data Extraction Programs
- Flight Measured to Ground Analysis Data Processing (DAP)

MISSION IMPACT:
- Reaction Control System (RCS) Propellant
- Minimal and Established Crew Activity
- Auxiliary Power Unit (APU) Fuel Requirements

COST: $144,000

H. INSTRUMENTATION QUALITY IMPACT ON AERO DATA EXTRACTION EXPERIMENT

SUGGESTOR: MDTSCO

TYPE: Aero Data Extraction

OBJECTIVES: Compare ADE results using sensors from FCS, ACIP or combinations of both

REQUREMENTS:
- Available Software Programs (MMLE, MLSIP)
- SEADS Reference Data (p, T, α, β, $\overline{q}$)
- Appropriate flight maneuvers

MISSION IMPACT: Small

COST: $50K - $75K

I. AERO DATA EXTRACTION EXPERIMENT

SUGGESTOR: MDTSCO , NASA/DFRC, NASA/ARC, MCAIR

TYPE: Aero Data Extraction

OBJECTIVES: Extract stability and control aero coefficients from in-flight dynamic maneuvers in regions not addressed by the mainline program during OFT. In those regions addressed by the mainline program, extract data to greater accuracy as a result of improved instrumentation (SEADS), optimized motion and refined program capability.

REQUIREMENTS:
- Shuttle Entry Air Data Systems (SEADS)
- Aerodynamic Coefficient Identification Package (ACIP)
- Programmed Test Inputs (PTI's) for in-flight maneuvers
- Aero Data Extraction Program
- Flight Measured to Ground Analysis Data Processing (DAP)

MISSION IMPACT:
- Reaction Control System (RCS) Propellant
- Auxiliary Power Unit (APU) Fuel
- Minimal and Established Crew Activity

COST: $112,500

J. HYPERSONIC VISCOUS INTERACTION AERO EXTRACTION EXPERIMENT

SUGGESTORS: MDTSCO, and Rockwell International/Space Division

TYPE: Aero Data Extraction

OBJECTIVES: From vehicle-induced motion, investigate stability and control derivatives at high altitude - high Mach conditions (viscous interaction parameter, $\bar{V}_\infty > 0.015$) where real gas and viscous interaction may substantially alter Orbiter characteristics, where no significant data base from other vehicle exists, and where the mainline program will perform a limited analysis.

REQUIREMENTS:
- Shuttle Entry Air Data Systems (SEADS)
- Aerodynamic Coefficient Identification Package (ACIP)
- Programmed Test Inputs (PTI's) for in-flight maneuvers
- Aero Data Extraction Program
- Flight Measured to Ground Analysis Data Processing (DAP)

MISSION IMPACT:
- Reaction Control System (RCS) Propellant
- Auxiliary Power Unit (APU) Fuel
- Minimal and Established Crew Activity

COST: $90,000

K. INFLUENCE OF REACTION JET FIRINGS ON ORBITER FLIGHT CONTROL CHARACTERISTICS EXPERIMENT

SUGGESTOR: DFRC, MDTSCO, and Honeywell, Inc.

TYPE: Aero Data Extraction

OBJECTIVES: Determine magnitude of total change in Aero coefficients due to firing RCS jets; determine breakdown of components into impingement, interaction, and carry-over.

REQUIREMENTS:
- SEADS
- SUMS
- Additional analysis
- ADE Programs to identify RCS impact

MISSION IMPACT: Small

COST: $110K - $170K

L. ANALYTICAL REDUNDANCY FOR DETECTING SENSOR FAILURE EXPERIMENT

SUGGESTOR: Honeywell, Inc.

TYPE: Miscellaneous

OBJECTIVES: Use the Orbiter as a test bed to perform a logical follow-on to the HI A-7D flight test program which is being performed in the near future to provide credence to the maturity and feasibility of analytical redundancy. The Shuttle will afford a much wider flight environment to test the capabilities of analytical redundancy.

REQUIREMENTS:
- Develop off-line capability to design and analyze Analytical Redundancy for Orbiter.
- Utilize man-in-loop facility (SAIL) to finalize Analytical Redundancy scheme.
- Develop Orbiter GPC software to perform redundancy scheme in parallel with mainline system.

MISSION IMPACT: Small

COST: $510K

M. ALIGNMENT TRANSFER FROM SHUTTLE TO PAYLOAD EXPERIMENT

SUGGESTOR: NASA/MSFC and MDAC-HB

TYPE: Miscellaneous

OBJECTIVES: Develop the flight software for one or more schemes to accurately transfer the Orbiter inertial reference to an inertial system in the payload bay. Demonstrate the feasibility and adequacy of the in-flight procedure for each scheme.

REQUIREMENTS:
- Strap down inertial measurement unit (IMU) and associated general purpose computer mounted in payload bay.
- Software modules for the Orbiter GPC's and the payload bay computer.
- Off-line analysis to develop candidate schemes.
- Timeline and procedure development on the Shuttle Mission Simulator.

MISSION IMPACT:
- On-orbit RCS propellant.

COST: \$200K to \$350K

N. ALIGNMENT VARIATION - REFERENCE TO CARGO BAY EXPERIMENT

SUGGESTOR: Rockwell International/Space Division

TYPE: Miscellaneous

OBJECTIVE: Measure on-orbit the Orbiter structural deformations due to thermal effects using a theodolite through the payload bay window. Apply the data to develop realistic misalignment predictions in order that unnecessary payload hardware or software requirements be avoided.

REQUIREMENTS:
- Theodolite, box, mounting jig, and targets
- Preflight crew training
- Software development for postflight reduction
- Crew timeline

MISSION IMPACT: Less than one hour crew time

COST: \$75K

5.0 SELECTED EXPERIMENT RANKING

A step-by-step selection criterion was established to impartially judge the relative merit of each experiment and definitively rank each experiment relative to the others. The three criteria selected for the comparision were 1) technical value, 2) feasibility, and 3) costs. The system of rank selected for technical value is shown in Table 3 and designed to discriminate on the basis of the potential for improvement in the Orbiter or future spacecraft systems. The system of rank selected for feasibility is shown in Table 4 and is designed to discriminate on the basis that the specified experiment is feasible with existing technology. The system of rank selected for cost is shown in Table 5. The various cost factors described were taken in account with the estimated dollar cost to arrive at a meaningful ranking within a low, moderate or high grouping.

TABLE 3

RANKING CRITERIA FOR TECHNICAL VALUE

RANK	TECHNICAL VALUE	DESCRIPTION
1 2 3	HIGH	HAS POTENTIAL FOR SUBSTANTIAL IMPROVEMENT IN OPERATIONAL EFFICIENCY, SAFETY OF SHUTTLE ORBITER OR FUTURE SPACECRAFT.
4 5 6	MODERATE	HAS POTENTIAL FOR SIGNIFICANT IMPROVEMENT IN OPERATIONAL EFFICIENCY OF SHUTTLE ORBITER OR FUTURE SPACECRAFT. MAY RESULT IN INCREASED RELIABILITY OR COST SAVINGS IN FUTURE SPACECRAFT DESIGN. PROVIDES AERODYNAMIC DERIVATIVES NOT DEFINED BY BASELINE FLIGHT TEST PROGRAM.
7 8 9	LOW	ADDS TO KNOWLEDGE OF FLIGHT CONTROL OR STABILITY AUGMENTATION PERFORMANCE. IMPROVES CONFIDENCE OF ESTIMATION OF AERODYNAMIC DERIVATIVES.

TABLE 4

RANKING CRITERIA FOR FEASIBILITY

RANK	FEASIBILITY	DESCRIPTION
1 2 3	CLEARLY FEASIBLE	MANEUVER WELL WITHIN PLANNED SPACE SHUTTLE ACTIVITIES. CREW WORKLOAD NEGLIGIBLE. REQUIRED INSTRUMENTATION AND HARDWARE ARE STANDARD. NO IMPACT ON SAFETY. MINIMAL SOFTWARE MODS TO GPC.
4 5 6	PROBABLY FEASIBLE	MANEUVER WITHIN SHUTTLE ORBITER OPERATIONAL CAPABILITY. CREW WORKLOAD MODERATE. SPECIAL INSTRUMENTATION OR HARDWARE REQUIRED. NO IMPACT ON SAFETY. MODERATE SOFTWARE MODS TO GPC.
7 8 9	POSSIBLY FEASIBLE	MANEUVER MARGINAL WITH RESPECT TO SHUTTLE ORBITER CAPABILITY. CREW WORKLOAD HIGH. INSTRUMENTATION OR NEW HARDWARE PUSHING STATE-OF-THE-ART. CONSIDERABLE SOFTWARE MODS TO GPC. MODERATE SAFETY IMPLICATIONS.

TABLE 5

RANKING CRITERIA FOR COST

RANK	COST	DESCRIPTION
1 2 3	LOW (LESS THAN $100K)	VERY LITTLE IMPACT ON MISSION TIME LINES. VERY LITTLE ADDITIONAL INSTRUMENTATION, HARDWARE, OR SOFTWARE REQUIRED. STANDARD DATA REDUCTION SATISFACTORY. DATA ANALYSIS TASK LOW. NO DEDICATED SIMULATOR TRAINING REQUIRED.
4 5 6	MODERATE ($100K TO $300K)	MODERATE IMPACT ON MISSION TIME LINES. SOME ADDITIONAL INSTRUMENTATION, SOFTWARE OR HARDWARE REQUIRED. SPECIAL DATA REDUCTION REQUIRED. DATA ANALYSIS TASK SUBSTANTIAL. MODERATE SIMULATOR TRAINING REQUIRED.
7 8 9	HIGH (GREATER THAN $300K)	SUBSTANTIAL IMPACT ON MISSION TIME LINES. HIGH COST ADDITIONAL INSTRUMENTATION, HARDWARE OR SOFTWARE REQUIRED. DATA REDUCTION AND ANALYSIS TASK FORMIDABLE. DEDICATED SIMULATOR TRAINING REQUIRED.

A ranking in the three categories (feasibility, technical value, and cost) was selected for each of the experiments described in Section 4.0. These are presented in Table 6. Each of the three ranking categories were given a weighting factor and combined to arrive at an overall rank. The magnitudes

TABLE 6

EXPERIMENT RANKING

EXPERIMENT	TECHNICAL VALUE	FEASIBILITY	COST	TOTAL WEIGHTED	TOTAL EQUAL
ACIP VS. PRIMARY SENSOR	5	2	3	3.05	3.33
ROTATIONAL HAND CONTROLLER	2	5	4	3.95	3.67
ALIGNMENT VARIATION	6	4	3	3.95	4.33
ADE	6	3	4	4.05	4.33
VISCOUS INTERACTION	6	3	4	4.05	4.33
DECREASED SAMPLE RATE	5	4	5	4.65	4.67
COMPARISON OF ADE TECHNIQUES	7	3	5	4.7	5
FREE DRIFT MODES	8	4	4	4.8	5.33
INFLUENCE OF JET FIRINGS	3	6	6	4.95	5
ORBITER FLYING QUALITIES	7	4	5	5.05	5.33
ADE MANEUVER FORMATS	7	4	6	5.5	5.67
ALIGNMENT TRANSFER	4	6	6	5.6	5.33
ADAPTIVE GAIN CHANGE	3	7	7	6.2	5.67
ANALYTICAL REDUNDANCY	6	7	8	7.25	7

of the weighting factors are:

- Technical value - 20%
- Feasibility - 35%
- Cost - 45%

These weights were coordinated with the LARC technical monitor and were based on the following philosophy:

- "Technical value" is very subjective and hence was given the lowest weight
- Considering the overall OEX program, funding is limited and hence "cost" was given the highest weight.

The combined rankings are also shown in Table 6 and the experiments are listed in the order of highest (low number) to lowest (high number) rank; i.e. most to least desirable experiment. Also for comparison, a combined rank is shown in Table 6 which was obtained assuming equal value for the three ranking categories.

As indicated in the table, the highest ranked experiment is the "ACIP vs. Primary Sensor" comparison. Cost of this experiment is low, feasibility is very good since it needs no special data or has any impact on the Orbiter, and it has reasonable technical value in illustrating whether flight control instruments can be used to obtain data for aero extraction or should have that requirement included during the initial procurement evaluation. In either case, savings can be realized by obtaining one sensor package for the combined task.

The lowest ranked experiment dealt with "Analytical Redundancy." Cost was high, feasibilty was low because of the considerable software additions required in the Orbiter GPC's, and the technical value is questionable;

i.e., add considerable software to save a minimum of LRU's.

In general, the aero data extraction experiments rank reasonably high because they had little effect on Orbiter hardware or software. In addition, it was assumed that SEADS, SUMS and ACIP were available to aid in these experiments without charging to their cost.

The highest ranking "flight control" experiment was the "Modified RHC". It was ranked as having the highest technical value based on potential for substantial improvement in Orbiter operational efficiency.

It is of interest to note that the combined rankings didn't vary much between the weighted combinations previously described and the equal value combinations.

6.0 CONCLUSIONS

Feasibility has been established for 14 experiments which can be classified into the general experiment categories defined by the study SOW. Potentially, the experiments will provide enhancement of the operational efficiency of the Shuttle or contribute to the research and technology base for future spacecraft design. All of the 14 experiments are good candidates for the OEX program regardless of the relative ranking established in Section 5.0. None have any significant safety implications for the Orbiter, and most will cost less than $200K.

It is also recognized that a number of the selected experiments border on being very close to the mainline Shuttle Program interests and effort. They were selected by the study contractor as experiments because of MDTSCO's knowledge of the present mainline effort. However, what is considered an experiment today, may tomorrow become part of the mainline effort.

APPENDIX A

SELECTED EXPERIMENT STUDY RESULTS

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EXPERIMENT A: MODIFIED ROTATIONAL HAND CONTROLLER EXPERIMENT

1.0 Background and Objectives

A pilot-induced oscillation (PIO) occurred during the Shuttle ALT Free-Flight 5 in the vicinity of touchdown. During this flight, the pilot was trying to touch down as close as possible to a spot on the concrete runway. (On all of the previous ALT flights, the vehicle was flown to the lakebed runway with little concern about the actual touchdown point). With his approach condition not quite right for landing at the designated spot, the pilot attempted some last-second maneuvers to correct his conditions and induced the PIO. Considerable post-flight analyses and man-in-the-loop simulations have been performed to determine the cause of the PIO and to make the Orbiter manual control system less susceptible to the increased pilot "gain" during "stress" conditions. One modification that was implemented was to increase the rotational hand controller (RHC) spring forces in pitch and roll along with some changes in the RHC output parabolic shaping software which intended to reduce the increased pilot gain effect.

Through our (MDTSCO) involvement with the PIO problem, we learned of a similar problem that McDonnell Aircraft Company (MCAIR) had on their past Survivable Flight Control System (SFCS) Program (680J) performed for the Air Force Flight Dynamics Laboratory (AFFDL), Wright-Patterson Air Force Base. The F-4 pilots for this fly-by-wire control system complained of roll hand controller sensitivity during flight at altitude, but lower stick gain resulted in PIO near touchdown under conditions of turbulence. The MCAIR Digital Flight Control System (DFCS) studies encountered similar problems on a hybrid simulator. Subsequently MCAIR developed a combination roll hand controller spring force and output shaping that was test flown on the CALSPAN NT-33A test aircraft and was acceptable to the pilots.

The MCAIR development was suggested as a potential improvement in reducing the Orbiter manual FCS PIO tendency. However, the time required to implement the hardware modifications to the Orbiter RHC was not compatible with the Orbiter simulation schedules and the overall improvement program. In lieu of incorporating the MCAIR development in the present mainline Orbiter FCS, an experiment utilizing the unique MCAIR output characteristics in the roll channel is warranted for the sake of evolving technology improvement and to possibly provide more desirable on-orbit RHC spring forces. (The basis for this latter objective will become apparent following the discussion of the MCAIR scheme.)

2.0 Feasibility

The MCAIR development took place following the completion of the SFCS program, during which the pilots were never satisfied with the roll RHC characteristics as explained in the first section. Considerable trial and error variations were tried during the DFCS studies by shaping the output command with little success. Variation in stick force gradients resulted in the most favorable pilot response. However, no one force gradient was acceptable for all flight missions. A two-slope stick force gradient was proposed. This was combined with output shaping which produced a linear roll rate command vs. stick force gradient and provided the most acceptable output characteristic to the pilots. These characteristics are shown in Figure A-1.

The values shown are those used in the CALSPAN NT-33A test aircraft. The linear roll rate command vs. stick force is a natural characteristic for most pilots since this is normally attained in an aircraft with mechanical

NT-33A ROLL CONTROLLER FLIGHT TEST GRADIENTS

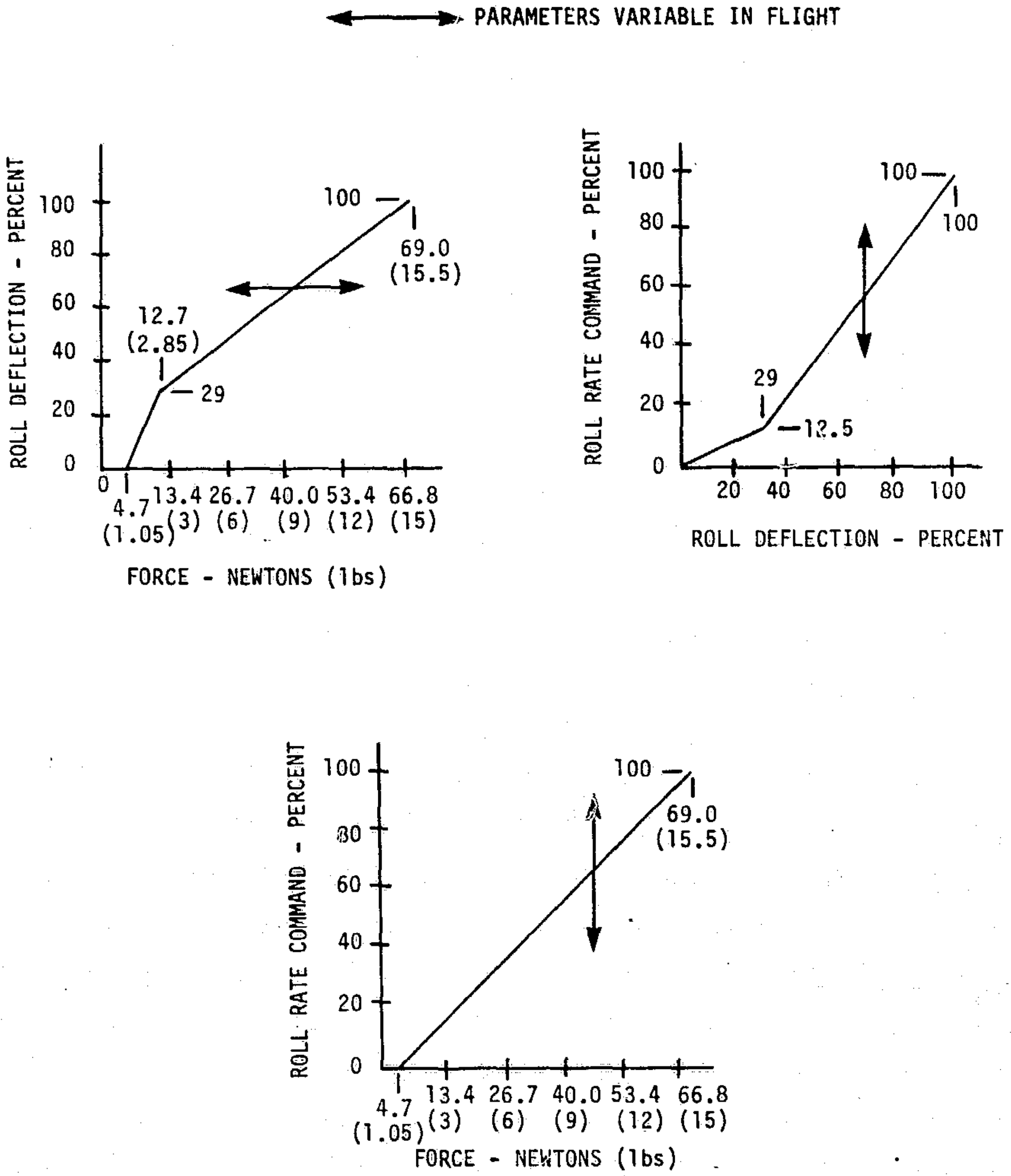

FIGURE A-1

controls and a center stick.

The roll characteristics of the Orbiter RHC are shown in Figure A-2 for both the ALT configuration and the new configuration developed during the post ALT PIO improvement program. The basic objections to the ALT RHC roll characteristics were the low rate output for stick deflection after breakout and the relatively light spring force which allowed the pilot to easily over-command in response to not achieving much output after breakout. The changes to the characteristics which were developed on the Orbiter Aeroflight Simulator (OAS) in January were intended to rectify the pilot objections. However, the resulting characteristics are considerably different than the MCAIR configuration. In addition, by doubling the spring constant, the forces are rather large for on-orbit control. With the MCAIR configuration, the initial spring force would remain the same as the ALT values, which were originally designed to accommodate both on-orbit and aeroflight requirements. Proposed roll RHC characteristic are also shown in Figure A-2. A more sophisticated characteristic could incorporate different spring constants and shaping for roll right or left.

3.0 Requirements

Basic requirements for this experiment are:

- Modified RHC for simulator studies.
- Pre-flight evaluation on Shuttle Mission Simulator (SMS) or AMES motion base simulator.
- Flight-qualified modified RHC.
- Modifications to FCS or RHC SOP software.
- Installation of modified RHC(s) in Orbiter.

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ORBITER RHC ROLL OUTPUT CHARACTERISTICS

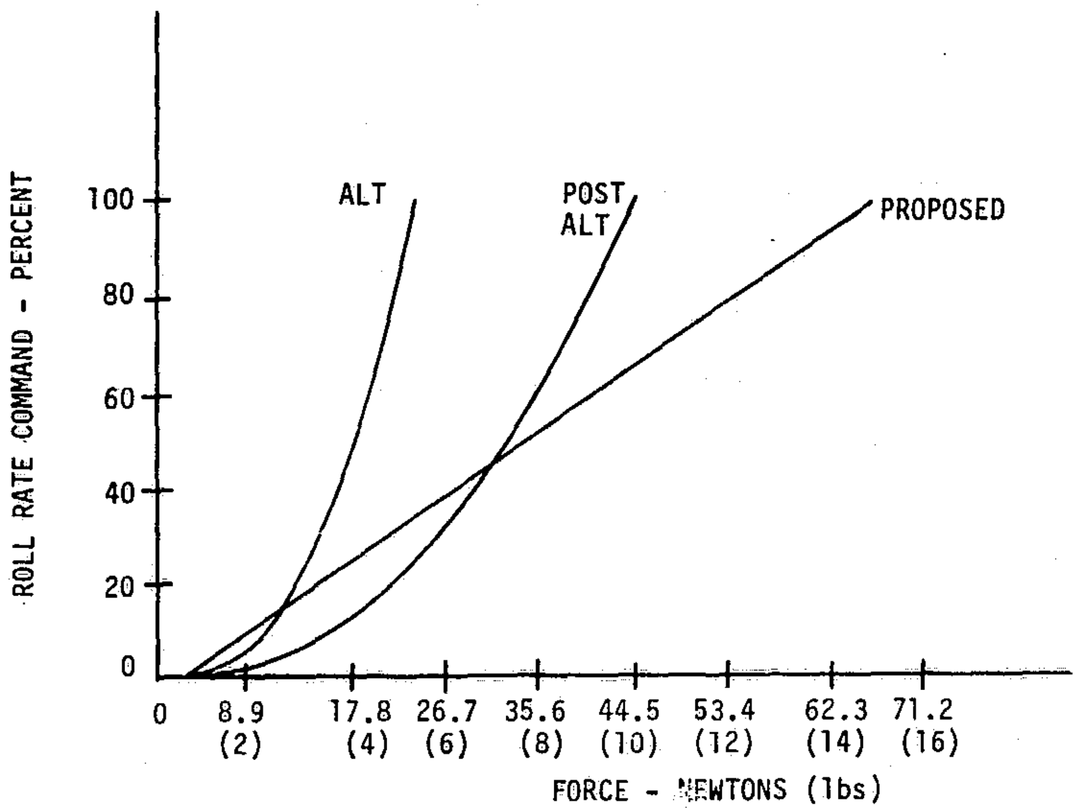

FIGURE A-2

forward crew positions. Modified RHC's at both positions have greater hardware costs, while software costs are greater for a modified RHC at only one position. The experimental value of different RHC's in the two crew positions is obtained through the direct in-flight comparison capability.

RHC Modifications

Modifications to a hand controller required for the simulators can be performed for little cost at the JSC Engineering Standards Calibration Laboratory (ED8) as were the mods which recently doubled the spring constants. A preliminary inspection indicated that adequate space in the base enclosure is available for the dual spring system that is required to implement the MCAIR roll configuration. It is also believed a dual spring system can be developed for the pitch channel to provide a relatively low spring constant for the on-orbit attitude control and yet provide the desired characteristics for the landing task.

Informal discussions with Honeywell personnel to obtain rough order of magnitude (ROM) cost estimates for flight-qualified units resulted in the following:

- The ROM costs discussed are based on the present Orbiter RHC production line being active. This will be true for the next two or three years.
- A RHC that requires considerable redesign would cost approximately $50K for modifying one of the production articles.
- The possibility of modifying the present soft stop breakpoint (85% of RHC deflection), so it would break at a new desired point (approx 33%), would reduce the cost to about $20K.
- Requiring a complete new build, would cost approximately $100K.

Software Modifications

The magnitude of the software (S/W) modifications will depend on whether one or two modified RHC's are installed at the crew station positions and if both roll and pitch channels are modified. Using only one modified RHC, a software module must be added to the RHC system operating program (SOP) which will provide the shaping to the modified RHC output prior to summing the right and left RHC outputs. This module must provide the two-slope shaping and the inverse of the baseline shaping in the entry FCS S/W which must remain unchanged for the ummodified RHC. This is illustrated in Figure A-3. For two modified RHC's, the baseline quadratic shaping

TWO AXES RHC SOP FUNCTIONAL BLOCK DIAGRAM

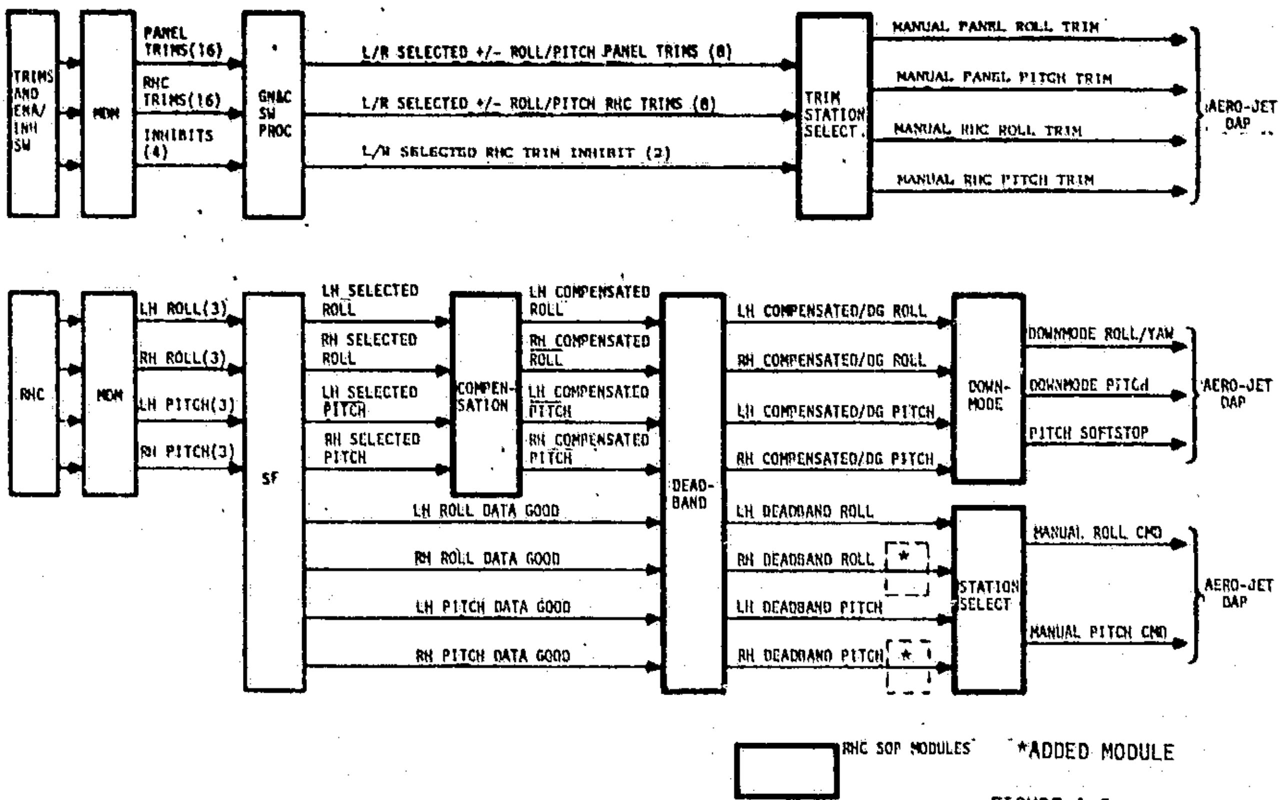

FIGURE A-3

Figure 4.53-1. 2AXRHC SOP Functional Flock Diagram

software in the FCS must be changed to the linear two-slope shaping; however, no SOP modifications are required. Based on our simulation coding of the FSSR's and a one-to-one ratio between FORTRAN and HAL code, it was estimated that 166 words of core were required for a one RHC modification in roll and pitch or 51 words for modifying both controllers. Based on ROM costing discussed in Section 4.0, this would amount to approximately $50K or $15K, respectively.

Simulation Evaluation

A motion base simulator will be required to evaluate the modified RHC in stress situations in order to judge the effects of its characteristics on PIO tendencies versus the baseline and to obtain pilot comments. No direct costs would be incurred for time used on the SMS which will be operational throughout the STS program.

Some costs may be incurred for S/W patches to modify the simulator tapes; however, it is assumed these are part of the overall costs of S/W modifications described in the preceding paragraph. Scheduling of time on the SMS may be the major obstacle encountered; however, it is premature to consider any problem at this time.

Orbiter Integration

Installing the modified RHC will be straightforward since the electrical characteristics and the base envelope will be unchanged.

4.0 Mission Impact

Very little impact on mission requirements will be incurred. With only one modified RHC in the Orbiter, direct comparisons can be made for certain maneuvers on-orbit and during entry. If both stations have the modified controllers, evaluation will be based on pilot comments for prescribed situations.

EXPERIMENT B: ADAPTIVE GAIN CHANGER

1.0 Background and Objectives

The baseline Orbiter entry Flight Control System (FCS) utilizes dynamic pressure ($\bar{q}$), Mach number (M), and angle of attack (α), derived from the Air Data System (ADS) to vary gains below Mach 2.5. However, the ADS is not quad-redundant. Dual or single point failures exist, resulting in dilemma situations which Redundancy Management (RM) cannot resolve. For OFT, a "default system" will send a pre-stored $\bar{q}$ profile to the FCS for gain changing in the event a dilemma situation arises. This profile shape is trajectory and weight dependent and hence will burden the operational vehicle with I-load changes for each mission. The experiment concept discussed herein is a potential solution for making the FCS independent of the ADS and making the gain changing function quad-redundant.

This experiment was suggested by MCAIR personnel. The basic AGC concept was instrumental for use with the Survivable Flight Control System (SFCS) on the 680 J program sponsored by AFFDL in the early 1970's. In addition, it was also considered in recent F-18 design activities, also pertaining to ADS/FCS redundancy considerations.

The SFCS was a quad-redundant analog fly-by-wire control system demonstrated in an F-4 airplane. The AGC uses the theory that pitch rate loop high frequency gain margin is proportional to $KM\delta_E$; where K is the flight control gain and $M\delta_E$ is the aerodynamic gain. Hence, if measure of $M\delta_E$ can be obtained, K can be adjusted to maintain acceptable gain margin. The scheme was demonstrated during the 680 J flight test program and found to work successfully only in regions of low elevator effectiveness. However, feasibility studies discussed in Section 2.0 demonstrated that the SFCS problems can be circumvented. Hence, the prime objectives of this experiment are the following:

- Demonstrate the capability to make accurate continuous measurements of elevator effectiveness.
- Demonstrate that the concept can also be used for changing gains in the lateral-directional channel.
- Demonstrate that the required dither signal to the elevators is not objectionable to the crew and will not result in inadvertent secondary actuator tripouts or excessive consumption of hydraulic resources.
- Obtain confidence in the system as a means to make the FCS independent of the ADS.

2.0 Feasibility

The present Orbiter FCS utilizes parameters (α_N, $\bar{q}_N$, M_N) derived from the navigation (NAVDAD subroutine) system to vary several gains in the longitudinal and lateral-directional channels throughout the entry phase down to the TAEM interface (Mach 2.5). At the TAEM interface, these parameters are available from the ADS. Since the $\bar{q}_N$ errors from NAVDAD will eventually become excessive in the pressence of 3σ design winds, ADS parameters are currently switched in to control the gain variations. However, as previously mentioned, RM dilemma situations do exist for the ADS and a "default" system was judged necessary for OFT. In the event "default" is initiated, Mach from NAVDAD and a fixed value of α are sent to the FCS. A canned $\bar{q}$ profile as a function of earth relative velocity is used for the dynamic pressure parameters. Feasibility studies by MDTSCO indicated that NAVDAD Mach number and a fixed value of α are adequate, since the dependency of the FCS on these parameters is not very significant, at least below

Mach 2. Hence, if a system can be devised to determine an accurate measurement of aerosurface effectiveness in the presence of winds, the FCS can be made independent of the ADS.

Longitudinal Channel

As previously mentioned, the AGC is based on using measured elevator effectiveness to maintain a constant stability margin in the Shuttle longitudinal axis. Results presented below indicate that implementation of this concept can be realized in a straightforward manner with minimal impact to other mission objectives.

The complete frequency response for pitch angular acceleration per elevator deflection, ($\dot{q}/\delta_E$) is a function of $C_{m\delta_E}$ and the other pitch derivatives. For frequencies of interest to Shuttle short-period stability (3 to 8 rad/sec) all terms except for that containing $C_{m\delta_E}$ contribute negligibly Hence $\dot{q}/\delta_E \simeq C_{m\delta_E} \dfrac{S\bar{q}c}{I_{yy}}$ is a fully adequate approximation for this analysis. This can also be written $\dot{q}/\delta_E = M\delta_E$.

Constant closed-loop gain margin can be obtained by allowing for $C_{m\delta_E}$ Mach dependence and scheduling forward loop gain based on dynamic pressure ($\bar{q}$). The current baseline FCS is mechanized in this manner.

Alternately, one could measure $\dot{q}/\delta_E$ directly inflight and use this value as a gain divisor in the FCS. Such a scheme would not rely on dynamic pressure. Since adequate information for angle of attack and Mach could be obtained from the NAV function, dependency on the air data system could be avoided completely. Furthermore, $C_{m\delta_E}$ variation with Mach would not have to be known in advance.

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Attempts to measure $M_{\delta E}$ inflight are not new. MCAIR has researched the problem for other aircraft using only the existing control surface commands from a rate-augmented FCS. Problems were encountered during periods of low control activity and hence they added a sinusoidal dither signal to drive the elevators. Problems were still encountered in their scheme due to actuator non-linearities. The approach discussed below uses an auxiliary dither signal at a fixed frequency and direct measurement of elevator motion to insure reliable M_{δ_E} estimation.

Figures A-4 and A-5 show the shuttle $\dot{q}/\delta_E$ and N_Z/δ_E response for Mach 2.5 and .5 respectively. From the $\pm 10\%$ $M\delta_E$ contours it is clear that $\dot{q}/\delta_E$ is a legitimate indicator of $M\delta_E$ above 2 rad/sec. Note the attenuation of N_Z in the cockpit relative to N_Z at the CG for the higher frequencies.

ORBITER PITCH ACCELERATION ($\dot{q}$) AND NORMAL ACCELERATION FREQUENCY RESPONSES FOR ELEVATOR INPUTS

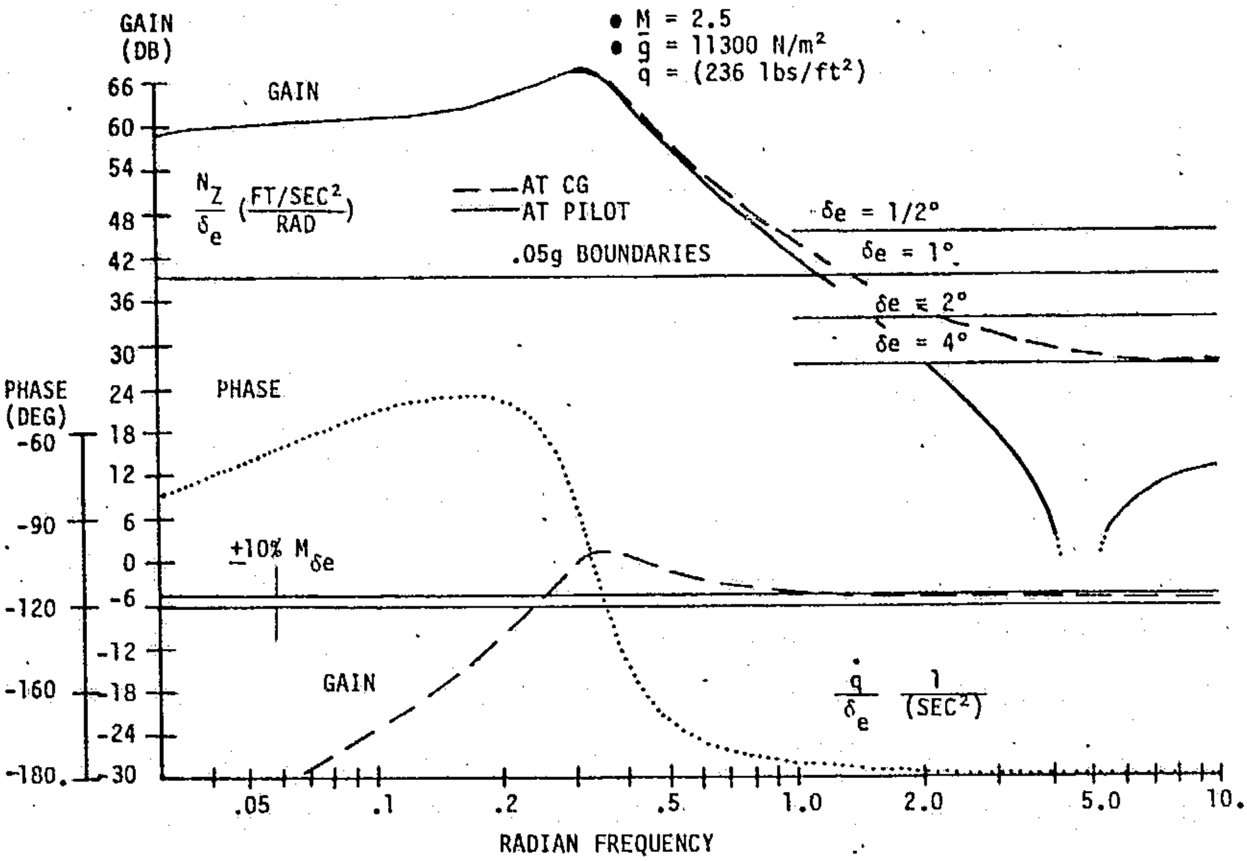

FIGURE A-4

ORBITER PITCH ACCELERATION ($\dot{q}$) AND NORMAL ACCELERATION FREQUENCY RESPONSES FOR ELEVATOR INPUTS

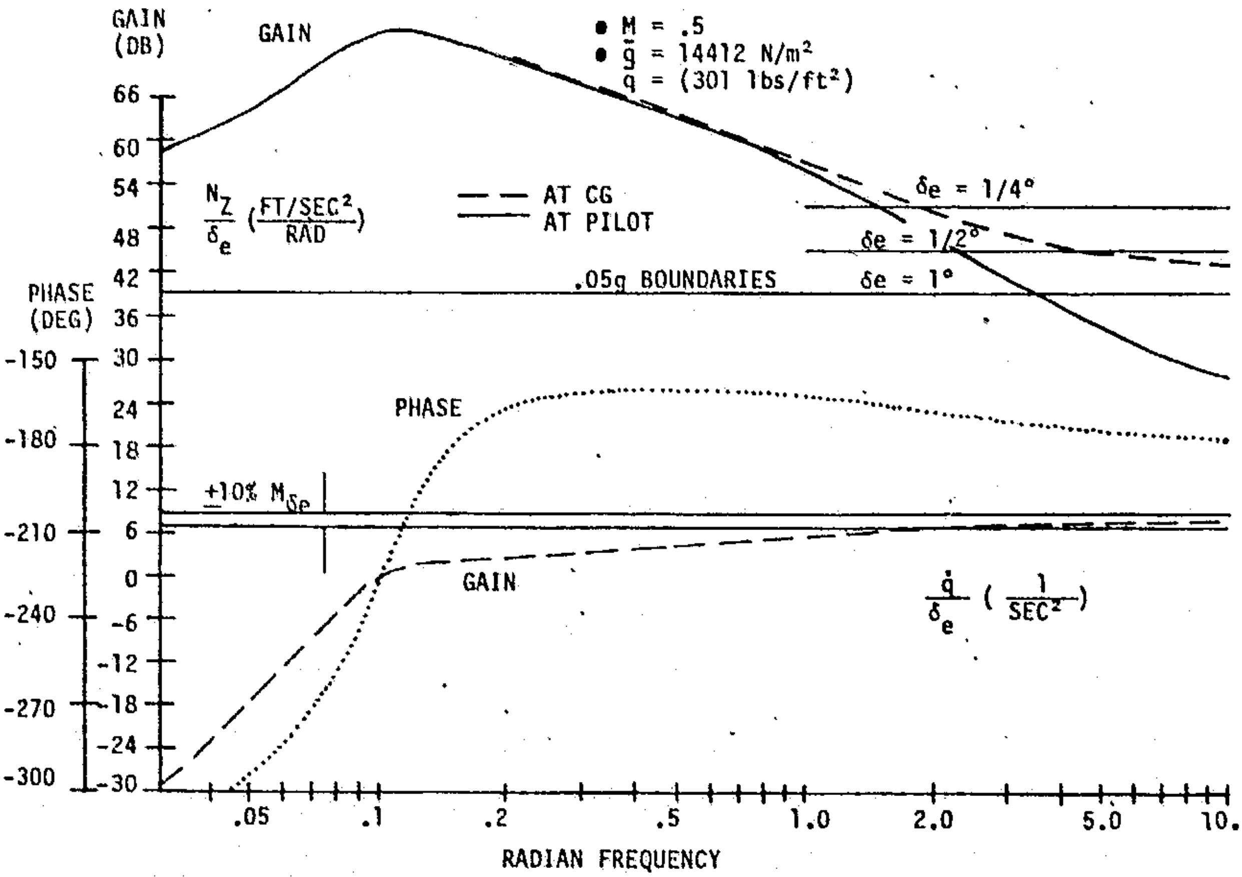

FIGURE A-5

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Figures A-6 and A-7 attempt to clarify the pertinent considerations. These are:

- M_{δ_E} error (<10%)
- N_Z at the cockpit (<.05g)
- Tail-wags-dog component of q (<10%)
- Pitch rate amplitude (>4 quantization increments)

The first two of these have already been mentioned. Tail-wags-dog results strictly from inertial coupling between the elevons and the vehicle. Its magnitude varies as the square of the frequency and contaminates the M_{δ_E} estimate rapidly above 6 to 10 rad/sec. Lastly, pitch rate (which decreases linearly with frequency) is quantized at .04°/s. It will be shown that rates as low as four times this level can be processed satisfactorily.

Consideration of the above issues pointed to 4 rad/sec as a viable dither frequency. Amplitude may be either fixed at 1° or allowed to track computed M_{δ_E} if desired.

ORBITER RESPONSES/BOUNDARIES

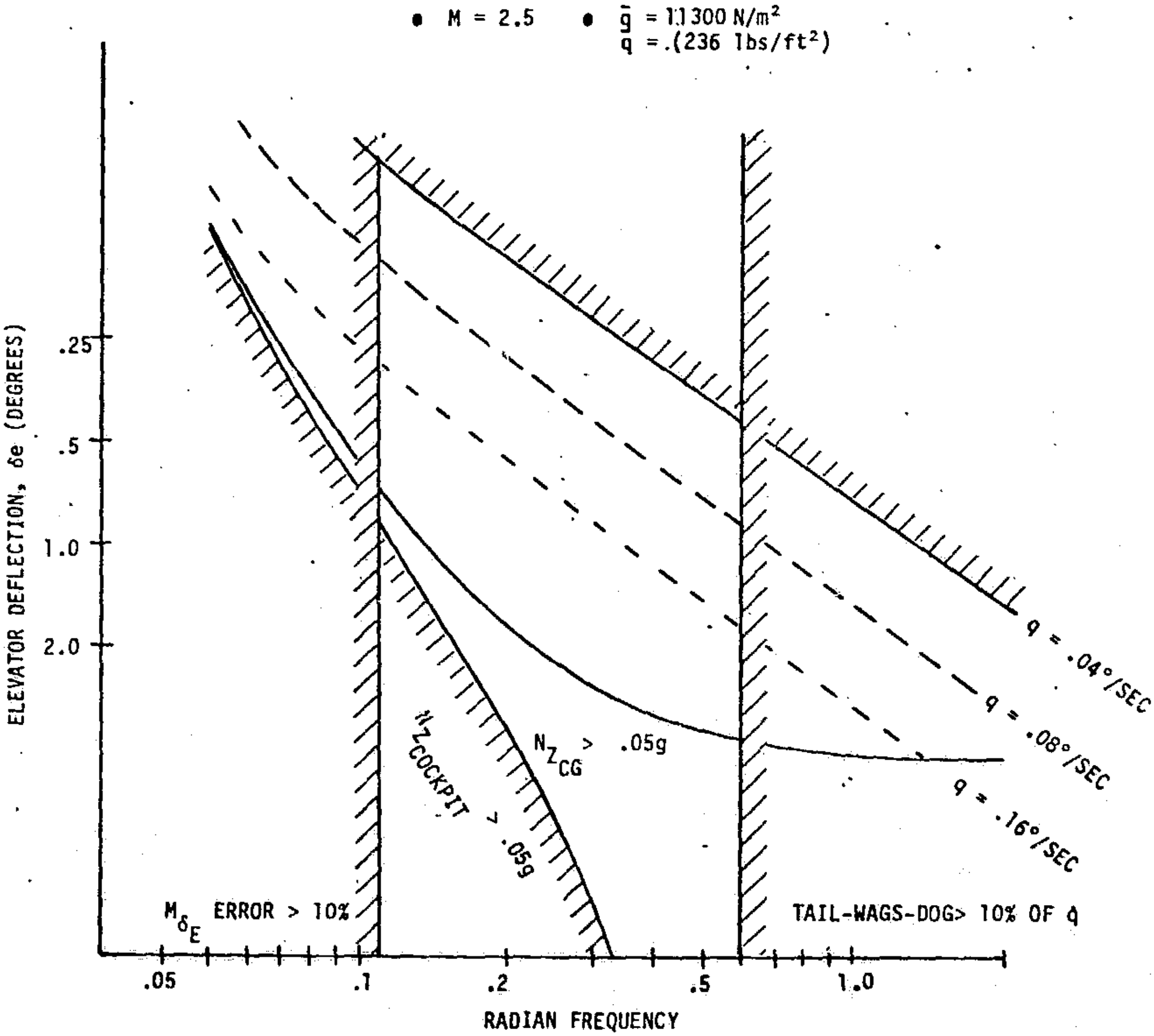

FIGURE A-6

ORBITER RESPONSES/BOUNDARIES

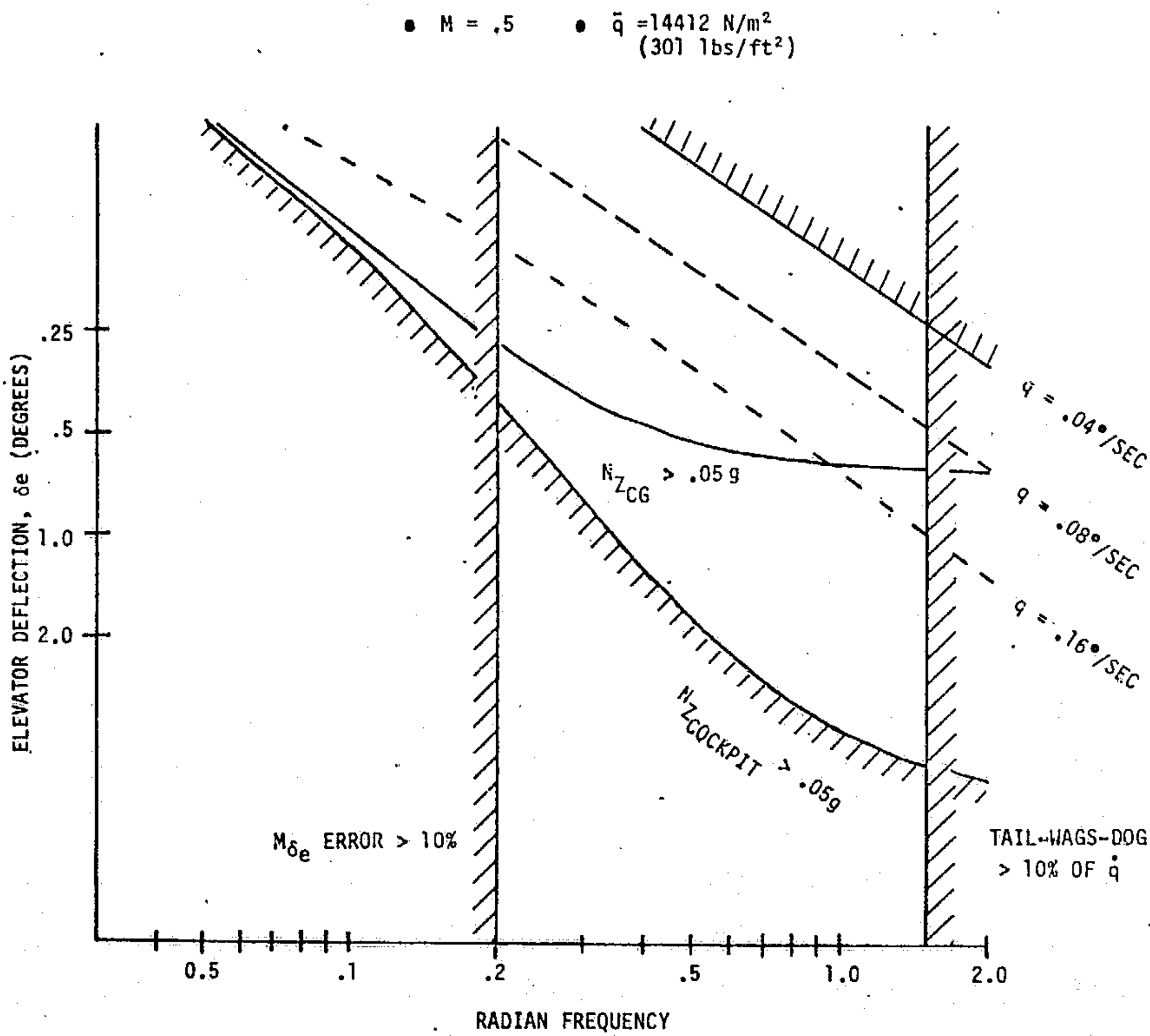

FIGURE A-7

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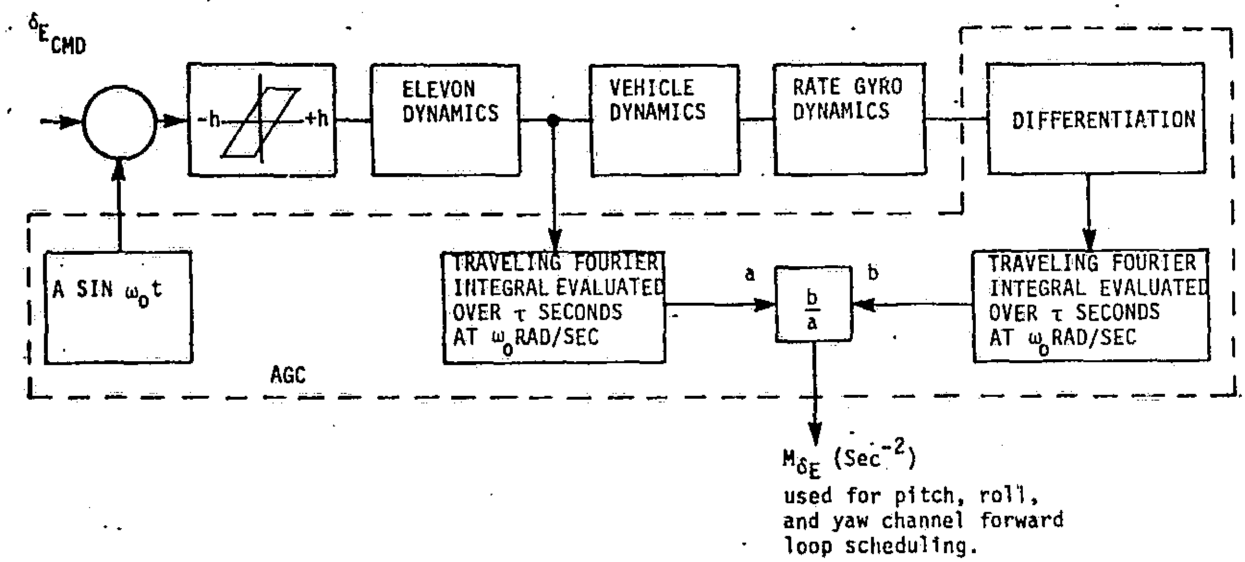
Figure A-8 shows the proposed configuration for the Adaptive Gain Changer (AGC). The only operations required are sinusodial dither injection, differentiation to obtain pitch acceleration, Fourier integration (mechanized as a traveling summation), and a division. The output (estimated $M\delta_E$) can subsequently be used as a divisor in the pitch rate forward loop control path (replacing dynamic pressure).

Figure A-9 shows the dependency of algorithm accuracy on integration interval (τ) and pitch rate quantization size. For $\tau > 3$ seconds, $M\delta_E$ convergence is not significantly improved yet additional delay is incurred due to reliance on past values. Because of this, plus the Fourier stipulation that τ be set to an integer multiple of the dither period, 3.14 seconds was adopted.

The profile of a nominal trajectory is shown from Mach 2.5 to touchdown in Figure A-10. Note the rapid change in dynamic pressure just prior to touchdown.

Figure A-11 contains performance data for a nominal run using AGC estimated $M\delta_E$ values for gain scheduling. The source of the 9% error incurred in the lower Mach regime is not as yet well understood. The deviation from actual $M\delta_E$ at touchdown is largely due to the inherent 1.5 second ($\tau/2$) delay in the $M\delta_E$ estimate during a period of rapid $M\delta_E$ change. The FCS gain variation differs considerably from the $M\delta_E$ estimates because the GDQ mach variation is very approximate in the baseline system.

MDTSCO ADAPTIVE GAIN CHANGER



TRIAL VALUES

τ	3.14 SEC (MUST BE A MULTIPLE OF ω_0 PERIOD)
ω_0	4 RADS/SEC (COMPROMISE BETWEEN M_{δ_E} ACCURACY AND GYRO RESOLUTION)
A	.25° TO 1.0° (A MUST PROVIDE $\lvert q \rvert \geq$ .16 DEG/S YET NOT RESULT IN $\lvert Nz \rvert >$ .05g AT THE COCKPIT)
h	.1° (FOUR TIMES THE ANTICIPATED ELEVON HISTERYSIS)

FIGURE A-8

PERFORMANCE OF FOURIER ALGORITHM

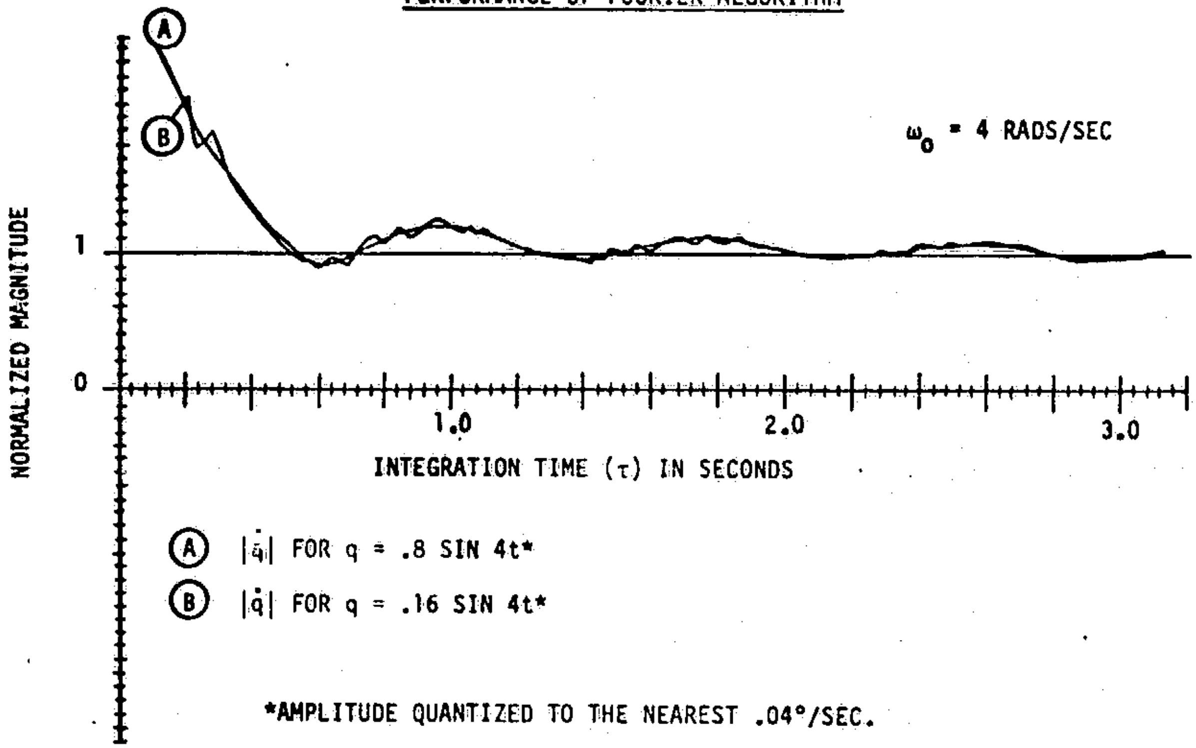


FIGURE A-9

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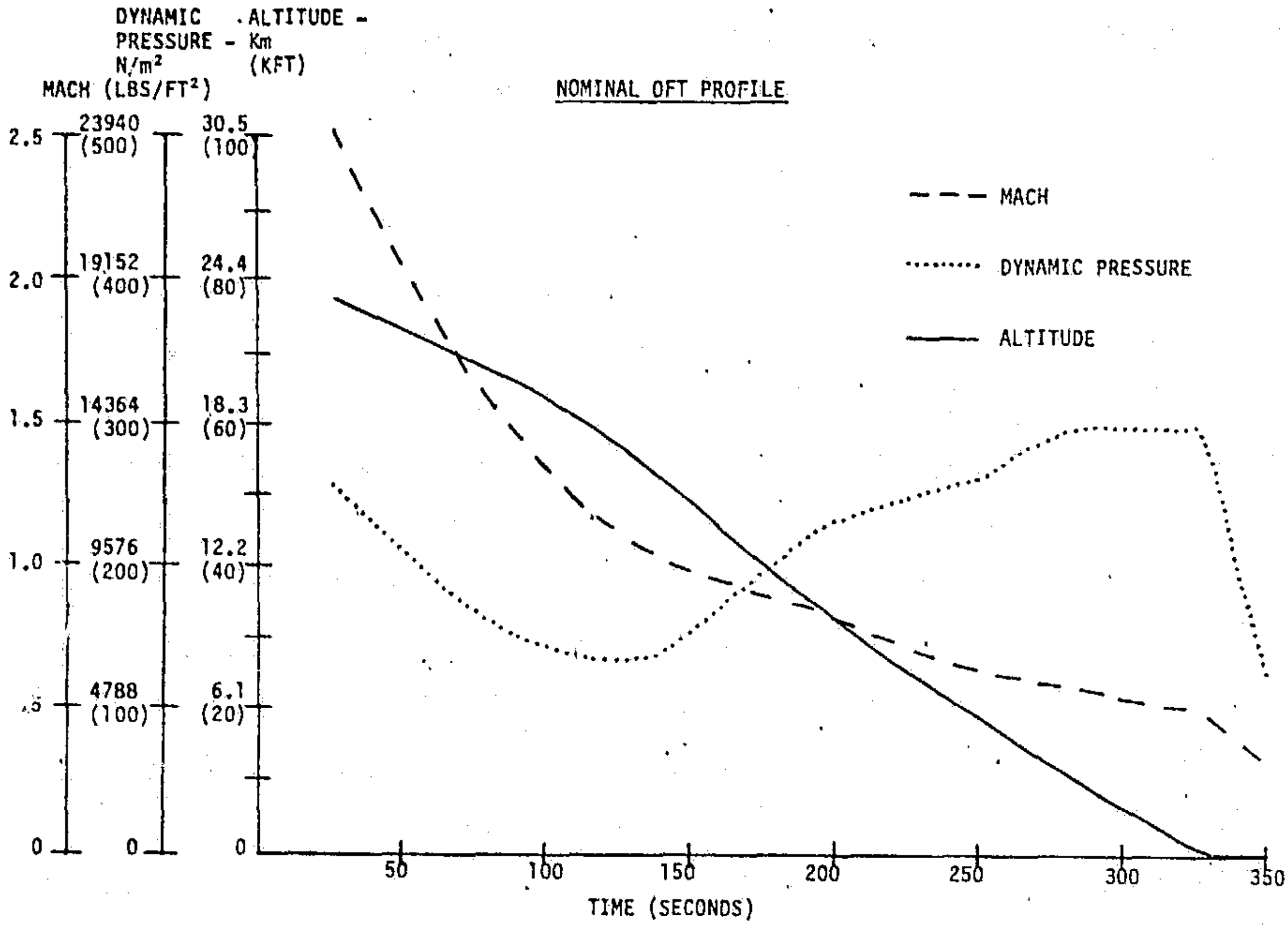
NOMINAL OFT PROFILE
DYNAMIC PRESSURE - N/m² (LBS/FT²)
ALTITUDE - Km (KFT)
MACH
2.5
2.0
1.5
1.0
.5
0
23940 (500)
19152 (400)
14364 (300)
9576 (200)
4788 (100)
0
30.5 (100)
24.4 (80)
18.3 (60)
12.2 (40)
6.1 (20)
0
MACH
DYNAMIC PRESSURE
ALTITUDE
50
100
150
200
250
300
350
TIME (SECONDS)

FIGURE A-10

AGC PERFORMANCE

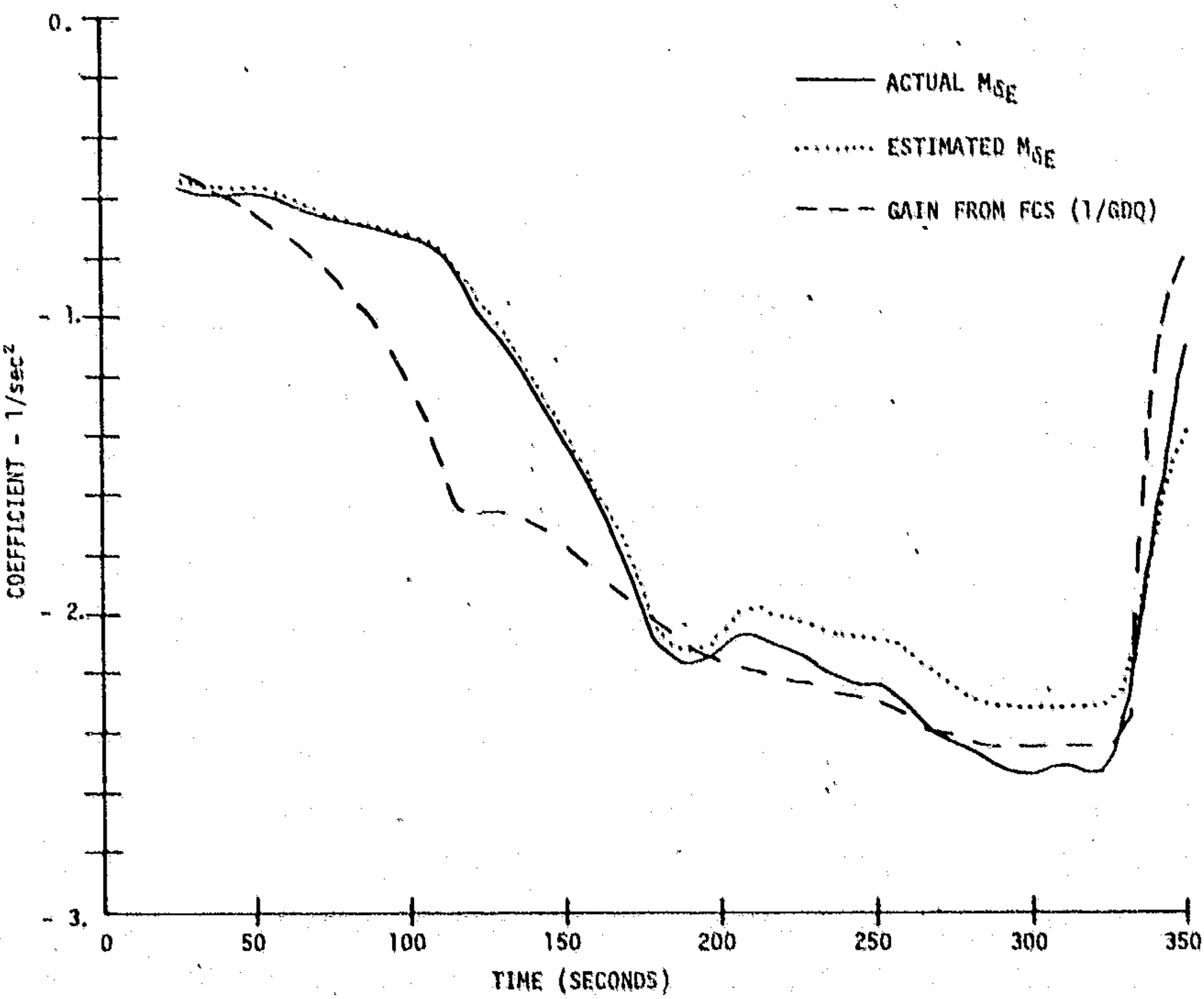

FIGURE A-11

Lateral/Directional Channel

Preliminary investigation of measuring aerodynamic control effectiveness in the lateral axis revealed new difficulties. Permissible lateral acceleration at the cockpit is limited to .02g for crew comfort (verses .05g for normal acceleration). This constraint combined with the roll rate quantization of .08°/s resulted in loss of accuracy for estimation of rolling moment due to ailerons (L_{δ_A}).

An alternative to making a separate determination of roll axis aerodynamic gain would be to simply use a value proportional to M_{δ_E}. This presupposes similar mach dependency for $C_{m\delta_E}$ and $C_{\ell\delta_A}$. Figure A-12 contains these plots as well as $C_{n\delta_R}$ for a Mach range of 0 to 3. With appropriate scaling the error resulting from using M_{δ_E} for the roll and yaw axes gain scheduling is $\pm$ 1.7dB and $\pm$2.5 dB respectively. Using this technique, software complexity is vastly reduced, hydraulic consumption is lowered, and undesirable vehicle excursions are minimized.

AERO EFFECTIVENESS FOR ELEVATOR, AILERON AND RUDDER

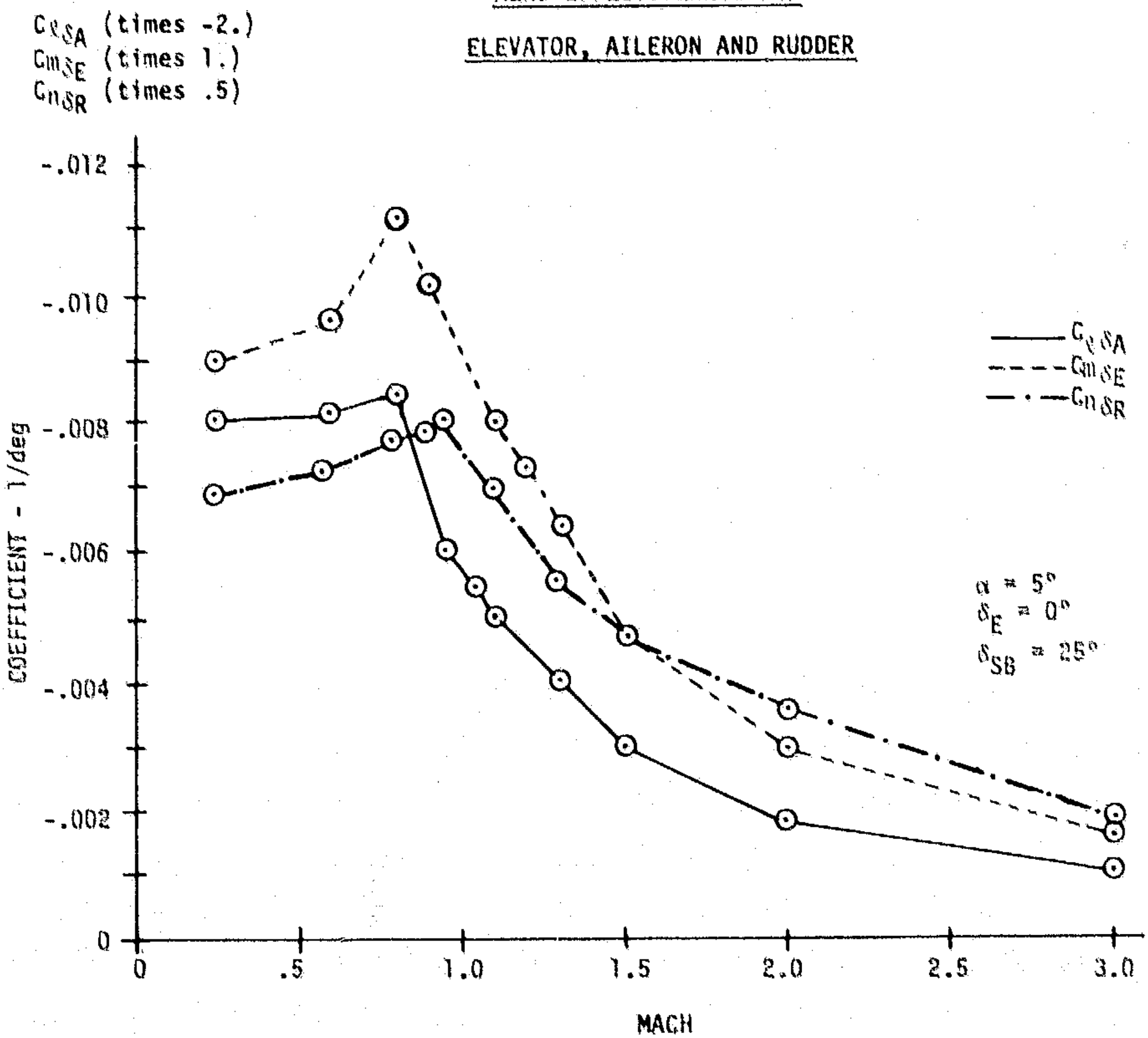

FIGURE A-12

3.0 Requirements

This experiment will be run in parallel with the baseline FCS. The performance can be assessed by comparing the computed gains from the AGC with the normal gains computed using air data parameters. The only direct influence on the FCS will be the low amplitude dither.

Basically the following will be required:

- Four to six man-months of analytical effort to provide support of algorithm checkout (using off-line simulation) and generation of coding specifications. Verification at the system integration level would be performed at SAIL. Impact to crew procedures is slight.
- Software subroutine to perform the required gain computations and and generate the elevator dither signal. The experimental software would reside in all four primary GPCs to allow the dither signal to be superimposed onto the nominal FCS elevator commands. Execution would occur at 25 Hz. It is estimated that approximately 1100 words of core are required.
- A means to activate and deactivate the dither signal. This can be accomplished through the keyboard, or more desireably, a panel switch. Following completion of OFT, one of the FCS downmoding switches may be available for this function.
- Sufficient data to fully evaluate the AGC scheme. The data can be downlisted or recorded onboard. A minimum of three parameters will be required.

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4.0 Mission Impact

The overall impact to mission objectives is minimal. Crew and ground support must verify that flight status is reasonably nominal prior to engage. The crew will initiate the test around Mach 2.5 and continue to monitor vehicle response for undesirable behavior. Cockpit motion resulting from the AGC dither should be barely noticable. Crew must disengage the experiment prior to touchdown. Disengage at gear down is recommended.

A conservative estimate of hydraulic consumption has been obtained assuming that the maximum fixed elevator dither amplitude (1°) is required for 325 seconds. At a rate of 6 lbs. hydrazine per 1000 cumulative degrees of elevon command, the anticipated usage would be 5 lbs. per hydraulic system. The increased hydraulic usage need only be considered if supplies become critical at Mach 2.5. The current Fault Detection and Annunciation (FDA) function will sound an alarm at 20% fuel remaining which could be used as a basis to forego engagement of the experiment.

Some degree of surface rate capability has also been sacrificed. This value can be as high as 4°/sec. It should be verified prior to flight that sufficient margin exists in the nominal elevon rate capability such that vehicle handling is not impaired.

EXPERIMENT C: DECREASED FLIGHT CONTROL SAMPLE RATE

1.0 Background and Objectives

This experiment was suggested by MDTSCO and General Dynamics/Convair Division. The sample rates for digital control systems have usually been chosen during the preliminary design phase and have not been parameters in the final design studies. Sample rates have been selected primarily on the basis of approximating continuous control system performance.

It is reasonable to believe that the design of digital control systems using sample data methods with sample rates as one of the design parameters would lead to more efficient digital control systems. For example, the load on the Orbiter hydraulic system could be reduced by using a lower rate to drive the control surfaces. Also, in the event that the timing load on the Orbiter GPC's becomes excessive because of future desired software additions, a reduced FCS sample rate can substantially relieve the timing problem.

The basic objectives of the experiment are to:

1. Perform a parametric stability and performance analysis of the entry control system with step size as the variable.
2. Demonstrate in-flight operation of the control system at a selected reduced FCS sample rate.

While feasibility can be adequately demonstrated by off-line analyses, the actual flight(s) will assure the community that the higher frequency bending modes and other system non-linearities will have no adverse effects in the presence of the reduced sample rate. For example, past feasibility studies of possible flight control configurations for the Saturn booster showed that bending mode effects could be reduced by decreasing the flight control sampling rate.

2.0 Feasibility

A sampled-data stability analysis of the Orbiter entry flight control system at sample rates of 12.5 and 25 Hz has shown that the 12.5 Hz rate reduces the gain margins a maximum of 1.3 db. A complete summary of the gain and phase margin differences at the various flight conditions analyzed for the lateral/directional channels is presented in Table A-1.

TABLE A-1

FREQUENCY RESPONSE COMPARISON FOR 12.5 AND 25 HZ RATE

	OPEN AT ROLL FORWARD PATH			OPEN AT YAW FORWARD PATH		
FLIGHT CONDITION (SAMPLE FREQ.)	GAIN MARGIN (DB)	PHASE MARGIN (DEG)	FREQ. (RADIANS PER SEC)	GAIN MARGIN (DB)	PHASE MARGIN (DEG)	FREQ. (RADIANS PER SEC)
MACH .9 (25 Hz)	13.0	87	9.5 1.7	5.0	45	6.0 3.3
(12.5 Hz)	12.3	85	8.6 1.7	3.7	34	5. 3.3
MACH .7 (25 Hz)	11.9	93	9.8 1.8	5.9	53	6.2 2.8
(12.5 Hz)	11.2	90	8.9 1.7	4.8	45	5.3 2.8
MACH .5 (25 Hz)	11.8	91	10.2 1.9	8.0	52	6.2 2.3
(12.5 Hz)	11.1	88	9.4 1.9	6.9	45	5.2 2.3

This analysis was carried out without any changes to the system other than changing the filter coefficients to be compatible with the sample rate. The maximum gain margin loss is not excessive; however, it is felt that by designing the gains and filters as a function of the 12.5 Hz rate that the margins could be increased and the difference in margins between the two rates would be decreased.

The safest way to perform an in-flight demonstration of the reduced sample rate is to mechanize the required software in each primary GPC. The added software would run in parallel with the baseline system and switching from one path to the other would not require any re-initialization. In this manner, the system would remain quad redundant. The alternative of using the backup (or fifth) GPC would have the obvious disadvantages of being a single string system and needing a major primary software modification to provide the capability to switch back to the primary software, which is presently impossible during flight.

3.0 Requirements

The following is required to implement the experiment.

- Interface Definition - The FCS software could be initiated from the keyboard or more desireably by a discrete from a panel switch. After OFT one of the downmoding switches may be available.
- Software Requirements - The software changes needed for operating the Entry Flight Control at a reduced rate instead of 25 Hz are not very extensive. The requirements include:

  1. Logic to reduce the input rates of roll, pitch and yaw rates and lateral and normal accelerations (or hold those inputs over a number of 25 Hz samples) when the panel switch is set

to "experiment".

2. Logic to reduce the output rates of the elevon, rudder and aileron commands when the panel switch is set to "experiment".
3. Parallel modules for the bending filters and the compensating filters in the pitch and yaw channels must be added. These modules would be computed continuously at the reduced sample rate. Logic is required from the outputs of the nominal modules to the reduced rate modules when the panel switch is set to "experiment".

All modules of the normal system would function continuously and only those changes described above would be activated during the experiment. The reduction of input and output rates along with the reduced filter rates would cause the flight control to perform as if all applicable computations were being performed at the reduced rate. A list of the Entry FCS Functional Subsystem Software Requirements (FSSR) modules that need to be added in parallel are listed in Table A-2. Additional core requirements should be less than

TABLE A-2

REQUIRED PARALLEL MODULES

(ENTRY FCS FSSR NOMENCLATURE)

PITCH	YAW	ROLL
NZ_FILTER	YBB_FILTER	ABF
ELVCMD	RRCS_FEED	
FLERROR		

250 words.

- Analysis Requirements - Analysis will be performed to determine a recommended rate. Software design considerations will influence the choice of multiple or non-multiple increments of the present sample rate. Analysis will also be required to design the compensation and bending filters for the selected rate. It is estimated one man-year will be required.
- Verification - Complete closed-loop testing would be required on the SAIL to fully verify the primary to experiment to primary switching.
- Cost - Based on the above software requirements, a cost of $125K was estimated for the additions and change to the logic to reduce the sample rate. An increase to $500 per word was assumed, since changes to the executive program would be required. One man-year of analysis costs $60K.

4.0 Mission Impact

No major impact on the mission is required. Crew will switch to the slow rate flight control and compare to the nominal system. With the proposed mechanization, the system would operate in a normal fashion with regard to redundancy management.

EXPERIMENT D: FREE DRIFT ATTITUDE MODES EXPERIMENT

1.0 Background and Objectives

The "Free Drift Attitude Modes" (FDAM) experiment was suggested by Rockwell International/Space Division. It evolved from their work performed under Contracts NAS5-23203 and NAS2-9231. FDAMs offer the potential of extended time periods (20 minutes to many hours) without Reaction Control System (RCS) firings relative to the standard Orbiter attitude control system. They can be achieved at a trivial expense relative to other alternatives such as control moment gyros. The desirability for the performance extension afforded by these modes is to satisfy the RCS contamination and zero "g" level requirements of numerous sortie mission payloads.

FDAMs achieve along time periods without RCS firings through use of control policies which more clearly recognize the ambient disturbance environment (predominantly gravity gradient torques) and its effect on spacecraft attitude. The FDAMs consist of six gravity gradient (GG) (local vertical/horizontal) orientations and three quasi-inertial (QI) orientations. Of the GG orientations, there are three that appear to be most useful from the standpoint of payload bay viewing freedom. One of these is stable providing very long drift times, whereas the other two are unstable with drift times on the order of one-half hour.

The QI modes provide an approximate inertial orientation but librate through small angles in response to the cyclical gravity gradient torques. They are advantageous for inertially pointed payloads and offer drift times in excess of half an hour.

A FDAM experiment performed with the Shuttle Orbiter during the OFT missions will provide the following:

- It will improve confidence in the use of these modes. No flight applications of the unstable GG and QI FDAMs are known to exist.
- It will provide the FDAM design and operational experience in order to define a better operational flight control system than would be possible without this experience.
- It will provide better estimates of uncertain parameters such as principal axis location variations and uncertainties, crew motion disturbances, earth magnetic disturbance torques, etc., which are required to achieve desired performance levels in an operational system implementation.

2.0 Feasibility

The key to successful FDAM operation is the management of the disturbance torque sources. Typical techniques include: managing ventings (such as scheduling flash evaporator operation) so that they do not occur during free drift periods, selecting orbit altitude to minimize aerodynamic torque effects, and applying constraints to crew and equipment motion. The magnetic torques are expected to be very small. The gravity gradient torque is relatively large (maximum of approximately 10 ft-lb) and hence the FDAM's must use selected attitude orientations. The gravity gradient free-drift modes minimize the torque by flying at gravity gradient torque nulls; e.g., principal axes horizontal. The quasi-inertial FDAM's result in an approximate inertial orientation and experience attitude libration due to the cylical gravity gradient torques.

The six possible gravity gradient attitudes are illustrated in Figure A-13. The various orientations are designated as stable (S) or unstable

GRAVITY GRADIENT ATTITUDES AND THEIR STABILITY

ATTITUDE		GRAVITY GRADIENT STABILITY		
AXIS P.O.P.	AXIS II TO NADIR	IN-PLANE	OUT-OF-PLANE	BOTH
X	Y	S	S	S
	Z	U	U	U
Y	X	S	U	U
	Z	U	U	U
Z	X	S	S	S
	Y	U	S	U

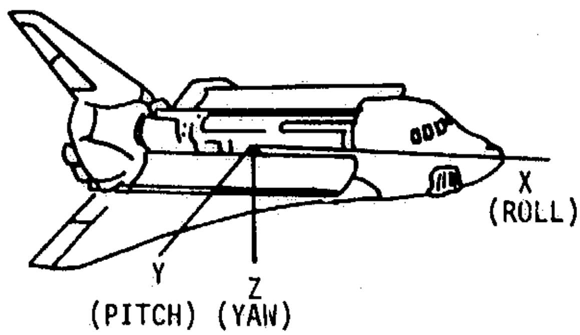

FIGURE A-13

(U) depending on whether the restoring torques are positive or negative. The two attitudes in the dashed boxes have stable gravity gradient torques in both axes.

A three-axis simulation study was performed under NAS 5-23203 by Rockewll to define the performance of the FDAM's. A pitch-yaw-roll Euler angle order as shown in Figure A-14 was employed in all the simulations runs presented except for the X-POP case which utilizes a pitch-roll-yaw sequence. All simulation runs presented show the effects of gravity gradient torques and neglect other disturbances (such as aerodynamic torques, ventings, crew motions, etc.). The runs are for initial attitude and rate errors of one degree and 0.001 degree/second respectively, taken in the worst sense. The one degree error on attitude is due primarily to the error

EULER ANGLE CONVENTION AND AXIS SYSTEM

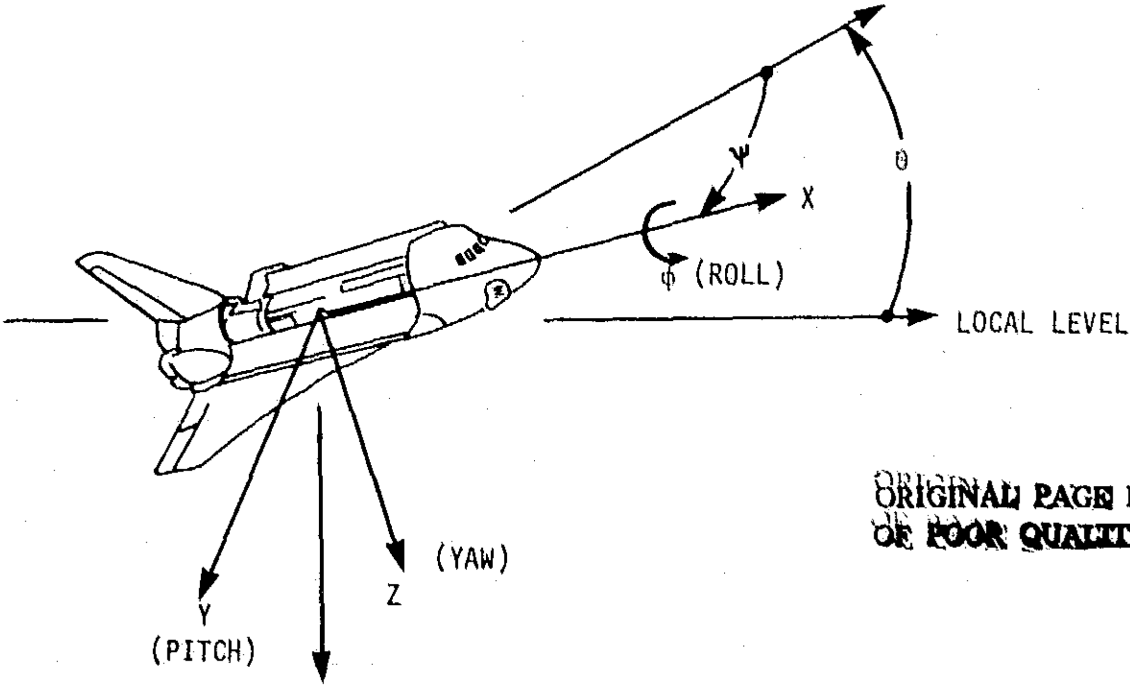

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FIGURE A-14

in knowledge of the orientation of the principal axes of inertia. The rate error is approximately the minimum impulse capability of the vernier RCS. These flight conditions are thought to be achievable for flight above approximately 200 n.mi. of altitude.

Figure A-15 shows the simulation results for the Y POP, X nadir gravity gradient orientation. This initial orientation has positive restoring torques in pitch and yaw. Therefore, the pitch and yaw attitude angle response to the initial conditions is cyclical and limited to an amplitude of approximately ± 1 degree. The roll axis diverges due to the initial conditions and will limit the free drift period.

Simulation results for the unstable gravity gradient Y POP, Z nadir ("airplane"

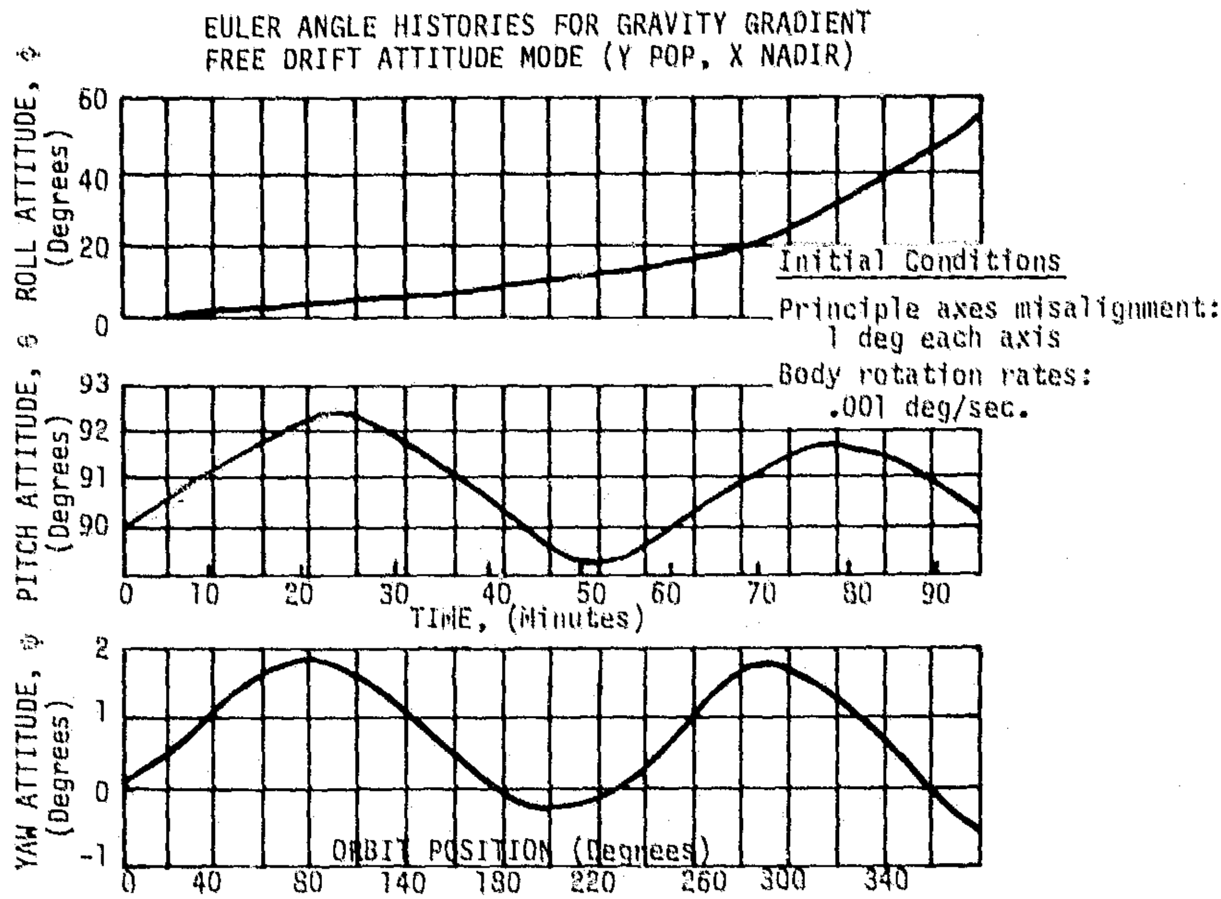

FIGURE A-15

mode) oriention are shown in Figure A-16. This attitude provides favorable viewing from the payload bay for many applications. It is aerodynamically stable and relatively unaffected by low-altitude operations. This initial orientation results in negative restoring moments. The pitch and roll attitude may be seen to diverge more rapidly than the other cases presented. However, enough free drift time is available for many applications.

Figure A-17 shows phase-plane trajectories for the Y-POP orientation which is useful in defining the initial conditions necessary for the Quasi-Inertial (QI) FDAM. It illustrates the strong effect of the gravity gradient torques. In the absence of the gravity gradient torques, the trajectories would

EULER ANGLE HISTORIES FOR GRAVITY GRADIENT FREE DRIFT ATTITUDE MODE (Y POP, Z Nadir)

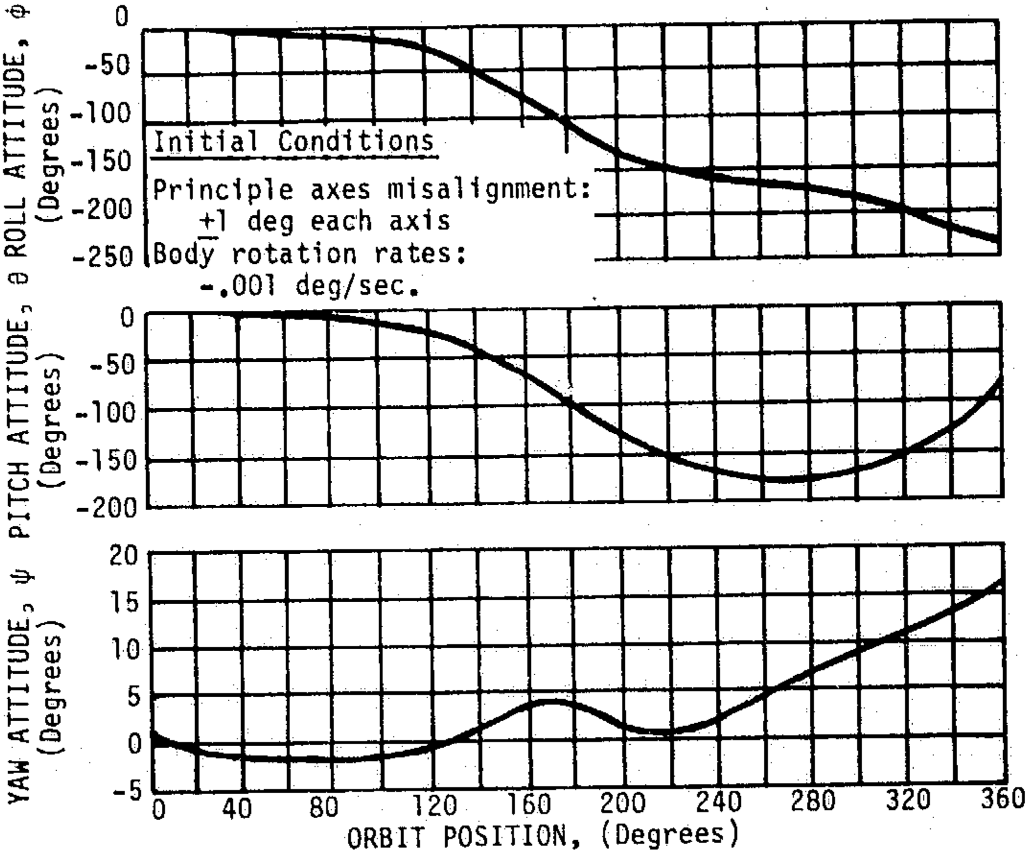

FIGURE A-16

FREE DRIFT PHASE PLANE TRAJECTORIES (Y-POP)

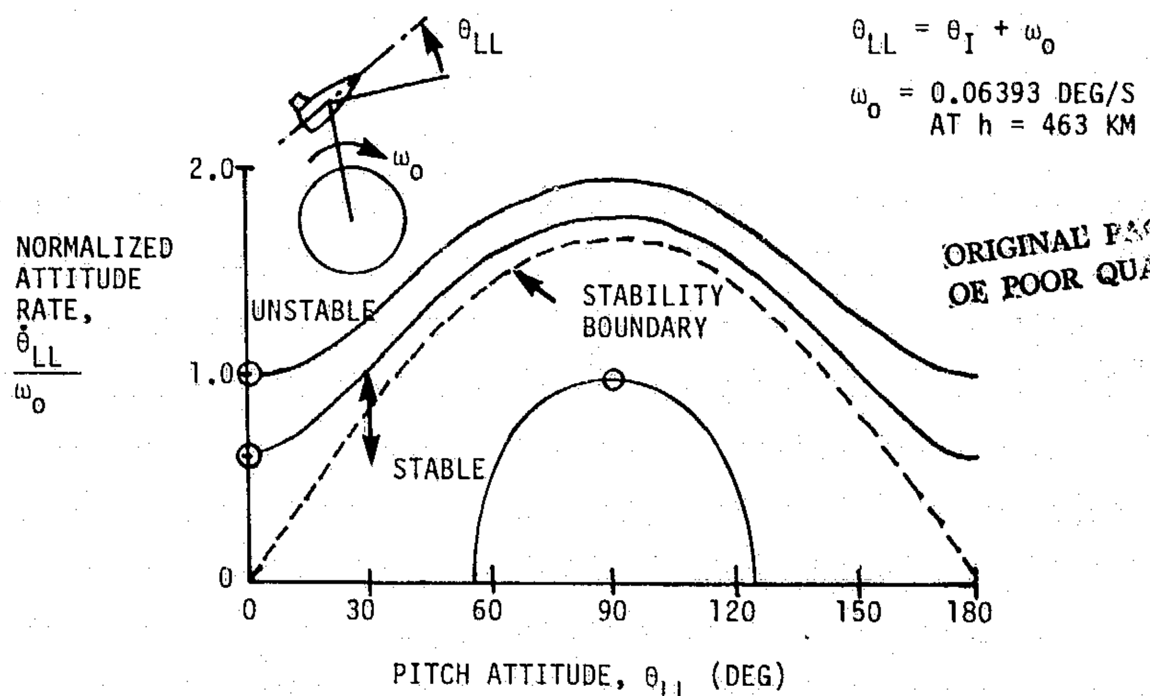

FIGURE A-17

simply be horizontal lines. For initial conditions below the dashed line, the gravity gradient torques "capture" the vehicle and it will librate about the 90-degree point. For initial conditions above the dashed line, the behavior is secular with respect to local level coordinates. The first curve above this dashed line is the preferred trajectory for the QI mode. A simulation run illustrating the attitude behavior in local-level and inertial coordinates is presented in Figures A-18 and A-19, respectively. It may be seen that the inertial pitch librations are approximately +18 degrees. This value is acceptable for most gimbal-mounted payload instruments.

EULER ANGLE HISTORIES FOR QUASI-INERTIAL FREE DRIFT MODE (Y-POP)

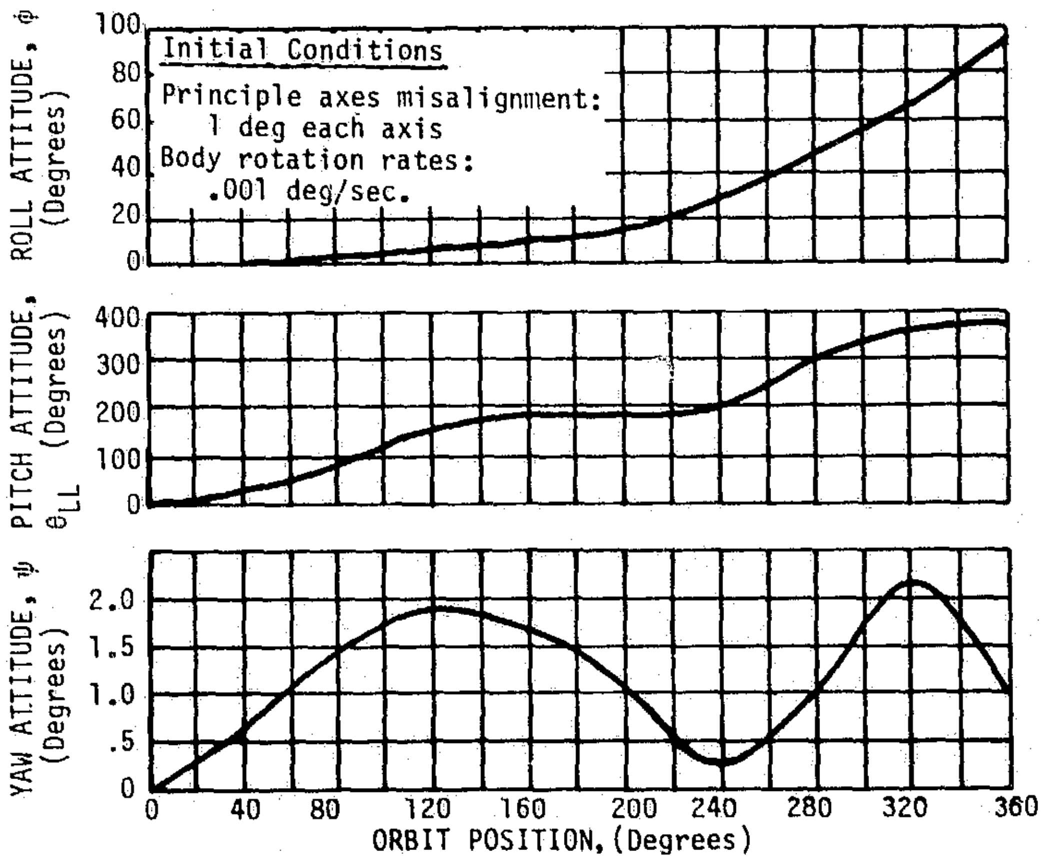

FIGURE A-18

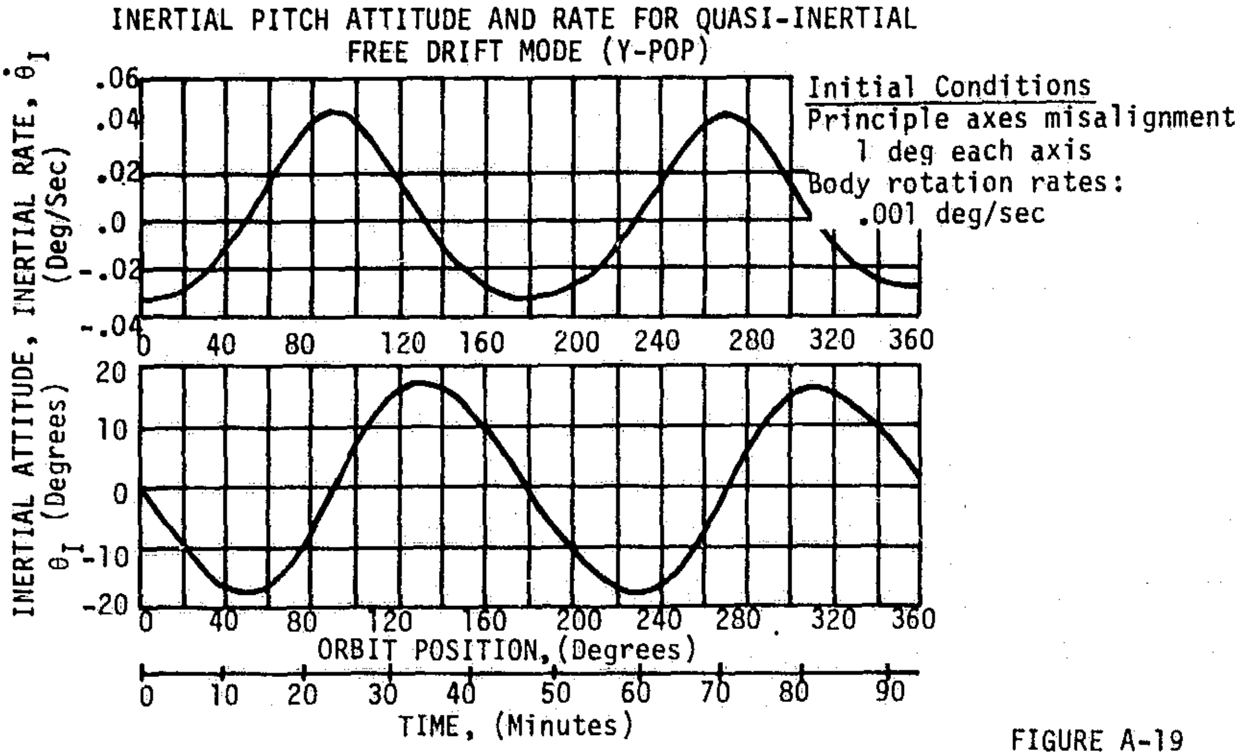

FIGURE A-19

A tabular summary of the simulation results is presented in Table A-3. The attitudes given are those that appear to have the most potential applicability due to the favorable payload bay viewing directions. The available drift times for attitude limits of 10 and 20 degrees are given. These drifts times exceed that provided by normal vernier RCS operation by approximately an order of magnitude. The stability gradient attitude provides the longest drift time and the unstable attitudes result in the shortest drift times.

It is noteworthy that the misalignment of the principal axes of inertia has a strong effect on the drift time available in the unstable gravity gradient and quasi-inertial modes. The use of a statistical technique to estimate the misalignment parameters offers promise for significant

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TABLE A-3

SUMMARY OF FREE DRIFT SIMULATION RESULTS

MODE	INITIAL ATTITUDE*		DRIFT TIME (MIN) TO EXCEED:		AXIS OF DIVERGENCE
	AXIS P.O.P	AXIS// TO NADIR	10°	20°	
GRAVITY GRADIENT	Y	X	50	66	ROLL
	Y	Z	24	31	PITCH
	X	Z	39	45	YAW
QUASI-INERTIAL	Y	Z	50	59	ROLL
	X	Z	29	38	YAW

*PRINCIPAL AXIS MISALIGNMENT = $\pm$1.0 DEG, BODY RATES = $\pm$0.001 DEG/SEC., WORST CASE SENSE

drift time performance improvements. Further investigation of this approach is recommended.

The preliminary analysis of FDAM's has indicated their feasibility. They are attractive because of the low implementation cost and because of their utility in many mission applications. Drift times in excess of those indicated here may be feasible through use of a principal axis orientation estimation technique.

3.0 Requirements

It is suggested that five of the more desirable gravity gradient attitudes and two quasi-inertial cases be flight tested. This would require a minimum test time of approximately 17 hours which would include repeating some of the tests based on real-time ground analysis of the data and arriving at a new estimate of the principal axis orientation.

No new software would be required, and the only hardware penalty would the propellant cost to reinitialize the vehicle attitude between tests. This would amount to approximately 33 pounds of propellant for each individual test. All experiment data will be derived from the existing Orbiter reference system.

It is anticipated no special man-in-the-loop simulations would be required to establish the feasibility of the FDAM's. Additional runs from an off-line program would be warranted to establish the time history characteristics resulting from various uncertainties. Sensitivity to principal axis mis-alignment could be established from this data and used by the ground support people for real time updating of the vehicle orientaion for test reruns.

A typical FDAM experiment cycle is shown in Figure A-20. The initial attitude alignment of the Orbiter will be accomplished by the crew. The attitude control deadband will then be opened to a large value, and the Orbiter will drift for periods ranging from approximately 20 to 90 minutes. Attitude

TYPICAL FDAM EXPERIMENT OPERATION

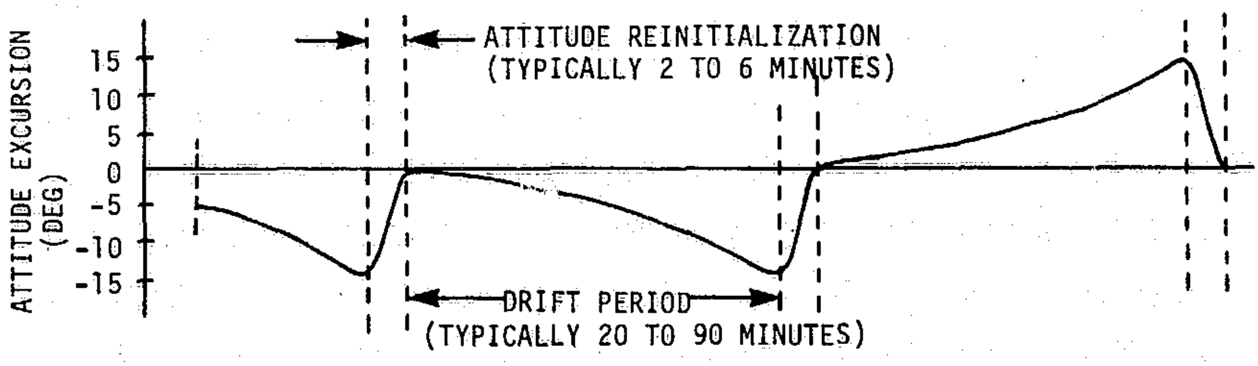

FIGURE A-20

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excursions in the order of 10 to 15 degrees are desired, but adequate data can be taken for excursions of 3 to 5 degrees. At the end of the drift period, the attitude control systen is reactivated and the attitude state reinitialized for the next test cycle.

After a drift cycle, the attitude divergence data can be analyzed at the Payload Operations or Mission Control Center, and new estimates can be made for the FDAM bias parameters (principal axis orientation). The new bias data could be uplinked to the Orbiter for drift time performance improvement during subsequent test cycles.

4.0 Mission Impact

During most of the drift tests, it is desirable that the crew motion be relatively restrained and that ventings be scheduled to occur outside the drift periods. In order to investigate sensitivity to small disturbances, controlled tests will be conducted with prescribed crew motion exercises and venting.

By flying the experiment on several of the OFT missions, it can be accomplished with a minimum of interference to any of the presently planned flight tests.

EXPERIMENT E: ORBITER FLYING QUALITIES AND FLIGHT CONTROL SYSTEM PERFORMANCE

1.0 Background and Objectives

This experiment was proposed by both Langley and Ames Research Centers. The determination of Orbiter flying qualities and flight system performance is a natural follow-on to the mainline OFT development effort. Pilot comments will be combined with analytical determination of vehicle transfer functions to provide a complete Orbiter flying qualities data base. The data base will provide historical documentation for future programs and the basis for orderly expansion of the Orbiter flight envelope.

The objective of this experiment is to combine in one document the pilot comments and the flight control system performance relevant to Orbiter flying qualities.

2.0 Feasibility

Collection of pilot comments is a relatively straightforward task. The quality of the comments gathered can be improved by preparing a briefing on the experiment objectives and by preparation of a standard form for the Orbiter commander and pilot to complete at the conclusion of the flight.

Determination of the Orbiter transfer functions is relatively more involved. In general, there are three flight techniques from which the transfer functions can be obtained. They are the steady flight technique, the transient response technique, and the sinusoidal oscillation technique.

- Steady Flight Technique - The steady flight technique can be subdivided into steady-straight and steady-turning flight. In general, the steady-straight technique can be used for the pitch axis and the steady-turning for the roll/yaw axis. By stabilizing the Orbiter in straight flight at different airspeeds and at different center of gravity locations, and then measuring the elevator required for trim, it is possible to obtain numerical values for certain combinations of transfer function coefficients which represent the steady state. The explicit value of each steady state numerator/denominator coefficient cannot be obtained separately unless the values of one of the coefficients can be assumed or estimated from a different flight test technique.

In the roll/yaw axes, the aileron and rudder subsonic, and the aileron and RCS supersonic derivative terms are intimately coupled through the Orbiter flight control system. It is difficult to separate, for instance, the rolling moment due to rudder from the rolling moment due to aileron. The Orbiter has a unique feature which should allow the effects to be separated. The base design of the OV-102 includes downmoding switches in the cockpit. With each pitch or roll/yaw three-way switch position, there is an associated set of forward loop gains. Following completion of OFT, these switch positions and their corresponding S/W I-load values will not be required. It would be possible to use these switches to achieve isolation of the rudder and aileron channels for purposes of this experiment.

- Sinusoidal Oscillation Technique - The sinusoidal oscillation technique is the most elaborate method of establishing the transfer function of the vehicle. The sinusoidal oscillation technique consists of injecting a sinusoidal control input into the control system and measuring the corresponding phase and magnitude of the resulting body rates and accelerations. The Orbiter through the Program Test Input (PTI) specialist function has this capability design in and it will be available to support this experiment. Figure A-21 provides a summary of the PTI specialist function.

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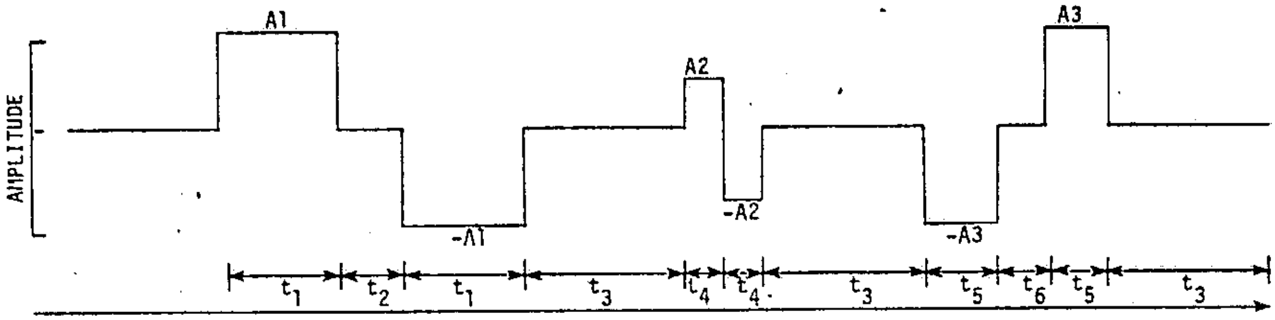
ENTRY OFT FTR PTI FORMAT

TYPICAL PTI SEQUENCE (1 TEST SEQUENCE OF 3 DOUBLETS SHOWN AS AN EXAMPLE):



FOR EACH FLIGHT -

- A STRING OF 14 DOUBLETS IS LOADED. EACH DOUBLET HAS ASSOCIATED WITH IT:
  - AN AXIS (PITCH, ROLL, YAW)
  - AN AMPLITUDE (A1, A2, A3 ABOVE); ONE SCALAR NUMBER
  - A PULSE TIME DURATION (t_1, t_4, t_5 ABOVE); INTEGER MULTIPLE OF 80 MSECS
  - A PULSE TIME GAP (t_2, t_6 ABOVE); INTEGER MULTIPLE OF 80 MSECS
- 8 TEST SEQUENCES ARE LOADED. EACH SEQUENCE HAS ASSOCIATED WITH IT:
  - A STARTING DOUBLET FROM THE STRING OF 14, AND AN ENDING DOUBLET FROM THE STRING OF 14 (I.E., EACH OF 8 TEST SEQUENCES COULD SPECIFY PERFORMANCE OF DOUBLETS 1 THROUGH 14, OR SEQUENCE 1 COULD SPECIFY DOUBLETS 2 THROUGH 5, SEQUENCE 2 COULD SPECIFY DOUBLETS 4 THROUGH 7, ETC.)
  - A TIME GAP BETWEEN DOUBLETS (t_3 ABOVE); INTEGER MULTIPLE OF 80 MSECS

FIGURE A-21

The advantage of the sinusoidal oscillation technique in establishing transfer functions is that it gives fairly accurate results over a wide range of frequencies. It is useful in certain frequency ranges where the exact form of the transfer function is in question. It can also be useful in establishing the existence of unsteady flow phenomena (flutter) or in correlating theoretical predictions of unsteady flow phenomena.

The disadvantage of the sinusoidal oscillation technique is that it requires much more flight testing time than the transient technique because the airplane must be stabilized at each value of input frequency, and many stabilized points are required to define the complete transfer function of the airframe over the frequency range of interest.

• <u>Transient Response Technique</u> - The transient resonse technique is the most widely used method for determining both the stability derivatives and the transfer functions of the airframe. In this technique, some measurable input is applied to the airframe, and transfer function coefficients are determined from the resulting transient response data.

The advantages of the transient technique over the sinusoidal flight testing technique are that much less flight time is required and that coefficients can be obtained directly from the transient data without performing a frequency response analysis.

The disadvantage of the transient technique is that the form of the equations of motion of the airframe must be known or assumed when transfer functions are to be derived directly. The problem then usually arises as to whether or not higher order derivatives, unsteady flow effects, actuator non-linearities, and aeroelastic modes should be included.

Once transient flight data are available, there are four general methods of extracting information from them. They are:

•• Transient Inspection Method

•• Response curve fitting method

•• Fourier transform method

•• Simulation matching method

The transient inspection method consists of using the short period approximations of the airframe and matching period, damping, and steady state gain to the airframe response by inspection. Because of the restrictive assumptions and approximations that have to be made to get transfer function coefficients by this method, the results can be relied upon only to the extent that the assumptions and approximations have been established as applicable to a given case. The chief advantage of this method is that it permits the order of magnitude of certain transfer functions to be established quickly and easily without resorting to complex analytic procedures. This technique was used after ALT free flight 1 to determine if it was necessary to change any control system gains before the next free flight.

The response curve-fitting method consists basically of matching transient responses by assuming the form of the equations of motion of the system to be known; the unknown coefficients of the various terms in the equation are then evaluated to match transient data as well as possible. Various mathematical techniques have been devised to get transfer functions directly from transient flight data and literature on the subject is very extensive. Nearly all these techniques are iterative in nature and utilize some form of least squares criterion, Newton-Raphson steepest descent method, or Kalman filter to fit the theoretical motion to the flight test data. Their success is to some extent dependent upon flight regime (subsonic, transonic, supersonic), upon how accurately the values of stability derivatives are known prior to flight test, and indeed upon the experience of the person performing the analysis.

In the Fourier transform method, the transient response flight data are converted mathematically to frequency response form by application of the Fourier integral. These frequency response data can be analyzed for values of the transfer function coefficients. This technique has the advantage that the form of the equations of the motion need not be assumed as it does for the curve fitting methods. Disadvantages are that the size and shape of the input used in the flight test, and the downlist data rate have a considerable influence on the accuracy with which the frequency response can be derived. For these reasons, this technique was used as a check during ALT to determine the lower frequency components of the transfer function.

The simulation matching method consists basically of modifying by a trial-and-error procedure the aerodynamic data base in a simulation to match the transient response data as well as possible. Once the match is obtained, the transfer function associated with the airframe can then be determined. This technique has the advantage that all of the stability derivatives and the corresponding transfer functions can be obtained; it is also useful in refining stability derivatives obtained by any other flight test technique. The success of this technique depends upon how accurately the aerodynamic data is known prior to flight test and on the experience of the person performing the analysis.

In summary, the experiment is feasible and desirable from a technical standpoint. The combining of vehicle transfer functions, flight control system definition, and pilot flying qualities comments in one document, while somewhat novel, is an excellent idea in that it combines in one place all the information necessary for future designers to make extrapolations.

3.0 Requirements

The primary requirement for this experiment is orbiter flight data. Data from the Shuttle telemetry system is required for both pitch and roll/yaw transfer function determination. Data from the Shuttle Entry Air Data

System (SEADS) is required throughout the entry to landing phases for roll/yaw transfer function determination. In addition it is also required for supersonic pitch axis transfer function determination. Data from the Aerodynamic Coefficient Identification Package (ACIP), while not mandatory, would enhance the quality of the transfer function determination process.

3.1 Interface Definition

No experiment specific interfaces are required for this experiment. Interfaces with the onboard flight software are required for specification of PTI inputs and the forward loop gain definitions associated with the downmode switches. This does not represent a unique interface, only a redefinition of existing Shuttle I-Load constants. Experiment objectives can be accomplished without specifying these constants; however data quality would be enhanced with their specification.

3.2 Software Requirements

Software requirements can be divided into three types for this experiment. These are:

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- Shuttle Flight Software
- Data Collection Software
- Transfer function determination software

Shuttle flight software code modifications to shuttle I-Loads, which are optional, were addressed under interface requirements.

Data collection software is required for three elements. These are the SEADS, the ACIP, and the OV-102 Telemetry Downlink Data. In each case data processing and reformatting is required for interfacing with the transfer function determination programs. This process may be somewhat involved as tape conversion from the JSC Univac computers for usage on the LARC Cyber (CDC) computer may require extensive effort. As an example of the complexity, the aerodynamic data tapes for the Shuttle are available on the JSC Univac System. To obtain aero data in a format suitable for use in the Terminal Area Control Simulation, which is run on a Cyber 73 at JSC, the following steps must be taken. A job is run on the Univac system to convert the data tape to an E format that can be read by other than 1108 hardware; the output tape is then taken to a JSC CDC 6400 computer. Another program is then run to format the data for use on the CYBER computer. This data collection process is involved and potentially expensive for this experiment because data formats for the SEADS have not been established.

The transfer function determination program will be developed by the prime investigators. The program requirements are a function of the technique used.

3.3 Hardware Requirements

No experiment specific hardware is required. Availability of the SEADS and ACIP packages does constrain experiment performance. If the SEADS is unavailable, pitch axis transfer functions can be determined below Mach 2.5 only. It is the only source of accurate angle of attack above this Mach number. In addition, the SEADS is the only source of sideslip angle for roll/yaw transfer function determination.

3.4 Simulation Requirements

Offline simulation is required to determine the magnitude and frequency of the PTI inputs required for transfer function extraction. In addition, off-line analysis programs and man-in-the-loop simulations must be run to assess the safety aspects of using the downmode discretes to set the forward loop gains in the roll or yaw axis to zero.

Man-in-the-loop simulations (SMS and SAIL) are required to address the mission timelines and mission performance aspects of the experiment.

3.5 Cost

Costs are somewhat subjective for this experiment because of the possible options available. No costs are projected for utilization of the SMS and SAIL facilities as their cost is assumed to be absorbed by the Shuttle Program in the normal mission development. Similarity, use of the downmode discrete is an option. The breakdown of costs is:

Data acquisition -	50K
Data reduction and analysis -	100K
Downmode discrete option -	25K
	$175K

4.0 Impact of Experiments on Mission

Timeline development for using the PTI function is required. The crew interface is through the onboard CRT and relatively simple keystroke actions are required to execute the PTI function. The downmoding discretes are keyed with the throwing of three switches. The safety aspects of using the downmode discretes, as previously stated, must be evaluated.

EXPERIMENT F: COMPARISON OF AERO DATA EXTRACTION TECHNIQUES

1.0 Background and Objective:

Proposed experiments in which vehicle motion during entry will be utilized for extraction of aerodynamic stability and control derivatives have been received which employ three different aerodynamic parameter identification programs. These are:

(1) Advanced Modified Maximum Likelihood Method (MMLE) - NASA DFRC,

(2) Maximum Likelihood System Identification Program (MLSIP) - MDTSCO, and

(3) Pseudo Complete Math Model - NASA Ames

Since each of these programs requires essentially the same input data, i.e., vehicle motion, control surface activity and vehicle state information, it appears desirable to use a common input data base and have each experimenter independently arrive at his best estimate of the aero parameter under consideration with associated uncertainty. For compatibility with the purpose of OFT placard removal within the fast turnaround requirements between flights, the mainline Shuttle program will utilize the basic 3-DOF linear MMLE program for the majority of its analysis. The 6-DOF non-linear MLSIP program will be applied to special situations. For the proposed experiments, in order to assure the highest quality in state information and to preclude interference with the mainline effort during OFT, it appears desirable to await incorporation of the Shuttle Environmental Air Data System (SEADS). In addition, in order to obtain the highest quality vehicle motion, the Aerodynamic Coefficient Identification Package (ACIP) should be retained from OFT. The mainline program data analysis (DAP) can be used for generation of a common data base for analysis.

The overall objective of this experiment will be to better define and improve the state of the art of the aero extraction capabilities by allowing for direct comparisons of results from various techniques. Analysis in flight regions where the data are more likely to be known can be contrasted to areas where significant uncertainty is anticipated. Analysis of simple as well as complex motion can be utilized to support uncovering deficiencies in the programs. The results of this experiment can be used to support expansion of the Orbiter operational capabilities as a result of more accurate stability and control derivatives as well as improve the aero extraction state of the art for application to future vehicle designs.

2.0 Feasibility

This experiment is a logical follow-on to the type of analysis performed on OFT which will essentially verify the basic ingredients required for this study. This experiment will emphasize research-oriented studies rather than the day-by-day engineering type of studies required by OFT. The advanced MMLE program to be used on this experiment is an expanded version over that to be used for engineering purposes on OFT. The feasibility and desirability of evaluating the advanced MMLE program can best be summarized by referring to the DFRC proposal.

> "During the past decade, stability and control characteristics have been derived from flight tests by means of a modified maximum likelihood method developed at the NASA Dryden Flight Research Center. Over 3000 maneuvers have been successfully analyzed with this method for twenty-five different aircraft tested at the Center as well as many other aircraft tested by various aircraft companies and other government agencies. These aircraft range from lifting bodies to several large transports, including a large supersonic bomber. The Shuttle is

expected to differ from the earlier applications principally in the type of maneuvering required during entry, the influence of control augmented damping, the transient nature of the flight test conditions, and the degree of coupling between the structural and aerodynamic modes. It is anticipated that the additional complexities introduced in the reentry environment can be better handled by a more generalized recently developed version of the method currently used for most aircraft. In particular, allowances are made for rapid variations of velocity and dynamic pressure during maneuvers and for structural/aerodynamic mode coupling. The generalized method has been partially verified on the basis of simulated data and is now in a test phase on several aircraft, including the B-1. The possible lack of precise air data measurements at hypersonic speeds may present some difficulty in reducing the flight data to dimensionless coefficients obtained from wind tunnel tests and analytic studies. Also, if the motion during maneuvers during flight are greatly restricted, the usefullness of the results will be significantly reduced."

It is also noteworthy that DFRC will supply manpower and facilities from in-house resources.

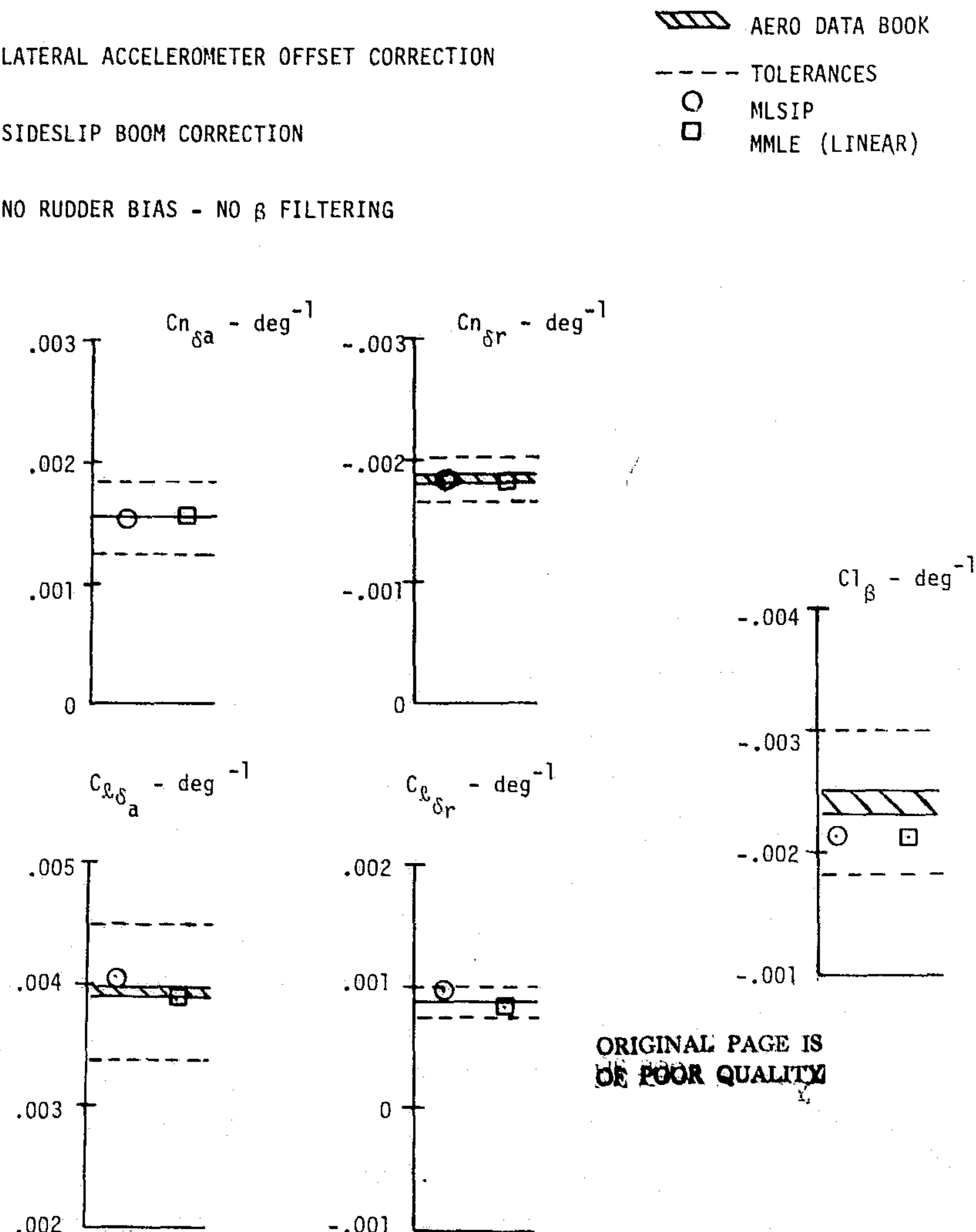
The MDTSCO nonlinear and 6 DOF MLSIP program has been applied to the F-4 at high angles of attack and the Orbiter 101 briefly during ALT. Figures A-22 through A-25 present MSLIP results for a lateral-directional maneuver during ALT. Note the prediction of nonlinear $C_{n\beta}$ and identification of the time skew in the data from the match of sideslip angle. Since ALT, MLSIP has been expanded to accommodate RCS firings and changes in the stability and control derivatives with rapidly decreasing Mach number. This experiment

AERO DATA EXTRACTION COMPARISON
ALT FREE FLIGHT 2 RESULTS

LATERAL ACCELEROMETER OFFSET CORRECTION

SIDESLIP BOOM CORRECTION

NO RUDDER BIAS - NO β FILTERING



ORIGINAL PAGE IS
OF POOR QUALITY

FIGURE A-22

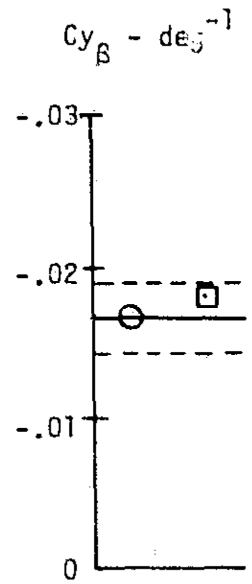
AERO DATA EXTRACTION COMPARISON
ALT FREE FLIGHT 2 RESULTS

AERO DATA BOOK
- - - - TOLERANCES
○ MLSIP
□ MMLE (LINEAR)

SIDESLIP BOOM CORRECTION

NO RUDDER BIAS - NO β FILTERING

LATERAL ACCELEROMETER OFFSET CORRECTION



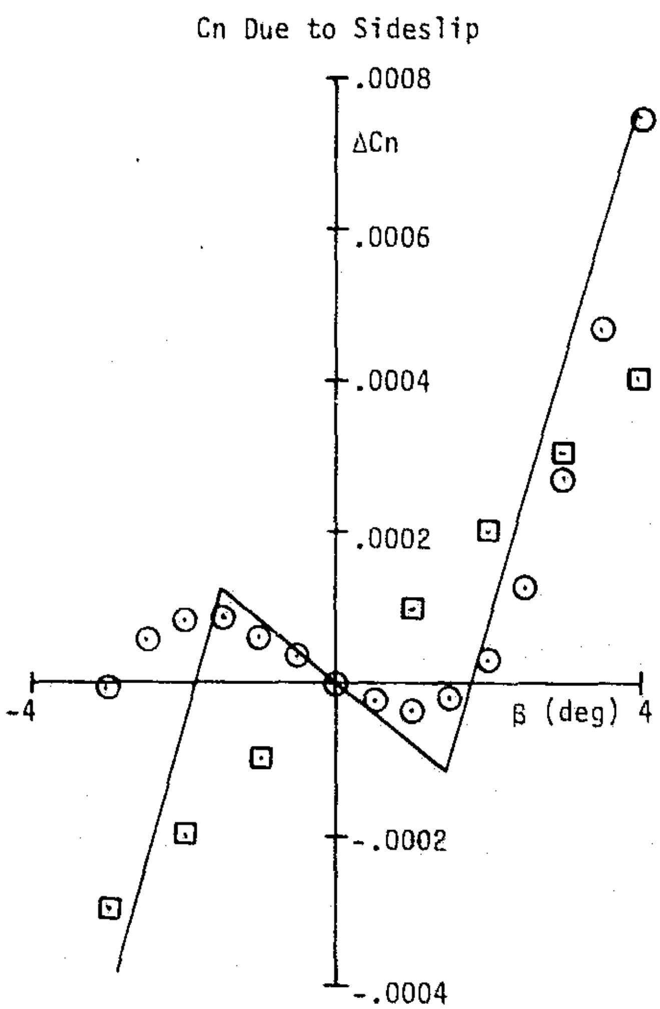
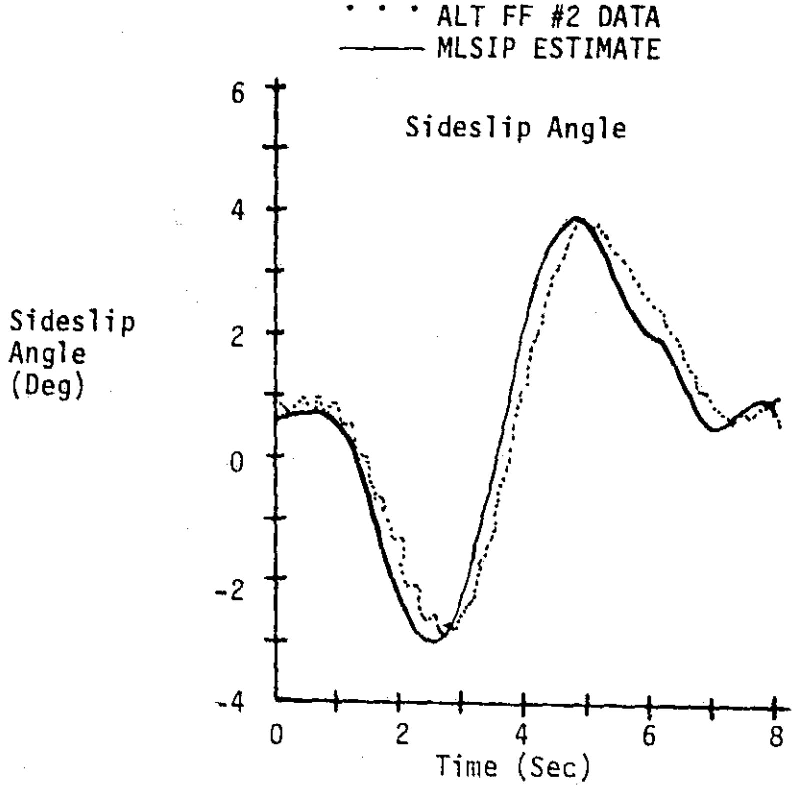


FIGURE A-23

AERO DATA EXTRACTION MOTION MATCHES



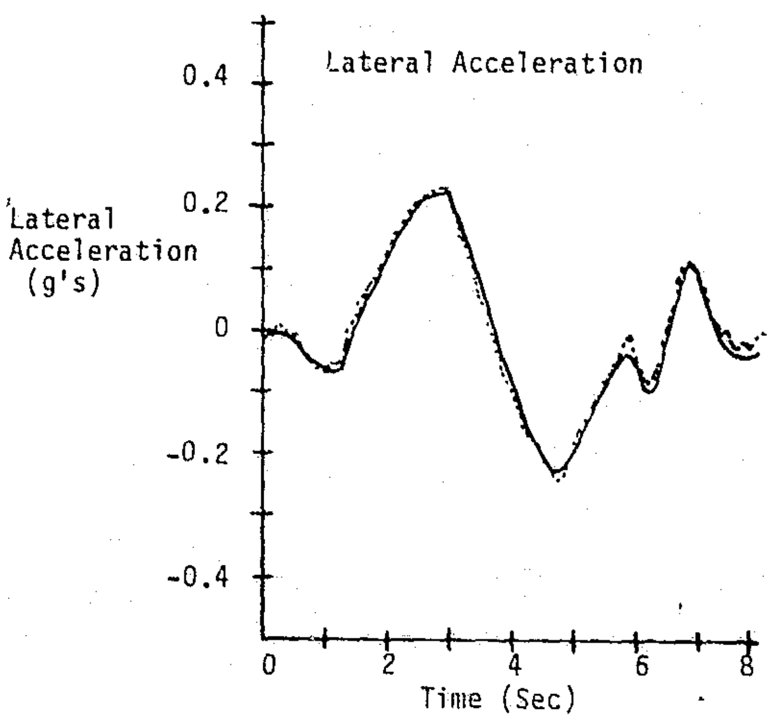


FIGURE A-24

AERO DATA EXTRACTION MOTION MATCHES

· · · ALT FF #2 DATA

——— MLSIP ESTIMATE

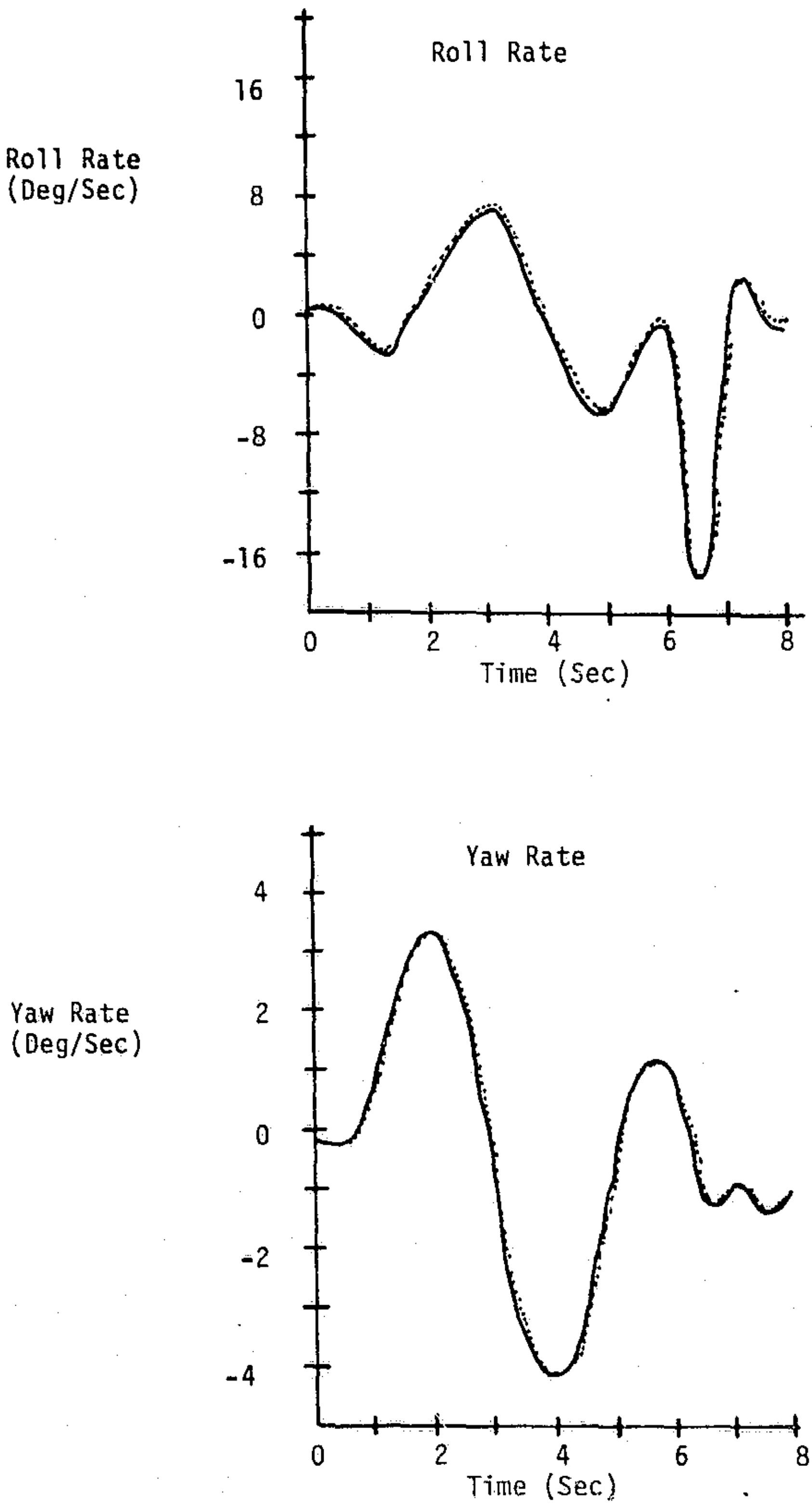


FIGURE A-25

will allow for a thorough evaluation of the capability of MLSIP with respect to other techniques for complex extraction situations as well as the more straightforward scenarios.

The advantages and desirability of including the AMES program in this experiment can be amply described by referring to their proposal.

"There are several research groups now applying parameter identification for a variety of aerospace vehicles. The standard techniques use simplified (linear) math models and perform analyses on a very short time history (≈10 sec) of the transient response, resulting from a control pulse input. In recent work at Ames, we have applied parameter identification incorporating flight data from normal flight maneuvering (such as may be available from the shuttle orbiter). This technique uses a pseudo-complete (non-linear) math model allowing analysis over a much longer length of data, thus providing a significant improvement in the accuracy of the identified parameters. This analysis typically has used about 5 minutes of aircraft flight data during which several types of maneuvers were performed such as control pulses, climbs, dives, turns, etc. This analysis of long data lengths, combining both static and dynamic conditions was found to minimize many of the problems that are usually associated with parameter identification from flight data. For instance, the use of long data lengths was found to minimize the scatter (i.e., variance) in the parameter estimates. The combined use of both steady-state and dynamic portions of data were found to minimize the problems of estimating those parameters which are inherently difficult to separate (i.e., parameter identifiability). Also, the use of a rather complete math model was found to minimize some of the bias errors associated with the more standard, simplified math models (i.e., modelling errors)."

3.0 Requirements

The requirements for this experiment include SEADS, ACIP, specific maneuvers at given flight conditions, flight measured-to-analysis data processing capability, and the generation of a common data base for the experimenters. SEADS and ACIP were originally conceived with this type of experiment in mind. The data handling and manipulation techniques (DAP) being developed for OFT by the NASA JSC Engineering Analysis Division (EX) should suffice with minor changes for this experiment.

Programmed test inputs (PTI's) should suffice for the generation of the common maneuver for analysis. It is suggested that this experiment be used as a predecessor to other ADE experiments as illustrated in Figure 1, thus assuring valid and efficient extraction capability and system capability prior to addressing the other more expansive investigations.

Planning, analysis and documentation costs for this experiment will vary with the organization performing the study as well as the number of flights and maneuvers per flight. Hardware and software costs for this experiment appear to be nil by utilizing in-place items such as SEADS, ACIP, and DAP. Based upon a three (3) flight program it is estimated that this experiment will require 1,200 manhours per contractor.

4.0 Mission Impact

This experiment could have impact with respect to the mission as a result of RCS propellant requirements. These requirements can be kept at a minimal by selecting maneuvers in regions where the expenditure would be minimal. It is not the objective of this experiment to evaluate parameters over the entire flight regime and thus considerable flexibility is available for minimizing the propellant requirements. Current MDTSCO analysis of SPS and SSFS simulations indicate typical RCS propellant requirements as follows:

• Pitch Doublet	$q < 20$	60 lbs.
• Roll Doublet	$M = 5.5$	120 lbs.
• Roll Doublet	$M = 4.8$	100 lbs.
• Roll Doublet	$M = 2.3$	30 lbs.
• Roll Doublet	$M = 1.4$	0 lbs.
• Yaw-Roll Doublets	$M = 2.3$	60 lbs.
• Yaw-Roll Doublets	$M = 1.4$	20 lbs.

Current studies indicate that 10-15 seconds of crew time will be required for a given maneuver. At this time no conflicts are apparent, and the crew time demands seem reasonble. Although effects of guidance interruptions required to perform the maneuver must be addressed on an individual maneuver basis, the Shuttle Procedure Simulator (SPS) engineering simulations did not reveal any problem for maneuvers above $M = 1$. Safety aspects are similar to those for the baseline system. For this experiment, there is adequate flexibility to select flight regions with the greatest safety margins.

EXPERIMENT G: EVALUATION OF ADE MANEUVER FORMATS

1.0 Background and Objective

During the major portion of Orbiter entry, the flight control system utilizes RCS effectors, elevators, ailerons, rudder, speedbrake and body flap to perform its tasks. Classically, aero coefficient parameter identifications are best obtained by maneuvers which isolate the effect of individual derivatives contributing to the motion (i.e., aileron motion with rudder and yaw jets inactive). When several control parameters are desired (i.e., δ_A, δ_R, Jets), sequential independent excitations of each input should yield the most favorable result. This "isolation" of independent control effectors precludes "trading" by the data extraction programs. As shown by Table A-4 except very low dynamic pressure flight, the Orbiter flight control system prevents the use of the above described optimal aero data extraction maneuver format. For reasons of flight safety, software verification, and other considerations, the aero data extraction maneuvers for OFT must be performed within the constraints of the baseline flight control system. As an example, at M = 3, this restriction can result in simultaneous activity by the aileron, rudder, and yaw jets from a roll stick input. In addition, the flight control system, in general, intentionally restricts the magnitude of sideslip angle, making identification of this parameter difficult. Direct inputs to specific controls could be made available on later flights when the basic vehicle safety has been established.

Within the baseline FCS constraints depicted in Table A-4, various maneuver options are available including variations in input type, duration, magnitude and timing. The variation in parameter identification capability as a

TABLE A-4

ENTRY CSS MODE FLIGHT CONTROL EFFECTOR UTILIZATION

CHANNEL	PITCH		ROLL				YAW			BODY FLAP	SPEEDBRAKE
MANUAL INPUTS	RHC; PANEL TRIM; RHC TRIM		RHC; PANEL TRIM; RHC TRIM				RUDDER PEDALS; PANEL TRIM			MANUAL B.F. CMD	MANUAL S.B. CMD
EFFECTOR	PITCH JETS	ELEVATOR	YAW JETS	ROLL JETS	AILERON	RUDDER	YAW JETS	AILERON	RUDDER	BODY FLAP	SPEEDBRAKE
FLIGHT REGION											
400,000 FT. (ENTRY INTERFACE)	CONTROL		CONTROL	COORDINATION						TRIM	FIXED SCHEDULE
$\bar{q}$ = 0.5 PSF		PITCH JET COMPENSATION									
$\bar{q}$ = 2.0 PSF		CONTROL; TRIM			CONTROL; TRIM; COORDINATION						
$\bar{q}$ = 10.0 PSF				INACTIVE							
$\bar{q}$ = 20.0 PSF	INACTIVE										
MACH 4.5						TRIM; COORDINATION	CONTROL	COORDINATION	CONTROL		
MACH 4.0 PLUS 40 SECONDS					CONTROL						
MACH 1.5											
MACH 1.0			INACTIVE				INACTIVE				
MACH 0.9											ENERGY MANAGEMENT

function of these options is not always readily apparent, and preflight simulations alone are not adequate to rigorously address this problem. For OFT the maneuver formats are being developed based upon ALT experience, practice extractions from simulated OFT maneuvers, and basic conceptual criteria.

The basic objective of this experiment is to use the Shuttle as a test bed for evaluation of various maneuver formats to support other OEX aero data extraction experiments and for substantially improving the maneuver format requirement technology for application to future developmental programs similar to that of the Shuttle.

2.0 Feasibility

The basic conceptual feasibility of this experiment will be verified by the aero data extraction efforts of the mainline program during OFT. Data extractions for the OEX programs will be enhanced by the availability of the Shuttle Entry Air Data System (SEADS) which will complement the already in place Aerodynamic Coefficient Identification Package (ACIP) in providing excellent environmental knowledge. MDTSCO has analyzed different maneuver formats for OFT inputs utilizing motion from Shuttle Procedures Simulator (SPS) man-in-the-loop simulation of OFT flight test requirements. Figure A-26 presents an example utilizing the roll stick at M = 4.8. The coefficient identifications were obtained with the MDTSCO MLSIP aero data extraction program. The three maneuvers were of the roll doublet type and can be differentiated as:

(1) Doublet with one second pause in between.

(2) Doublet with no pause, and

(3) Stick rap followed by doublet.

3.0 Requirements

In general the requirements for this experiment are similar to the other aero data extraction oriented studies requiring SEADS, ACIP, specific maneuvers at given flight conditions and flight measured-to-analysis data processing capability. SEADS and ACIP were originally conceived with experiments of this type in mind. The data handling and manipulation techniques (DAP) being developed for OFT by the NASA JSC Engineering Analysis Division (EAD) would be directly applicable to this experiment with some minor changes for SEADS.

The OFT programmed test inputs (PTI's) as currently conceived appear to be general enough to accommodate format alterations required by this

MACH 4.8 SPS MANEUVER FORMAT STUDY

MLSIP RESULTS

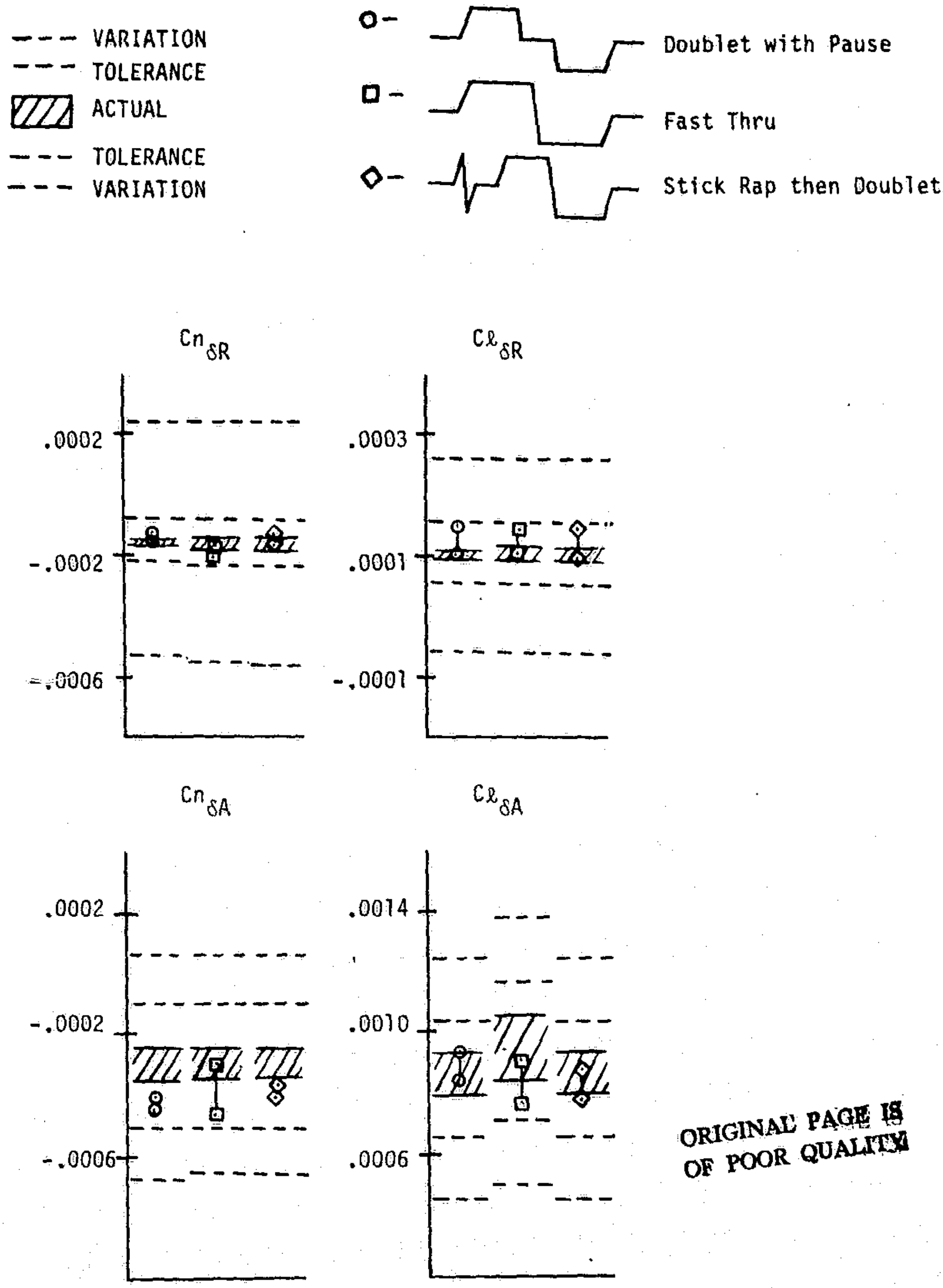


FIGURE A-26

experiment utilizing I-loads. Aerodynamic stick inputs (manual pilot inputs) are probably not acceptable for this experiment as the planned specific alteration in inputs probably could not be accurately accomplished by the pilots. It is important for this experiment that PTI capability is not deleted at the close of OFT.

Planning, analysis and documentation costs for this experiment would be a function of the organization performing the study as well as the number of flights and maneuvers per flight. Hardware and software costs for this experiment appear to be negligible by utilization of in-place items such as SEADS, ACIP and DAP. Based upon a ten (10) flight program it is estimated that this experiment will require 4800 manhours.

Should direct inputs to individual controls (non-baseline system software change) be pursued on later flights of this experiment, the software requirements would increase. Cost relative to software changes outside the normal FCS are difficult to ascertain at this time due to the prematurity of the situation. However, potential use of the present OFT downmoding switches in the cockpit may provide a means for pilot access to software that would provide channel separation. Some of the downmoding software may also provide the required function through appropriate software I-load changes and help minimize actual software changes.

4.0 Mission Impact

This experiment could have impact with respect to the mission as a result of RCS propellant requirements and possible crew activity conflicts with other experiments.

Current MDTSCO analysis of SPS and SSFS simulations indicate typical RCS propellant requirements as follows:

• Pitch Doublet	$\bar{q} < 20$	60 lbs.
• Roll Doublet	M = 5.5	120 lbs.
• Roll Doublet	M = 4.8	100 lbs.
• Roll Doublet	M = 2.3	30 lbs.
• Roll Doublet	M = 1.4	0 lbs.
• Yaw-Roll Doublets	M = 2.3	60 lbs.
• Yaw-Roll Doublets	M = 1.4	20 lbs.

Thus, some selectivity will be required with respect to the number of maneuvers performed on a given flight and to the flight region in which to perform the maneuver.

Current studies indicate that 10-15 seconds of crew time will be required for a given maneuver. At this time no conflicts are apparent, and the crew time demands seem reasonable. Although effects of guidance interruptions to perform the maneuvers must be addressed as an individual maneuver basis, the SPS simulation did not reveal any problems for the OFT maneuvers. Safety aspects (exclusive of direct inputs) are similar to the baseline system. Safety aspects would have to be given acute attention should direct inputs outside the baseline flight control system be utilized.

EXPERIMENT H: INSTRUMENTATION QUALITY IMPACT ON AERO DATA EXTRACTION

1.0 Background and Objectives

The design of instrumentation systems, which provide information needed to determine the aerodynamic characteristics of flying vehicles, is usually hampered by the difficulty in defining the requirements of the system. Cost constraints on sensor quality, data acquisition and transfer systems, and isolation of the sensors from adverse operating environments (temperature, vibration, etc.) combine with similar constraints on analysis resources to produce an apparently less-than-optimum resolution of the aero data.

The Shuttle Program has provided a unique opportunity to evaluate two different approaches to flight data analysis Instrumentation Systems. Early recognition of the probable incompatibility of the primary Flight Control System (FCS) sensors with the stringent aero data extraction requirements needed to resolve the Aero Flight Test Requirements (FTR's), permitted the introduction of the Aerodynamic Coefficient Identification Package (ACIP). This package was designed to satisfy the specific objectives of Aero Data Extraction (ADE). However, firm design requirements, which were difficult to define, were resolved by selecting sensors with range compromises to prevent inadvertent signal saturation and with quality that was not beyond the state-of-the-art. A data acquisition system with resolution capability up to 64 times better than the FCS was also incorporated. In addition, the package was also installed in the best possible location.

Although the basic FCS sensor and data resolvers were comparable to many existing systems used for ADE, it was not intended to be used for that purpose in the Shuttle. It is now possible to evaluate the cost effectiveness of the ACIP approach compared with a typical standard system. The suggestion for this experiment was made by MDTSCO personnel who participated in defining the design requirements for the ACIP.

The overall objectives of this experiment are to evaluate the differences between the two systems and the individual component sensors in the systems. This can best be accomplished by direct comparision of ADE results using various combinations of sensors from the ACIP and the FCS. Through carefully selected combinations of sensors, an evaluation of the quality of both the total systems and its parts can be obtained. Specifically, the detailed objectives of this experiment can be defined as follows:

a) Compare ADE results using all ACIP sensors and all similar FCS sensors.
b) Compare ADE results using differentiated body rates in place of angular accelerations.
c) Compare ADE results using FCS replacements for ACIP rates and accelerations independently.

These options are presented in Table A-5. The "baseline", all ACIP sensors, would be compared with the results obtained from the various other combinations of sensor outputs.

TABLE A-5

FCS/ACIP SENSOR EVALUATION FOR AERO DATA EXTRACTION

OPTION	LINEAR ACCELERATION A_X	A_Y	A_Z	ROTATIONAL RATES p	q	r	ROTATIONAL ACCELERATION p	q	r
BASELINE	A	A	A	A	A	A	A	A	A
1	F	F	F	F	F	F	F	F	F
2	A	A	A	A	A	A	d/dt (p,q,r from A)		
3	A	A	A	A	A	A	F	F	F
4	A	A	A	F	F	F	A	A	A
5	F	F	F	A	A	A	A	A	A

A - ACIP SENSOR; F - FCS SENSOR (ROTATIONAL ACCELERATIONS ARE OBTAINED FROM DFI ACCELEROMETERS)

2.0 Feasibility

The availability of two redundant instrumentation and retrieval systems provides an unprecedented opportunity to compare capabilities for certain selected redundant sensors. The redundancy primarily exists with the Orbiter rate gyros, accelerometers, and angular accelerometers which are duplicated in the ACIP. The Orbiter data is telemetered and recorded as it is used to provide inputs to the FCS. The ACIP data is completely passive and is recorded only. Table A-6 summarizes the range/resolution requirements of the systems while Table A-7 summarizes the characteristics of the respective sensors. All control surface and environmental support data, which is required for ADE, and is common to both systems, is derived from analyses of several sources.

TABLE A-6

ACIP RANGE/RESOLULTION REQUIREMENTS

Parameter	Units	OFT-1 Nom[4] Max	Min	Current MML Range	Required ACIP Range	Resolution ACIP[1] Required	Resolution MML Available	Resolution ACIP[6] Available
1. Roll Rate	Deg/Sec	12.8	-8.2	±40	±30	.014[5]	.078	.0037
2. Pitch Rate	Deg/Sec	3.7	-1.5	±20	±10	.005[5]	.039	.0012
3. Yaw Rate	Deg/Sec	3.1	-3.3	±20	±10	.005[5]	.039	.0012
4. Roll Accel	Rad/Sec^2	.26	- .18	±3.0	±2.0	.00229	.0234	.00024
5. Pitch Accel	Rad/Sec^2	.05	- .03	±1.0	±1.0	.00068	.0078	.00012
6. Yaw Accel	Rad/Sec^2	.03	- .03	±1.0	±1.0	.00048	.0078	.00012
7. Lat Accel	G	.022	- .021	±1.0	±.5	.00005	.0020	.00006
8. Norm Accel	G	0	-1.95	±4.0	±3.0	.00189	.0078	.00037
9. Axial Accel (2)	G	0	.8	--	±1.5	.00020[3]	--	.00018

NOTES

(1) REFERENCE MDTSCO 1.2-TM-B0705-1255 DATED 29 JULY 1977

(2) ESTIMATED FROM NORMAL AND DRAG ACCELERATION DATA

(3) LANGLEY REQUIREMENT

(4) OFT-1 DATA TAKEN FROM RI TRAJ 041079547 DATED 030977

(5) RATE RESOLUTION REQUIREMENTS ESTIMATED USING 12-BIT A-D CONVERSION

(6) USING 14-BIT AD CONVERSION

TABLE A-7

SENSOR ERROR BUDGETS

SYSTEM	SENSOR	LINEAR ACCELEROMETERS			ROTATIONAL RATE GYROS			ANGULAR ACCELEROMETERS		
	UNITS	(G)			(DEG/SEC)			(RAD/SEC²)		
	AXIS	A_X	A_Y	A_Z	p	q	r	$\dot{p}$	$\dot{q}$	$\dot{r}$
ACIP	RANGE	±1.5	±0.5	±1.0	±30	±10	±10	±2.0	±1.0	±1.0
	ERROR (1σ) (LOW RANGE)	±.0035	±.0018	±.0065	±.064	±.021	±.021	NOT AVAILABLE		
	ERROR (1σ) (HIGH RANGE)	±.0072	±.0055	±.0111	±.122	±.075	±.052	NOT AVAILABLE		
FCS	RANGE	-	±1.0	±4.0	±40	±20	±20	±3.0	±1.0	±1.0
	ERROR (1σ) MID RANGE	NOT AVAILABLE			±.100	±.100	±.100	NOT AVAILABLE		

The basic tools for analyses of data in this experiment are the Aero Extraction programs. Some versions of these programs have been used extensively and are fairly well understood. The Modified Maximum Likelihood Estimator (MMLE) is the basic 3-DOF linear program which exists in several versions including an advanced model. A more sophisticated 6-DOF nonlinear program known as the Maximum Likilihood System Identification Program (MLSIP) is also available and has demonstrated good capability in special situations although experience levels in using the program are lower than MMLE. Although either of these programs can be used, it is expected that a version of MMLE is the best compromise between cost and quality of results.

Another factor in this analysis is the selection of the ADE maneuvers that must be made to provide the necessary vehicle motion. All planned maneuvers will provide data for both sensor sets but analysis of all maneuvers is not required. Selection of several appropriate maneuvers in which roll, yaw, or pitch motion is developed must be made.

During the OFT portion of the Shuttle Program, it is planned to provide extensive analysis of post flight entry data in support of mandatory FTR's that qualify the Orbiter for operational use. Following the OFT phase, some of the Developmental Flight Instrumentation (DFI), such as the rotational accelerometers, is currently scheduled to be removed. This would impact a portion of the planned experiment if it was scheduled post-OFT. Analyses during OFT are expected to use both the MMLE and MLSIP programs previously described.

A successful conclusion to this experiment would lie in making available information to develop criteria for ranging, accuracy, and resolution requirements for the respective sensors. However, several problems areas have been identified which must be considered. This includes a time correlation phenomenon which is applicable to the FCS rate gyros and accelerometers data. This phenomenon occurs due to delays (staleness) in General Purpose Computer processed quantities being sent to the recorder or ground stations. Although techniques have been devised to deal with the delays, it requires careful supervision of the ADE input data to resolve the accountability of all time staleness problems.

Another problem area that could cloud all analysis is in the development of appropriate environmental data that is required for each analysis. Difficulties in defining the meteorological properties, although common to both analyses, may obscure other problems. Delaying the experiment until the Shuttle Entry Air Data Systems (SEADS) is available would minimize this impact. SEADS will supply atmosphere density, temperature, dynamic pressure, and angles of attack and sideslip data. A third area, where problems are anticipated, is the impact of noise on the rotational accelerometer output. Prior studies have shown that differentiation of the rates may be preferred to using the raw sensor data.

3.0 Requirements

The requirements to perform this experiment are minimal since all the materials, tools, and data are being provided for planned OFT studies. No additional requirements exist beyond the implementation of the various alternate options to produce the desired primary analysis. Resources required to complete this experiment include computer simulation time and manpower. Estimated resource levels are 3-4 man-months and 10-15 computer hours. If the experiment is delayed until the SEADS is available, additional requirements in the form of data preparation support will be needed.

4.0 Impact of Experiment on Mission Time

There are no special considerations required to support this experiment if it is accomplished within the OFT phase of the Shuttle program. All the maneuvers that are required to sustain ADE are planned in that phase. If the experiment is performed in the operational phase, then planning and scheduling of maneuvers will be needed to coordinate them with other planned ADE analyses maneuver requirements. The scheduling of maneuvers to support this experiment alone should be unnecessary.

EXPERIMENT I: AERO DATA EXTRACTION

1.0 Background and Objectives

The objective is to advance the state of the art with respect to ground (wind tunnel) test to flight test aerodynamic technology by means of extraction of aerodynamic stability and control derivatives during Orbiter entries. The Shuttle presents a unique opportunity for this experiment due to its extensive ground test data base (including wind tunnel uncertainties) as well as atmospheric flight in the supersonic and hypersonic flight regimes where data are scarce with respect to ground to flight accuracies and extrapolations. As indicated in Figure 1, several of the other proposed experiments are designed to provide the best possible technology in support of this experiment. The hypersonic viscous experiment has been retained in a separate category due to the unique relationship of viscous interaction and control surface effectivity.

As is the case for many of the other experiments, the combined use of the Aerodynamic Coefficient Identification Package (ACIP) and Shuttle Entry Air Data System (SEADS) will provide the highest fidelity instrumentation for acquisition of full-scale atmospheric flight data within the state of the art for the early 1980 time period. The orbiter coefficient uncertainties are either characterized as "tolerances" or "variations." The tolerances have been derived from analyses of Orbiter wind tunnel tests and are essentially ground test uncertainties. Thus, the tolerances are the minimum pre-flight uncertainties resulting from unexplained differences in test results from various wind tunnels with various models. The variations represent a historical comparison between flight and pre-flight predictions of aerodynamic coefficients for various aircraft and spacecraft. The variations are essentially ground test to flight test uncertainties. As expected, the variations are substantially greater than the tolerances. The Orbiter flight-to-flight uncertainties are anticipated to be on the order of the tolerances.

The mainline program has identified various flight situations where the aero variation would result in undesirable flight characteristics. These regions and the primary coefficients pertinent to these undesirable characteristics are summarized in Table A-8. These conditions will

TABLE A-8

AERODYNAMIC FLIGHT TEST REQUIREMENTS

IN ORDER OF PRIORITY:

FTR NUMBER	FLIGHT REGIME	PLACARD	DERIVATIVES IN QUESTION	PROBLEM DESCRIPTION
07VV007	ALL	AFT C.G. FWD C.G.	$\delta_{e_{trim}}$, L/D	OTHER FTR'S AND HEATING PROBLEMS RELATE TO TRIM. RANGING PROBLEMS OCCUR WITH LOW L/D VARIATION.
07VV009	HIGH SUPERSONIC (M = 3 TO 5)	Y C.G.	$C_{n_{\delta_r}}$, $C_{\ell_{\delta_r}}$, $C_{n_{\delta_a}}$, $C_{\ell_{\delta_a}}$ (C_{ℓ_β}, C_{n_β})	RCS PENALTY OR LOSS OF VEHICLE DUE TO LOW RUDDER EFFECTIVENESS IN LAT/DIR TRIM
07VV006 07VV008	TRANSONIC SUPERSONIC (M = 1.5-2.5 1.0 TO 1.5)	AFT C.G. (FCS SWITCH POINT)	$C_{n_{\delta_a}}$, $C_{\ell_{\delta_a}}$, $C_{n_{\delta_r}}$, $C_{\ell_{\delta_r}}$ (C_{n_β}, C_{ℓ_β})	LAT/DIR CONTROL PROBLEM DUE TO LOW RUDDER EFF. AND POSSIBLE LOSS OF ROLL CONTROL
07VV005	HYPERSONIC VISCOUS (LOW $\bar{q}$)	AFT C.G.	$C_{m_{\delta_e}}$ (PITCH JET EFFECTIVITY)	EXCESSIVE RCS FUEL USAGE DUE TO REAL GAS EFFECTS - RESULTS IN LONGITUDINAL TRIM PROBLEM
07VV003	HYPERSONIC (M = 5 TO 8)	FWD C.G. Y C.G.	$C_{n_{\delta_a}}$, $C_{\ell_{\delta_a}}$ (C_{n_β}, C_{ℓ_β})	EXCESSIVE RCS FUEL USAGE DUE TO LAT. DIR. TRIM PROBLEM WITH $C_{n\delta a}$ HAVING WRONG SIGN. (NO RUDDER)
07VV010	HIGH SUBSONIC (M = 0.9)	AFT C.G.	$C_{n_{\delta_r}}$, $C_{\ell_{\delta_r}}$, C_{n_β}, C_{ℓ_β} ($C_{n_{\delta_a}}$, $C_{\ell_{\delta_a}}$)	LOSS OF RUDDER EFFECTIVENESS AND Cn_β FOR δ_{SB}<15°
07VV004	TRANSONIC	FWD C.G.	$C_{H_{e_I}}$, $C_{H_{e_O}}$, $C_{H_{BF}}$, $C_{H_{R/SB}}$	CONTROL SURFACE STALL OR INADEQUATE RATE CAPABILITY

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probably be adjusted as a result of additional analysis at NASA JSC, but are representative of the scope of the mainline effort. Figure A-27 demonstrates where maneuver would be performed on a typical entry profile to address these "placards" on the operational envelope. The Shuttle Program will only extract limited aero derivatives to the accuracy required for assuring integrity of the Shuttle system. The OEX experiment will thus be directed

ALTITUDE VS. TIME FROM ENTRY INTERFACE FOR ORBITER OFT

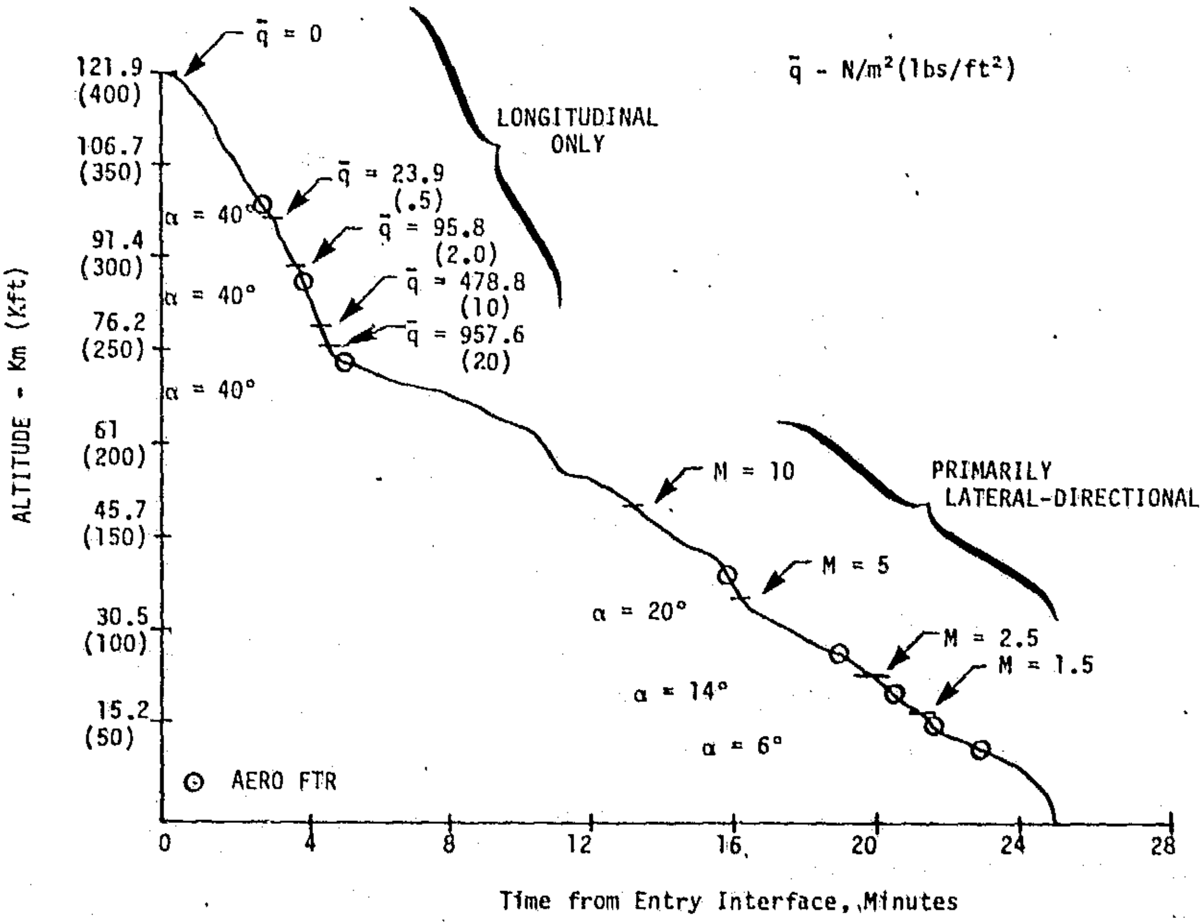


FIGURE A-27

toward those areas which will be given minimum scrutiny by the Shuttle Program due to the uniqueness of the design (physical and software) as well as to a more accurate and expansive determination of derivatives in areas investigated by the mainline program. This data should be very useful with respect to future Shuttle-like designs as well as other type high-speed vehicles.

2.0 Feasibility

The basic conceptual feasibility of this experiment has been verified for the Shuttle on ALT, is being verified by current simulation efforts, and will be verified by the mainline program during OFT. Figures A-22 through A-25 presented typical ALT results while Figure A-26 presented typical results from Shuttle Procedures Simulator man-in-the-loop studies. The MDTSCO Maximum Likelihood Systems Identification Program (MLSIP) was used for the coefficient identification. As noted in the preceeding section, several other proposed experiments plus the combination of SEADS and ACIP should further enhance this effort.

3.0 Requirements

The requirements for this experiment are generally similar to the other aero data extraction oriented studies requiring SEADS, ACIP, specific maneuvers at given flight conditions and flight measured-to-analysis data processing capability. SEADS and ACIP were originally conceived with experiments of this type in mind. The data handling and manipulation technique (DAP) being developed for OFT by the NASA JSC Engineering Analysis Division (EAD) would be directly adaptable to this experiment with some minor changes for SEADS.

The OFT programmed test inputs (PTI's) currently conceived by JSC appear to be adequate to accommodate this experiment. The format for these PTI's is shown in Figure A-21. No additional requirements are anticipated.

Planning, analysis and documentation costs for this experiment would be a function of the organization performing the study as well as the number of flights and maneuvers per flight. Since SEADS and ACIP are required, this experiment would presumptively only be performed with the vehicle in which these items are installed. Thus, hardware and software costs for this experiment appear to be negligible by utilization of in-place items such as SEADS, ACIP, DAP, and the baseline PTI format. Based upon a fifteen (15) flight program it is estimated that this experiment will require 3800 manhours.

4.0 Mission Impact

The impact of this experiment on the mission is similar as that of the "Comparison of Aero Data Extraction Techniques" and the "Evaluation of ADE Maneuver Formats" experiments.

EXPERIMENT J: INVESTIGATION OF HYPERSONIC CHARACTERISTICS DUE TO VISCOUS INTERACTION AND REAL GAS EFFECTS

1.0 Background and Objective

MDTSCO and RI proposed experiments in the high-altitude hypersonic flight regime in order to obtain a better understanding of the effects of viscous interaction and real gas on control surface effectivity. In this region, the interaction of the shock wave with the boundary layer is complicated by the influence of real gas effects, which, when considered in conjunction with separated flow resulting from the downward (compressive) control deflections, present a substantial technological challenge. The Orbiter 102 aero data book under nominal conditions predicts control surface (elevator) reversal for elevator deflection (δ_e) in excess of 10 deg. downward at values of the viscous interaction parameter, $\bar{V}'_\infty$ (Table A-9), of 0.04 or greater. In addition, due to the very large uncertainties in the pitching moment characteristics of the Orbiter in this altitude region (65 or 100 KM), the mainline program is required to perform a flight investigation of sufficient accuracy to verify that the Orbiter can safely be trimmed and controlled over the design center-of-gravity envelope (65 to 67.5% ℓ_b).

As shown in Figure A-28, the preflight pitching moment (C_m) uncertainty (variation) in the very high viscous interaction regime ($\bar{V}'_\infty > .03$) is 5 to 6 times greater than that for the non-viscous interaction flight conditions ($\bar{V}'_\infty < .005$). Operational longitudinal CG placard removal can be accomplished with a 50-percent reduction in the high viscous interaction uncertainty. Since the resulting uncertainty would still be 2.5 times greater than the preflight non-viscous interaction uncertainty, and 5 times greater than the corresponding ground test uncertainty, ample room for further

TABLE A-9

$\bar{V}_\infty'$ DETERMINATION

$$T' = 726.97 + .468T_\infty + 3.63921 \times 10^{-5} V_\infty^2 \qquad (°K)$$

T_∞ in °K

V_∞ in m/sec

$$C_\infty' = \left(\frac{T'}{T_\infty}\right)^{.5} \left[\frac{T_\infty + 122.1 \times 10^{-(5/T_\infty)}}{T' + 122.1 \times 10^{-(5/T')}}\right] \qquad (N/D)$$

$$\mu = \frac{1.458 \times 10^{-6} T_\infty^{1.5}}{(T_\infty + 110.4)} \qquad (Kg/m\text{-}sec)$$

$$Re_{\infty_{L_B}} = \frac{V_\infty \rho_\infty L_B}{\mu} \qquad (N/D)$$

ρ_∞ in Kg/m^3

L_B = body length = 32.766 meters

$$M_\infty = \frac{V_\infty}{a} = \frac{V_\infty}{\sqrt{\gamma R T_\infty}} = \frac{V_\infty}{\sqrt{401.874 T_\infty}} \qquad (N/D)$$

$$\bar{V}_\infty' = M_\infty \sqrt{\frac{C_\infty'}{Re_{\infty_{L_B}}}} \qquad (N/D)$$

improvement is available. The combined availability of ACIP and SEADS on the post OFT flights will result in a more accurate assessment of the Orbiter viscous interaction characteristics and a corresponding reduction in the uncertainties over that obtainable by the mainline OFT Program. Additional flights will also allow for tests at $\bar{V}_\infty'$ conditions not addressed by the mainline program. In addition to a more accurate assessment of the viscous interaction effects, which will be applicable for future vehicle designs, this experiment will also provide data for studies directed toward expansion of the Orbiter design CG envelope.

The basic objective of this experiment will be to perform maneuvers and evaluate the effects of the viscous interaction parameter, $\bar{V}_\infty'$ on elevon and body flap control surface effectivities. As previously shown in Figure A-28, the C_m uncertainty is a substantial function of $\bar{V}_\infty'$ the effect of which is aggravated by the aforementioned nominal control reversal at $\delta_E > 10^\circ$ and $\bar{V}_\infty' \geq .04$. Experiments directed toward pressure measurements and associated studies for determining boundary layer separation conditions as proposed by RI in addition to the above are considered to be best addressed by aero heating or flow-oriented aero research programs and will not be addressed by this flight control experiment.

2.0 Feasibility

The basic feasibility of the aero data extraction approach will be verified by the mainline program during OFT. The OEX program will then be called upon to better define these characteristics as a result of the excellent environmental knowledge which should be available with the combination of the Shuttle Entry Air Data System (SEADS) and Aerodynamic Coefficient Identification Package (ACIP). Man-in-the loop simulations have been

PITCHING MOMENT COEFFICIENT UNCERTAINTY

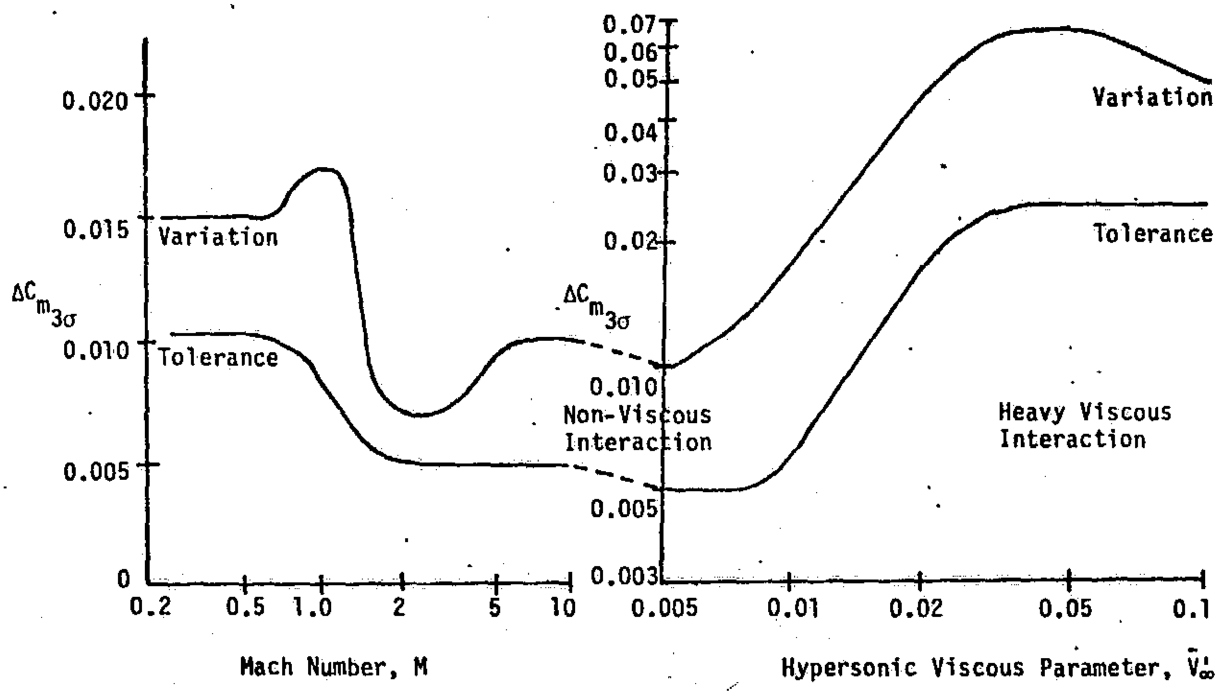


FIGURE A-28

performed on the JSC Shuttle Procedures Simulator (SPS) directed toward the basic feasibility of maneuvers in this region. Study objectives of these simulations included:

1. Verify that specified control inputs do not induce vehicle motion of sufficient magnitude to jeopardize vehicle control (with and without selected aero variations).
2. Verify that specified control inputs are of sufficient magnitude to induce adequate vehicle motion for aerodynamic data extraction (subjective).

3. Optimize control inputs with respect to crew procedures and techniques.
4. Verify that displays available to the crew are sufficient to perform the desired maneuvers.
5. Investigate the integration of planned maneuvers into OFT trajectories (what is a reasonable number of maneuvers per flight in appropriate flight regions with respect to pilot workload and auto-guidance interruptions?).
6. Determine which of the planned maneuver types and/or flight regions would be more conducive to Programmed Test Inputs (PTI's) than Aerodynamic Stick Inputs (ASI's).
7. Verify that the maneuvers can be easily and repeatably flown.
8. Investigate any problems which may be associated with returning to the automatic guidance system following performance of each maneuver.

Preliminary analysis of the simulation results verified the integrity of the concept and no substantial problems were encountered. Figure A-29 presents the standard condition where viscous maneuvers are planned for the mainline effort. Figure A-30 presents typical MLSIP results obtained from an equivalent off-line simulation at $\overline{q}$ = 3.0 PSF. Although the results represent analysis with "pure" data (no noise, perfect MET and air data) the agreement has been surprisingly accurate considering the discontinuities imposed by the up and down firing pitch jet activity and nonlinearity of the pitching moment characteristics. MLSIP identifies both nonlinear aero coefficients as well as resulting jet thrust effectivity (including interactions) in the presence of air-flow.

VISCOUS OFT REGIONS FOR EXTRACTION ANALYSIS

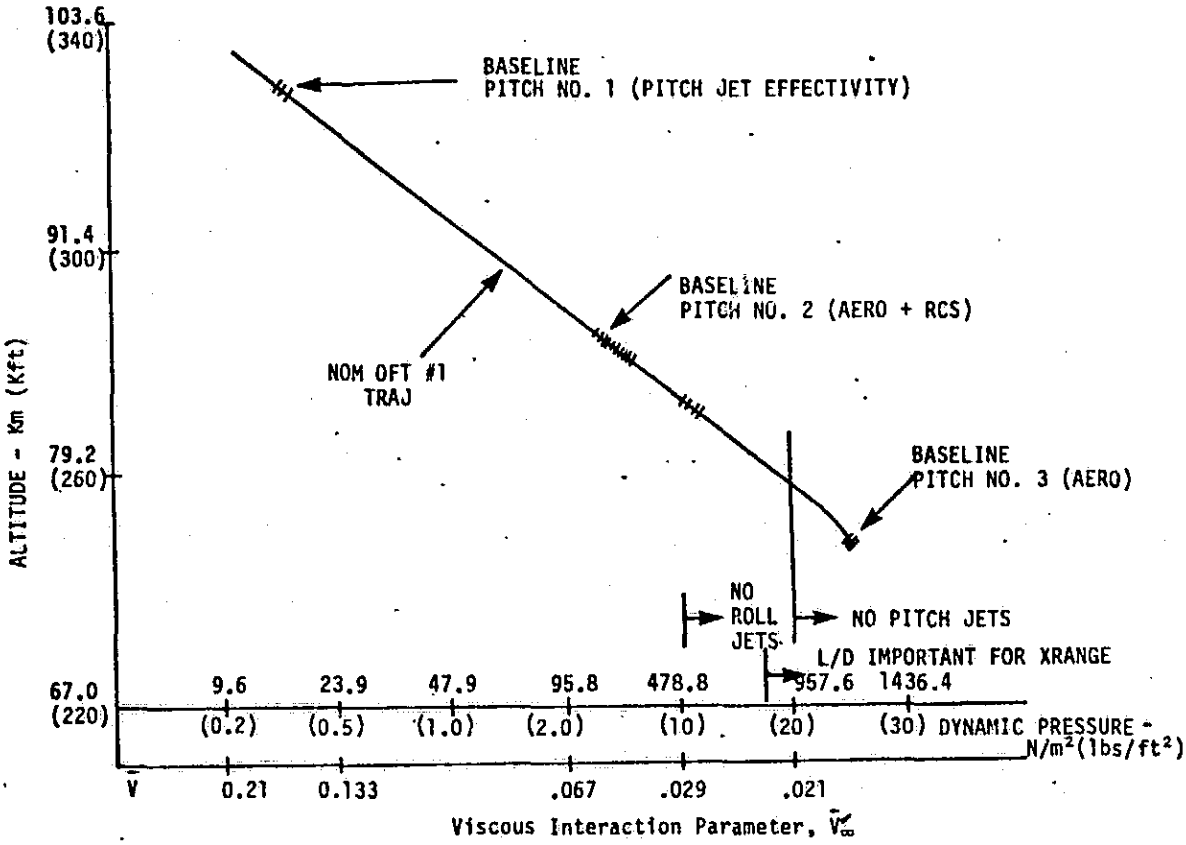
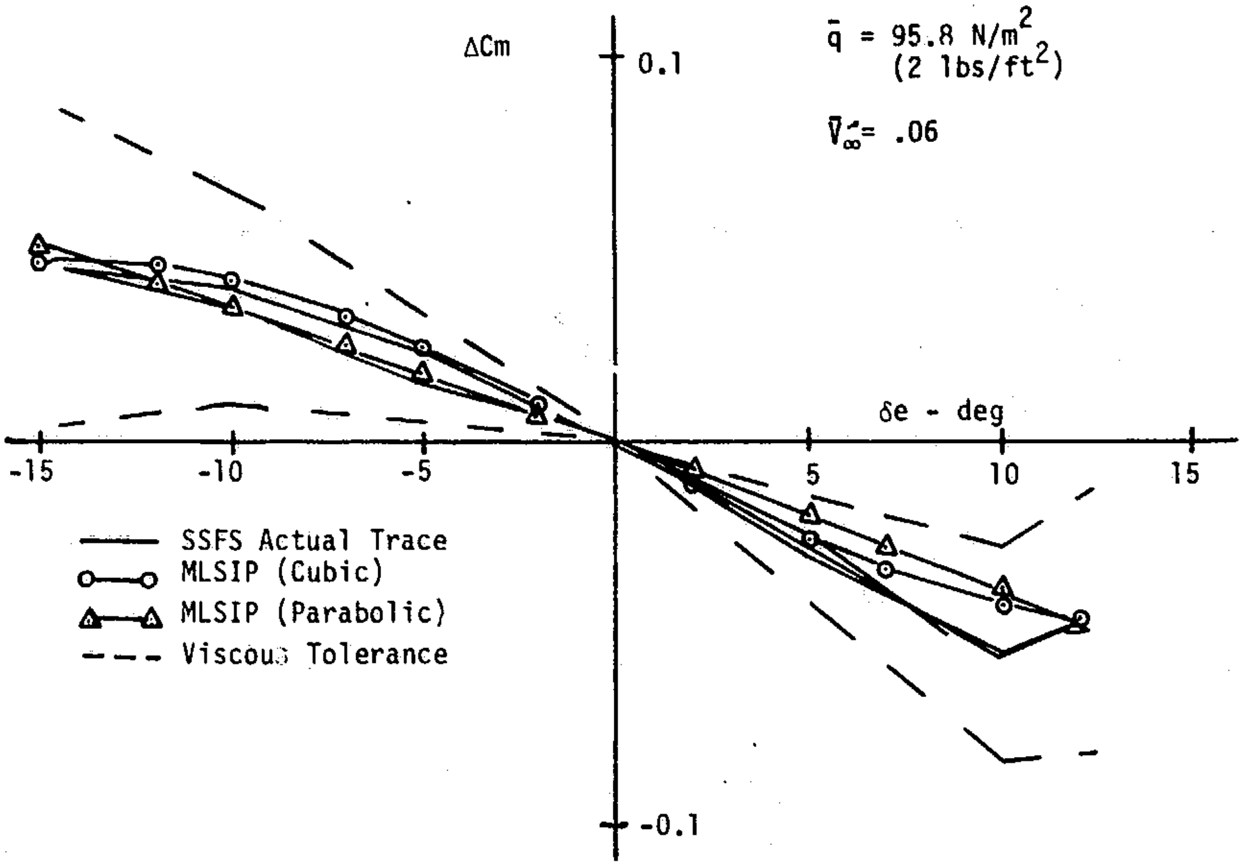


FIGURE A-29

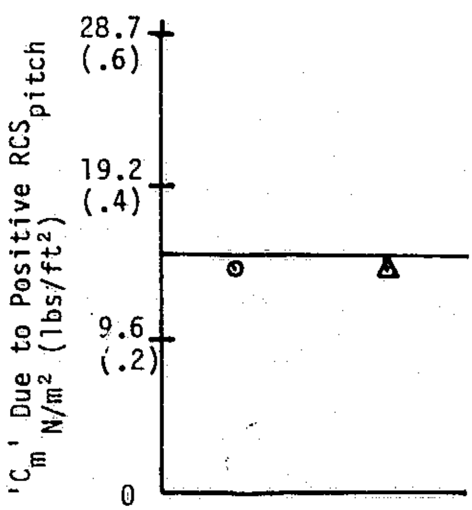
AERO DATA EXTRACTION SSFS MANEUVER SIMULATION

VISCOUS ELEVATOR EFFECTIVITY



PITCH JET EFFECTIVITIES

'C_m' = $M_{RCS}/(Sb)$



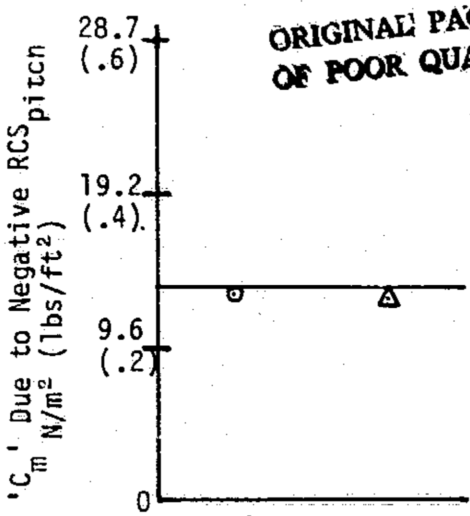


FIGURE A-30

3.0 Requirements

This experiment will require SEADS, ACIP, specific maneuvers to be performed at given flight conditions and essentially the same flight measured-to-analysis data processing capability that will be demonstrated on OFT. This process being developed by JSC-EX will be capable of providing processed (or non-processed if desired) flight motion at any flight condition in format compatible with the MMLE and MLSIP extraction programs. With some modifications to accommodate SEADS, this in-place capability would be ideal for OEX studies.

Unless special maneuvers considerably different than those utilized by the mainline program are required, minimal man-in-the-loop simulation and off-line simulations will be necessary. Since it has been decided to utilize programmed test inputs (PTI's) rather than aero stick (manual) inputs (ASI's), some software changes would be mandated should the mainline program remove PTI capability at the end of OFT. If the PTI capability is intact, only minor I-load changes will be required in order to further optimize the maneuver for purposes of aero extraction capability. Analysis costs would be a function of the organization performing the analysis and would consist of the standard costs including computer usage, technical analysis, report writing, etc. This cost would be a function of the number of flights and maneuvers analyzed. Hardware and software costs at this time to appear to be small by utilization of in-place instrumentation and data manipulation capability. Based upon a ten (10) flight program, it is estimated that this experiment will require 3000 manhours.

4.0 Mission Impact

This experiment could have impact with respect to the mission as a result of RCS propellant requirements and potential crew conflicts with other activities.

Current MDTSCO studies indicate that maneuvers in the viscous interaction region above $\bar{V}'_\infty = .02$ (the $\bar{q} \leq 20$ PSF point where the pitch jets are shut off) can result in 60 pounds of RCS propellant for a typical flight maneuver. At this time the maneuvers have not been optimized for RCS propellant usage and some relief may be available.

Current studies indicate approximately 10 to 15 seconds of crew time will be required for each maneuver. This is considered to have minimal effect, and no conflicts are apparent at this time. The safety aspects of this experiment are similar to that for the baseline effort and can be adequately accommodated and verified by additional off-line simulations if required.

EXPERIMENT K: INFLUENCE OF REACTION JET FIRINGS ON ORBITER FLIGHT CONTROL CHARACTERISTICS

1.0 Background and Objectives

A maneuvering vehicle in a zero or low dynamic pressure environment is usually controlled by reaction jet motors which provide the necessary impulses to stabilize its motion. The location of these jets can be critical to their effectivity due to impingement of the jets on vehicle surfaces and also due to interaction of the jets with flow around the vehicle as it enters the atmosphere. In addition, the firing of different combinations of jets may produce non-linear modifications to the impingement and interactions which are also functions of the relative wind vector, density and Mach number. Design information relative to jet firings is usually obtained by wind tunnel tests but the quality of information from the standard small sub-scale tests is usually suspect and validation is very difficult. This experiment will attempt an analysis of full-scale flight conditions and validation of the effectivity of the reaction jets. This validation is critical to Flight Control System (FCS) software which is impacted by logic required to deal with large losses in effectivity or even control reversal. This experiment was recommended by HI, DFRC, and MDTSCO.

The direct measurement of the forces and moments generated by the RCS during early entry ($\bar{q}$ < 20 psf) contains three contributions that modify the aero characteristics of the vehicle. These contributions come from jet plume impingement on the vehicle surfaces, jet interactions with flow around the vehicle and jet interferences due to various combinations of down-firing jets being used. A typical contribution is presented in Figure A-31 and shows the partial contributions from impingement, interactions, and carry-over (incremental change from symmetrical down-firing jets on

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TYPICAL RCS JET FIRING INFLUENCE ON AERO CHARACTERISTICS

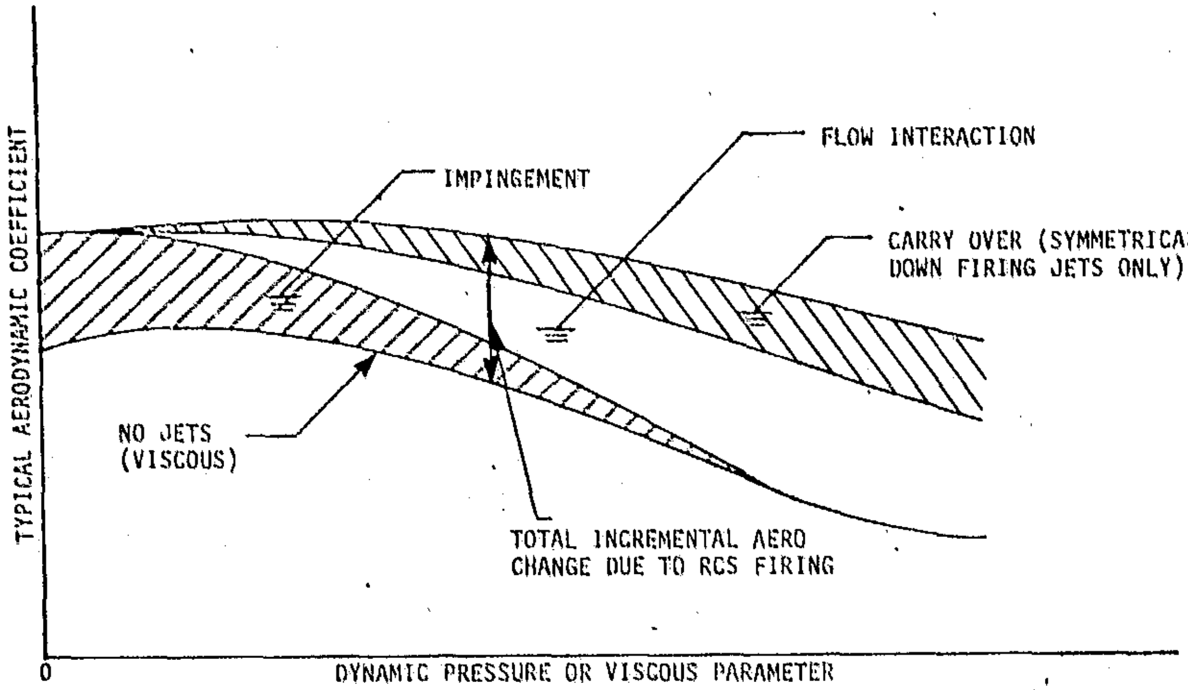

FIGURE A-31

either side) which modify the basic viscous aerodynamics.

Qualitative analysis of jet interaction effects from wind tunnel data is difficult to produce. Previous attempts have resulted in several approaches which occasionally have provided divergent solutions. The complexity of the problem, which is primarily influenced by scaling relationships is caused by the reduced size of the jet nozzles and interference from the models sting support which can be impacted by the jet plumes. In addition, the separate contributions from impingement on the Shuttle surfaces and interaction with the aerodynamic flow at various dynamic pressure ($\overline{q}$) levels are not easily measured and the division can only be resolved by analysis. Only at $\overline{q}$ of zero is the result clearly defined.

Impingement is eventually reduced to zero at the higher dynamic pressures as the jet plumes are bent away from the orbiter structure. However, the sting interference is a major hindrance in defining the total impact of the jet influence and its partial contributors. The full-scale flight test program will not have to contend with this impediment in the analysis and thus will improve the quality of the analysis to be performed.

The primary objective of this experiment is to determine the magnitude of this total contribution from $0 \leq \bar{q} < 20$ psf. The secondary objective will be to determine the magnitude of each of the three contributors over the same range. Satisfying these objectives will provide valuable support data into understanding the mechanism of the influence of RCS firings on control capabilities.

2.0 Feasibility

The resolution of the flight test data can only be done by the use of Aero Data Extraction programs which are capable of isolating RCS effects on body accelerations and rates. Tests of this capability have been performed on the Maximum Likelihood System Identification Program (MLSIP) with good results and additional improvements are expected as experience levels increase. Modifications, similar to those made to MLSIP, have been included in the Modified Maximum Likelihood Estimator (MMLE) and are expected to provide similar results pending completion of testing.

The feasibility of this experiment hinges on several important factors. The measurement of the motion data is currently planned for the Aerodynamic Coefficient Identification Package (ACIP) installation which appears to be capable of providing the desired level of accurate information. The conversion of this data to aerodynamic coefficients is dependent on high quality determination of the flight test environment. To date, the only available technique is the restructured atmospheres produced from interpolated data from sounding balloons and rockets. Although this data quality is marginal in the very low $\bar{q}$ regions, it is expected to be good enough to resolve the Flight Test Requirements (FTR) Placards. However, it is only through the use of the proposed Shuttle Entry Air Data System (SEADS) and Shuttle Upper Atmosphere Mass Spectrometer (SUMS) that improvements necessary to support the accuracy requirement of this very high altitude experiment can hopefully be obtained. This experiment should be delayed until SEADS and SUMS are available. The available extraction programs are expected to be able to define the desired aero data with and without the jets operating, but analysis of the RCS-on data needs to be supported by additional research. Most of this experimental analysis is a follow-on to analysis already planned for early Shuttle flights but, as proposed, requires additional capabilities not currently available.

DFRC had proposed instrumenting the Orbiter surface with pressure taps in the vicinity of the RCS pods. Although there is some merit to this approach, it does imply a cost option that may not be commensurate with results. The benefits from this instrumentation should aid in defining the total contribution of the change but is probably of lesser value in separating the contributing parts. The evaluation of this experiment does not include consideration of the pressure taps pending further examination of its effectiveness for the experiment.

3.0 Requirements

A successful conclusion to this experiment is predicated on the availability of the SEADS and the SUMS which provide the proper technical means to obtain reference atmosphere information. Since the aero coefficient data reduction is dependent on the derived dynamic pressure ($\bar{q}$), errors in $\bar{q}$, if force (acceleration) errors are minimal as expected, will produce similar errors in the aero coefficients. The object of accurate aero coefficients is to validate the wind tunnel tests and the uncertainties associated with them (tolerances). Validation becomes practically impossible if flight test data uncertainties are greater than the tolerances simply because of the inaccuracies in reducing the flight data.

Additional requirements are needed in establishing a jet operations test format. Although individual jets cannot be selected, it is possible to turn off (deselect) certain jets and thus exert some selectivity over desired jets. This selectivity will probably differ in detail from current maneuvers planned to support ADE. As a safety-of-flight issue may develop from these expected new maneuvers, a man-in-loop simulation would be needed in addition to the off-line studies.

Resources to support this experiment would require 1-2 man-years and a computer budget less than 100 hrs. If research into the theory and application of jet influence characteristics is considered, an additional 1 man-year could be expected. Several man-in-loop simulation periods of about 1 week each would be needed to support flight safety issues.

4.0 Impact of Experiments on Mission

The maneuvers, which are produced by the jet impulses, must be performed in the very early phases of entry and as such will raise only a slight safety-of-flight issue. However, it will still demand the pilot's attention and will therefore be reflected in his time-line. The pilot's participation will include control maneuvers, but these are expected to have minimal impact on trajectory variations. Approximately 60 pounds of RCS propellant will be required for each flight maneuver in the low dynamic pressure regime (less than 20 psf) where most of the experiment is conducted.

EXPERIMENT L: ANALYTICAL REDUNDANCY FOR DETECTING SENSOR FAILURE

1.0 Background and Objectives

Analytical redundancy for detecting sensor failure was suggested as an Orbiter flight control experiment by Thomas B. Cunningham, of Honeywell, Inc., Minneapolis, Minnesota. Mr. Cunningham, in conjunction with other individuals, has done extensive research in the field of analytical redundancy management. The progress of this research has been documented by numerous papers. The bulk of this text is taken from two such papers (Reference 1 and Reference 2). If a more detailed insight into the subject is desired, Reference 1 contains a comprehensive list of material covering the theory of analytical redundancy and its application.

Performance and reliability requirements in modern flight control systems has increased the number of sensors required and thus system cost. Performance drives system complexity up. Reliability requirements, particularly for fly-by-wire aircraft, have resulted in high degrees of sensor redundancy. Reducing the high costs associated with these increases has resulted in techniques to reduce the number of sensors required as well as the complexity of the associated redundancy management.

These techniques can be classified as (1) Control Law Modification and (2) Fault Tolerant Design.

- Control Law Modification

The technique of control law modification is to minimize the number of sensors that are required to meet performance requirements. The issues are complexity versus performance. Reduction in the number of sensors is traded off against increased complexity of the control laws.

• Fault Tolerant Design

The technique of fault tolerant design is to reduce the number of redundant sensors needed for reliability by 1) skewed and special sensors, 2) integration for redundancy management (sensor sharing between flight control and navigation), 3) in-line monitoring, and 4) analytical redundancy.

•• Skewed and Special Sensors - A skewed sensor arrangement can significantly reduce the number of sensors required for redundancy management. For example, with orthogonal gyros in a three-axis system, a total of 12 are required for a quad-redundant dual-fail-operative capability. The same system with skewed gyros requires only six for the same capability. However, skewing has practical limitations. For gyros, the scale and resolution requirements are different for the three axes. In a conventional (orthogonal) system, the roll rate gyro must have a larger scale or range than the pitch rate gyro.

Conversely, the pitch rate gyro requires more resolution. In a skewed arrangement all instruments must be the same. This will either limit the resultant signal quality or increase the component cost, potentially by more than the savings accrued by eliminating six conventional gyros.

•• Integration for Redundancy Management - Another way to reduce redundant sensors is through subsystem integration. The concept uses sensor data from subsystems which are not normally functionally related for monitoring and tie breaking.

•• In-Line Monitoring - Still another way to achieve fault tolerant design is through in-line monitoring. However, in-line sensor self-test feasibility is limited by several factors. The input to the sensor is unknown except when special test signals are introduced. Self-test techniques do not include sensor installation errors (base mounting). Finally, the additional complexity and cost associated with self-test may override the savings gained by reducing the number of sensors.

•• Analytical Redundancy - Analytical Redundancy is the least developed of these techniques, but offers the potential to signficantly reduce the number of redundant sensors required while maintaining system reliability. The basic idea of analytical redundancy is to use known relationships between different sensors in order to detect failures. Various theoretical and simulation studies have shown that sensor failures can be detected by exploiting known functional relationships between different sensors. For aircraft flight control sensors, these would be kinematic and dynamic equations of motion. The possibilities shown by these studies have opened up a whole new approach to failure detection with significant savings potential.

The feasibility of analytical redundancy has been adequately demonstrated by the studies that have been performed utilizing off-line and hybrid simulations of the A-7D aircraft and the A-7 flight control and sensor complement. To date, three different redundancy concepts have been studied using the A-7D aircraft control configuration. These concepts are:

CONCEPT I. Observer/Blender - Concept I specifically attempts to blend related sensors into a reconstructed output. An error signal is produced when the reconstructed output is compared with the actual sensed output. This concept has the advantage of low computational requirements but the disadvantage of degraded performance over Concept II or Concept III.

CONCEPT II. Kalman Diagnostic Filter (KDF) - Concept II uses an assembly of Kalman filters to produce a complete fault detection capability for a given set of sensors. Fault isolation is obtained by monitoring signals derived from the KDF's and standard comparators. Concept II has the disadvantage of extra computational expense as compared to Concept I.

CONCEPT III. Super-Diagnostic Kalman Filter - Concept III addresses the fault isolation problem as well as the detection problem. It also creates an error signal for each sensor treated. It has the disadvantage of greater computational requirements when compared with Concept I or II.

To add further credence to the maturity and feasibility of analytical redundancy, an analytical redundancy flight test program will be conducted utilizing the A-7D aircraft in the latter part of 1978. Basically, the tests will employ Concept II, hence that concept is described in the following paragraphs. Results of the flight tests will provide additional insight to the feasibility of Analytical Redundancy and provide a basis for whether a Shuttle OEX experiment is warrented. Assuming the A-7D test results are positive, an Orbiter experiment is a logical follow-on since the Shuttle will afford a much wider flight environment to test the capabilities of Analytical Redundancy.

2.0 Feasibility

The A-7D has dual Honeywell digital computers (HDC301) and dual servos in each axis. This aircraft and its sensor complement exemplifies typical sensor redundancy for mission reliability.

The sensor complement can be broken into two categories.

1. The mission essential sensors used in the basic Control Augmentation System (CAS) are the body rates (P, Q, and R) and the lateral and vertical acceleration (N_Y and N_Z).

2. The sensors which are not essential to the mission and are not used in the essential feedback structure are roll angle (ϕ), pitch angle (θ), yaw angle (ψ), altitude (H), angle of attack (α), and airspeed (U_{AS}).

The specific goals of the Analytical Redundancy for this aircraft were to obtain:

1. Fail-operative flight control for mission essential sensors (voting techniques with only hardware redundancy require three sensors measuring identical quanities).

2. Fail safe for non-mission essential sensors and mission essential sensors after one failure (voting techniques require two of each sensor).

To accomplish the goals of the A-7D program, an Analytical Redundancy scheme as shown pictorially by Figure A-32 was developed.

The goals of Analytical Redundancy are accomplished through fault detection of a failed sensor and then fault isolation to eliminate the failed sensor from the system. Fault detection is realized through the Kalman Diagnostic Filters (KDF) and the comparison monitors. The KDF requires accurate sensor characteristics (see Section 2.1) such as Biases (b) and Scale Factors (SF) and sensor inputs to generate the filter equation residual (ν_i). These residuals are used for fault detection as will be shown in

ANALYTICAL REDUNDANCY SCHEME FOR DUAL SENSORS

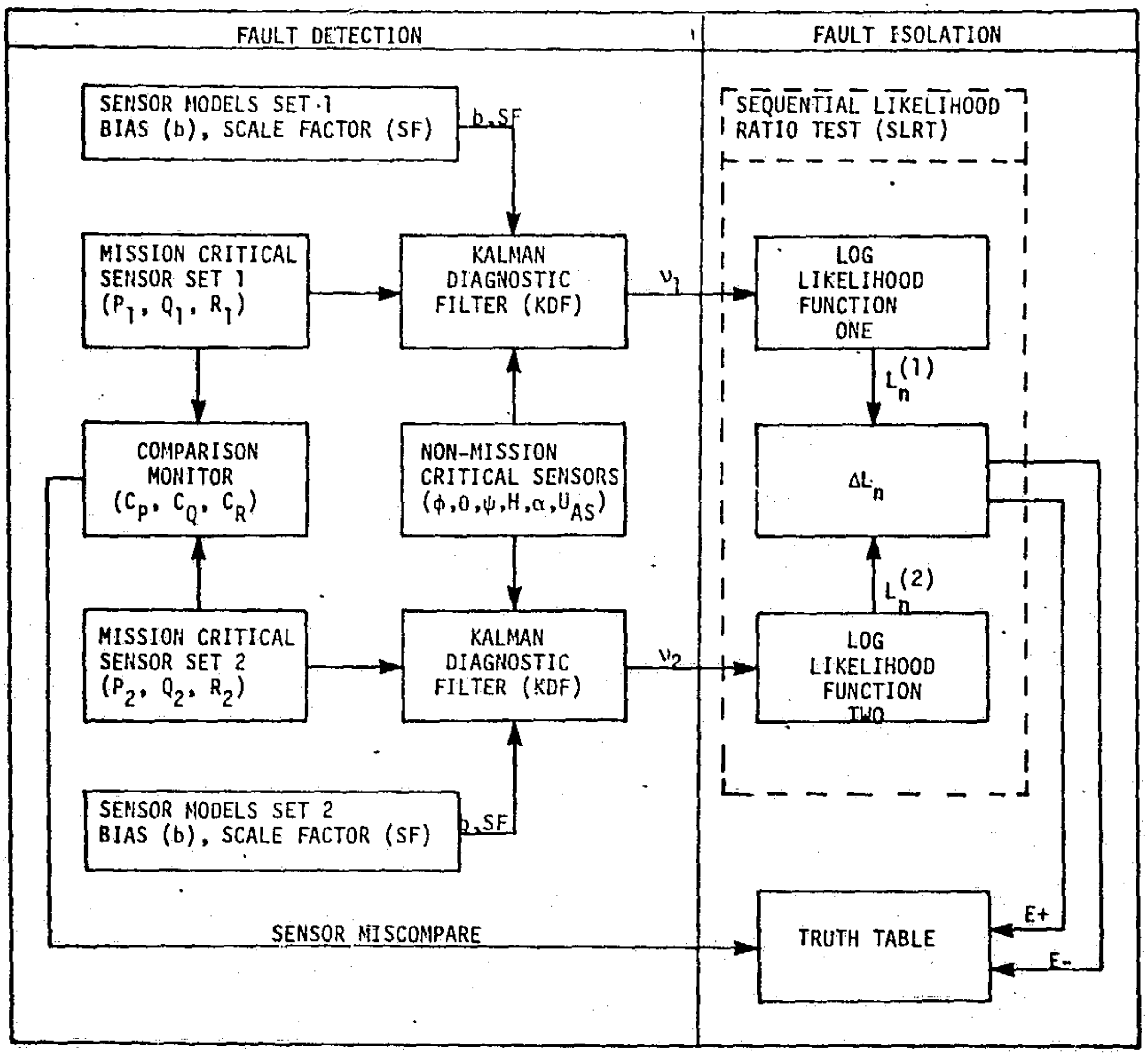

FIGURE A-32

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the presentation of the KDF design (Section 2.2). The comparison monitor detects faults between the two sensor set and issues a sensor miscompare to be used in fault isolation.

Once a fault has been detected, the fault isolation logic is activated. The Sequential Likelihood Ratio Test (SLRT) generates an error signal to be used in conjunction with the output from the comparison monitors. A truth table is generated to determine which sensor has failed. Section 2.3 will discuss fault isolation in more detail.

2.1 Sensor Models

An accurate representation of the sensor is essential. Sensor anomalies such as high frequency noise, bias, scale factor, and alignment play major roles in designing analytical redundancy schemes. Mission critical sensors (N_Z, N_Y, P, Q, and R) require extra attention. For off-line Analytical Redundancy design and analysis, flight test data should be used to approximate not only sensor high frequency noise, but unmodeled dynamics. Sensor fault models must be constructed after determining both the nature of faults and relative frequency of occurrence.

2.2 Diagnostic Filter Design

Concept II design is based on time domain synthesis techniques which employ Kalman filtering theory as the basic design tool. The initial design goals for Concept II are:

- n_y fault detection through lateral-direction equations of motion including aerodynamics
- improved U_{AS} fault detection
- α sensor diagnosis improvement by using wind gust estimation

- Body rate (P, Q, R) bias and scale factor estimation for reduced monitor levels.

The Kalman filter of Concept II is an observer which high passes the measurement vector, Y, and low passes the driving vector, U, when the plant matrix, A, is zero.

The Kalman Diagnostic Filter designs are based upon the following basic equations of motion:

$$N_Z = U\,(Q-\dot{\alpha}) + g \cos \phi \cos \theta) \qquad (1)$$

$$\dot{H} = U\,(\sin \theta - \alpha \cos \phi \cos \theta) \qquad (2)$$

$$\dot{\phi} = P + (Q \sin \phi + R \cos \phi) \tan \theta \qquad (3)$$

$$\dot{\theta} = Q \cos \phi - R \sin \phi \qquad (4)$$

$$\dot{\psi} = (Q \sin \phi + R \cos \phi) \sec \theta \qquad (5)$$

$$\dot{W}_g = a_g W_g + b_g n_g \qquad (6)$$

A generalized development of the discrete Kalman filter design for Equations (1) thru (6) is presented in Figure A-33.

A complete development of the Kalman filters and the residual index required to evaluate the Kalman filter performance is presented in Reference 1.

2.3 Fault Isolation

Two types of fault detection monitors were investigated. These are the standard multiple trip monitor and the Sequential Likelihood Ratio Test (SLRT). Monitor performance analysis showed that the SLRT of residual mean values is superior to the more standard multiple trip monitor. SLRT caught hardover failures almost by definition as no built-in delay is involved. SLRT also showed good soft failure identification characteristics, particularly scale factor changes that escaped the multiple trip monitor.

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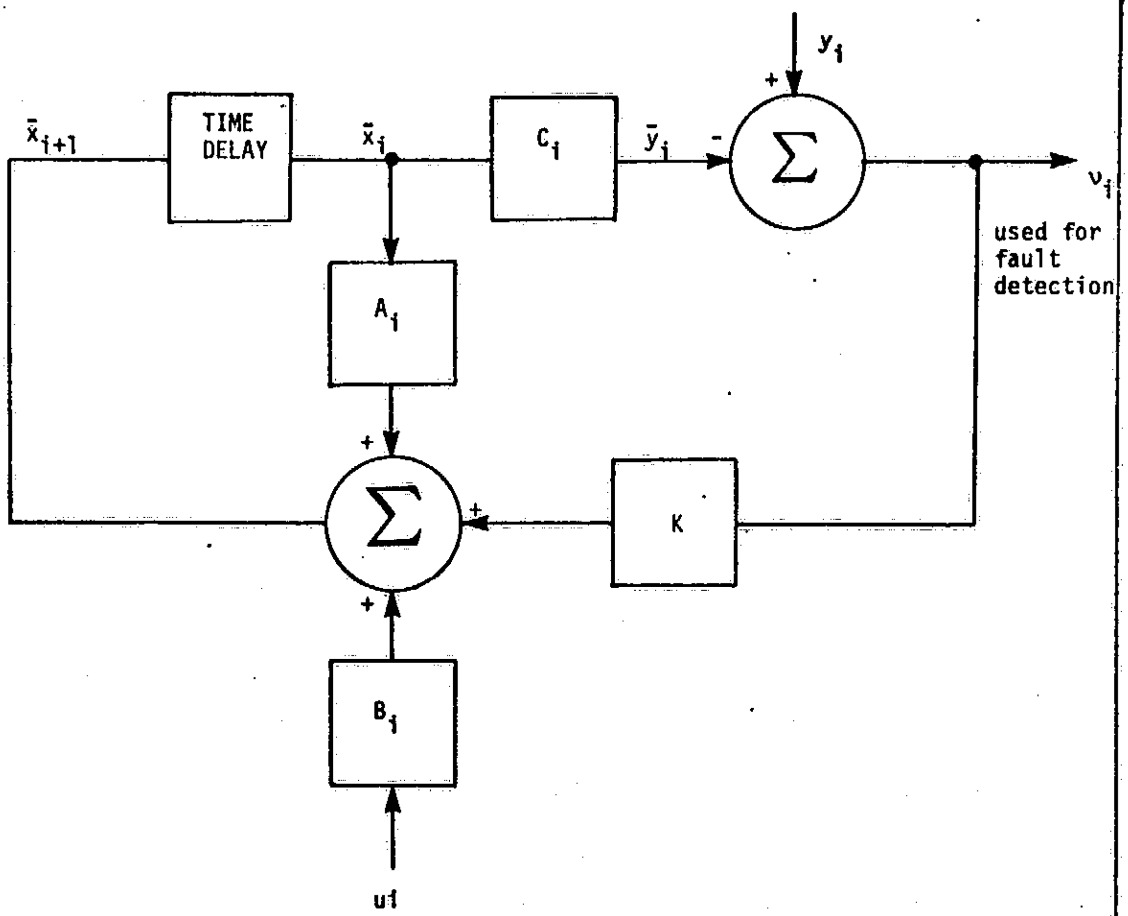
FIGURE A-33

GENERALIZED DISCRETE KALMAN FILTER STRUCTURE



FILTER EQUATIONS

Residual:	$\nu_i = y_i - \bar{y}_i$	(7)
Reconstructed: Output	$\bar{y}_i = C_i \bar{x}_i$	(8)
Estimate:	$\bar{x}_{i+1} = A_i \bar{x}_i + B_i u_i + K_{\nu i}$	(9)

FIGURE A-33

The SLRT is used with the comparison monitors to provide the isolation logic for the first failure of dual sensor pair. Upon a miscompare of two identical sensors, a log likelihood difference function of two filters is initiated.

$$\Delta L_n = L_n^{(1)} - L_n^{(2)} =$$

$$1/2 \sum_{i=1}^{n} \nu_i^{(2)} B^{-1} \nu_i^{(2)^T} - \nu_i^{(1)} B^{-1} \nu_i^{(1)^T} \qquad (10)$$

where $B = E(\nu\, \nu^T)$ ($n_r \times n_r$ covariance matrix).

$L_n^{(1)}$ is the log likelihood function for a Kalman filter(s) using sensor set (1)

$L_n^{(2)}$ is the log likelihood function for a Kalman filter(s) using sensor set (2)

Table A-10 provides the details of the truth table of Figure A-32. The truth table provides the required logic to isolate a failed sensor.

TABLE A-10

FAULT ISOLATION TRUTH TABLE

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FLAGS	FAULTS	NONE	P_1	Q_1	R_1	P_2	Q_2	R_2
COMPARISON MONITOR	C_P		1			1		
	C_Q			1			1	
	C_R				1			1
SLRT	E^+					1	1	1
	E^-		1	1	1			

3.0 Requirements

Honeywell estimated that to implement Concept II would require approximately 700 words of computer core. In addition, it is estimated that another 200 words of core will be required to support the failure injection module and data handling. Therefore, to establish the interface definition, software and hardware requirements and experiment cost, it will be assumed that sufficient computer core is available from the on-board Orbiter's computers. However, timing requirements will require further analysis to determine if the on-board computer can perform the required computation within a given pass.

3.1 Interface Definition and Software Requirements

Software must be developed to interface with the primary Shuttle system to monitor body attitudes, rates and accelerations, vehicle position and velocity, and all aerosurface positions. This information will be input to and processed by the Analytical Redundancy scheme to access the health of the Orbiter's systems.

Since failures cannot be expected to occur with high frequency in flight, the Analytical Redundancy scheme should include a failure injection module to provide in-flight simulated failures to test the Analytical Redundancy monitors. The software required to support the failure injection module is approximately 100 words of computer core.

A means of storing off-line the Analytical Redundancy results and/or the capability to down-link this information real-time must be available. The on-board core requirements for data handling is 100 words. A post-flight data reduction procedure to ascertain the success of the Analytical Redundancy scheme is also required. One man-year will be required to perform post-flight analysis of the data and evaluation of the feasibility of the Analytical Redundancy.

3.2 Hardware Requirements

If existing computer capability is sufficient, there are no hardware impacts.

3.3 Analyses Requirements

It will be necessary to develop the Analytical Redundancy system for the Shuttle with off-line programs and to verify this development with a man-in-the-loop simulator prior to implementation in the Orbiter. To minimize cost, as much use as possible should be made of existing off-line simulations and man-in-the-loop simulators (SAIL). The cost for the use of the SAIL facility is assumed to be absorbed by the Shuttle Program and is not included in this cost estimate. It is estimated that three man-years of analytical effort will be required.

3.4 Costs

Bases on the above requirements it is estimated that the Analytical Redundancy experiment will cost $510K.

4.0 Impact of Experiment on Mission

There is no impact on crew timelines since the Analytical Redundancy scheme requires no crew interface. Impact on mission safety is virtually eliminated since the experiment is total passive. That is, the Analytical Redundancy scheme will accept inputs from the primary system but will not provide any information to be used by the onboard system.

5.0 References

1. Cunningham, T. B. and Poynear, R. D., "Sensor Failure Detection Using Analytical Redundancy", 1977 JACC, San Francisco, June 1977.

2. Cunningham, T. B., et. al. "Fault Tolerant Flight Control Through Analytical Redundancy", AFFDL-TR-77-25, October 1977.

EXPERIMENT M: ALIGNMENT TRANSFER FROM ORBITER TO PAYLOAD BAY

1.0 Background and Objectives

A number of payloads like the Teleoperator Retrival System (TRS), Inertial Upper Stage (IUS), and Spinning Solid Upper Stage (SSUS) have inertial references which require alignment after the Orbiter has transported these vehicles into orbit. This experiment deals with demonstrating techniques for performing accurate alignment transfer from the Orbiter Inertial Measurement Unit (IMU) reference to an IMU located in the payload bay. In general, the techniques involve Orbiter rotational maneuvers, simultaneous measurement of attitudes, and subsequent comparison of the IMU readings to obtain an error matrix between the measurements.

The experiment was originally proposed by personnel at MSFC and McDonnell Douglas Astronautics Co. - Huntington Beach (MDAC-HB). At the time of the experiment suggestion, MSFC was considering the method for aligning the TRS IMU, and MDAC-HB was still involved in the competition for the IUS contract. Since then, MDAC-HB is no longer involved with the IUS. In addition, the TRS project has decided to perform a direct alignment transfer from the Orbiter to the TRS without any maneuvers and to accept the inaccuracies caused by structural deformation between IMU bases. This was done for the sake of quick development because the TRS has been scheduled for OFT-2.

Boeing, the IUS contractor, has decided to proceed with procurement of a star tracker system to be included as part of their system to be used as the IUS IMU reference. This was also predicated on the IUS development schedule, and the concern that the alignment transfer from Orbiter to

IUS IMU's may have some problems. However, Boeing has recommended that an alignment transfer scheme (particularly an approach developed by IBM) be developed for a block update of the "production" IUS's which would eventually eliminate the need for the star tracker system. With this in mind, the objectives of this experiment are to:

- Develop the required flight software for one or more transfer schemes.
- Demonstrate the feasibility and adequacy of the in-flight procedures for each scheme.
- Perform a trade-off between schemes and recommend one as the best overall operational system.

2.0 Feasibility

The basic idea behind transferring Orbiter IMU alignment to an IMU in the payload (P/L) bay is the following. The Orbiter performs rotations about two or more spatial axes. The rotations are jointly sensed by the Orbiter and the P/L IMU, affording common lines of reference in inertial space. For the Orbiter, the reference directions are expressed in the Orbiter's inertial coordinate frame. For the P/L IMU, the same reference directions are expressed in its unknown inertial frame. Since the reference directions are common to both the Orbiter and the P/L IMU, it becomes a simple matter to compute the orientation of the P/L IMU's unknown frame with respect to the Orbiter's frame.

Boeing recently performed a comparison of on-orbit alignment methods as potential candidates for the IUS system. Two transfer schemes were evaluated, one developed by IBM-Houston, and the other by TRW. The IBM scheme, which is described in JSC Report - 13838, "IUS Pre-Release Alignment", was reported to have some advantages over the TRW method and hence will be described in the following paragraphs as an example of feasibility.

The IBM scheme performs the Orbiter Star Tracker/IMU alignment procedure and the alignment transfer procedure simultaneously as follows:

(1) Orbiter takes alignment sighting on star #1, using either of its two star trackers, recording star tracker and Orbiter IMU gimbal angle measurements. Attitude from the P/L strap-down IMU system is simultaneously recorded in a computer located in the P/L or sent to the Orbiter computers.

(2) Orbiter rotates 180 degrees about star #1 line of sight (LOS) and then takes another alignment sighting (same star tracker) on star #1, again recording star tracker and Orbiter IMU gimbal angle measurements. P/L IMU attitude is again simultaneously recorded.

(3) The first and second set of Orbiter measurements are averaged, removing the body-fixed sensor misalignment effects from star #1 measurements. In addition, the eigenvector (eigenvector #1) associated with the 180-degree rotation is computed on both the Orbiter and the P/L system. The eigenvector essentially represents the axis of rotation.

(4) The Orbiter selects alignment star #2 and repeats (1), (2), and (3), using either of its two star trackers. This yields an averaged star measurement on star #2 and eigenvector #2.

(5) The Orbiter measurements, expressed in the Orbiter's IMU stable member inertial coordinate system, and the P/L IMU measurements, expressed in the P/L inertial coordinate system (orientation unknown at this point), are jointly processed. (This joint processing could be performed in either

the Orbiter or P/L computer. However, for an experiment, it would be recommended for the P/L computer, since this would have the least effect on the Orbiter system . The end result of the processing is a 3 x 3 matrix transformation that relates the P/L unknown inertial coordinate frame to the desired P/L inertial navigation coordinate frame (such as the M50 coordinate frame). Applying the matrix transformation to P/L body attitude (one shot computation) constitutes the P/L IMU alignment.

IBM shows that if the Orbiter IMU alignment and the Orbiter transfer maneuvers take place separately, then the alignment transfer error will be the sum of the Orbiter alignment, Orbiter IMU, and the payload IMU errors. If the alignment is performed as described in the previous paragraph, then only the Orbiter star tracker errors and the payload IMU errors impact the alignment of the payload system. If the alignment stars are 90 degrees apart, the per-axis alignment error of the IMU assumed for the IUS is 1.6 min (3σ) as shown in JSC-13838. If the Orbiter star tracker measurements are restricted to the central 4 x 4 degree field of view (full field of view is 10 x 10) degrees), then the per-axis error is 1.0 min (3σ). The average per axis alignment error degrades by the factor $K = (1 + 2\csc^2 A)^{1/2}/\sqrt{3}$, where A is the subtended angle between the alignment stars. When A = 90 degrees, K = 1. For 60 degrees $\leq A \leq$ 120 degrees, $K \leq 1.1$.

One of the prime concerns of the IUS community in performing the above type alignment is variations of P/L orientation relative to the Orbiter. Accurate Orbiter/P/L alignment transfer is predicated upon the assumption that the Orbiter and P/L rotate as a single unit during alignment transfer maneuvers. Changes in the P/L's navigation base relative to the Orbiter's navigation base, from measurement to measurement, will introduce errors into the alignment transfer process. Measurements of these potential movements can be made during initial IUS or SSUS flights as part of the "Alignment Variation - Reference to Cargo Bay" experiment.

3.0 Requirements

The basic requirements for this experiment are the following:

- Obtain a strapdown IMU system and associated general purpose computer for mounting in the payload bay.
- Develop the required software for the P/L computer and the Orbiter computers (both GN&C and SM).
- Provide support engineering for interfacing the P/L IMU/computer in the Orbiter.
- Develop mission timelines and procedures.

IMU/Computer

Potentially several systems can be considered for the experiment which would result in a minimum cost to NASA since they are already being developed for NASA vehicles. This would include systems being developed for the IUS, SSUS, Teleoperator, and Delta vehicles. All of these vehicles employ a strap down IMU and associated general purpose computer. The ready availability of units from these programs will probably dictate the system employed in the experiment. Prototype, qualification and production units can be considered for the experiment. Discussions with personnel involved in these programs is summarized in the following:

IUS - The inertial measurement system that will be used on the IUS consists of a Hamilton-Standard strapdown gyro package and a Delco MAGIC 362 computer. Boeing will start receiving production units from Hamilton-Standard in early 1980. There are no spares ordered, each package being assigned to a vehicle. At the present, the qualification unit is committed through mid-1980. Two other pre-production units on order have been assigned to Martin and a Boeing laboratory and are unavailable. Hence, it appears that the availability of IUS inertial system hardware precludes doing the experiment with this equipment until late 1980 or early 1981. Present schedules indicate three IUS's will have already been flown by Jan. 1981.

SSUS - The systems for this vehicle, being developed by MDAC-HB, will be fully defined in the early summer of 1979. Delivery schedules and availability of units are not yet defined.

TELEOPERATOR - This vehicle is going to use extra Viking hardware. At present, only three IMU packages and two computers are available for the program. If any other units materialize, the project office would feel more comfortable if they are committed to the Teleoperator program.

DELTA - In the near future, a strapdown IMU developed by MDAC-HB along with the DELCO MAGIC 350 computer will be used as the inertial measurement system for the Delta booster. MDAC-HB has contracted to build approximately 20 of these IMU's (called DRIMS) for NASA/GSFC at the rate of one per month. Production has started, and there are two DRIMS units completed, the engineering Development Test Unit, and the Qualification Unit, which could be made available depending on the experiment need date and what the Delta program needs are. It is also possible that a production unit could be diverted from Delta for the experiment in the March 1979 time period.

In summary, it would be desirable to use an IUS system in the experiment since the IUS program would incorporate the alignment scheme if successful.

However, earlier availability of the Delta system may dictate the use of this system in order that feasibility is confirmed early in the IUS program, and maximum cost saving can be realized by eliminating the need for the IUS star tracker.

Software

Assuming that the alignment equations would be processed in the P/L computer, Orbiter data would flow from the Orbiter GN&C flight computer to the Orbiter Systems Management (SM) flight computer to the P/L flight computer. A simplified block diagram is shown in Figure A-34. For the IBM method, a small software program would be required in the SM computer to control the alignment transfer. This program would accept keyboard inputs from the crew, notify the P/L computer of the impending alignment, monitor

ORBITER/PAYLOAD SOFTWARE INTERFACE

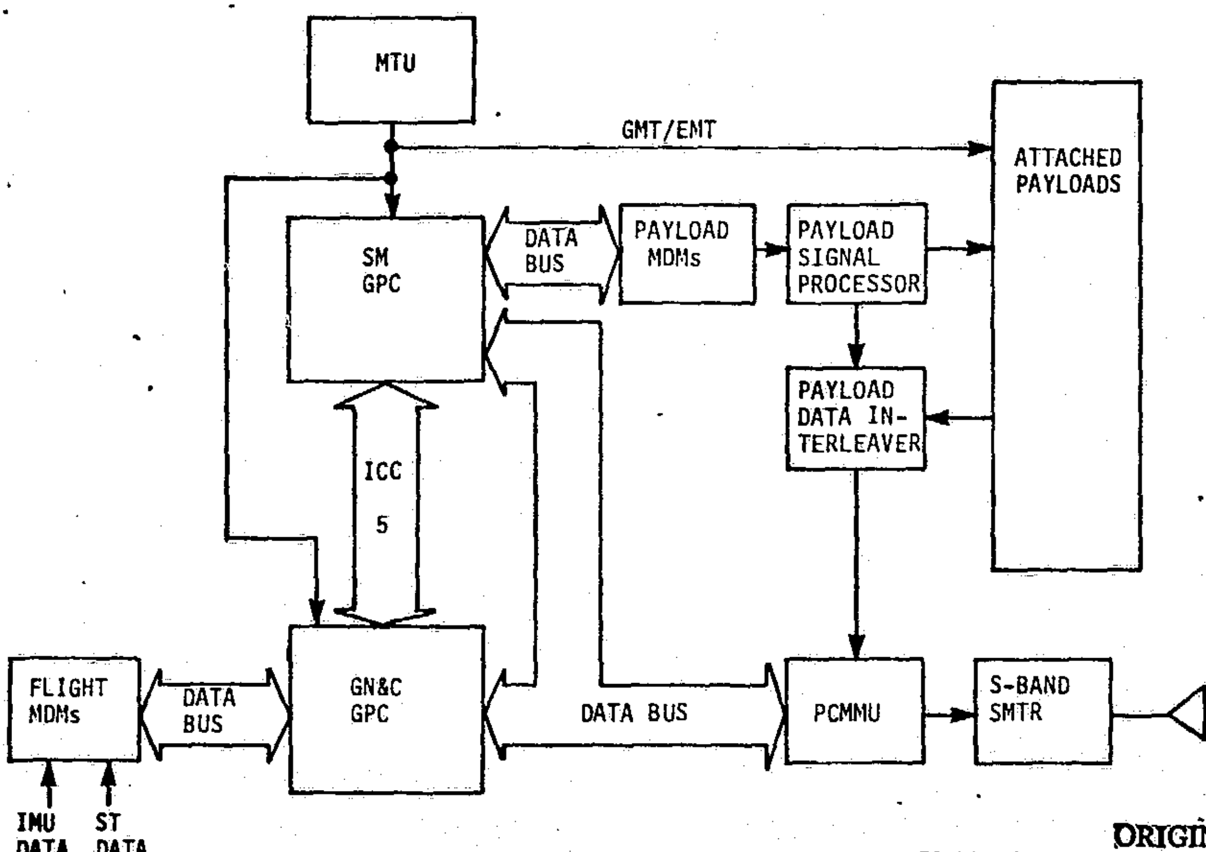

FIGURE A-34

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the GN&C computer for the start of each data collection period, signal the P/L computer to take data at the appropriate times, receive and transfer Orbiter data to the flight computer, and provide CRT displays for crew control.

The only new software required in the GN&C computer would be a flag in the COMPOOL data base, set and reset at the initiation and completion of each Orbiter data collection period (lasting 3.2 seconds). The SM computer would monitor this flag every 160 ms, when in the P/L IMU alignment mode. IBM personnel estimated that approximately 100 words of core are required in the Orbiter computers.

It is assumed that system software and software required for the strapdown IMU will be available in the P/L computer regardless of the system selected. A range of 500 to 2000 words have been estimated for the P/L computer alignment equations. Cost of this software will depend on the selected computer and the software contractor.

The IBM mechanization approach has no critical timing requirements between the Orbiter and P/L inertial systems. Data sent from the Orbiter system is time tagged, and the P/L systems has access (Figure A-34) to the Orbiter master timing unit such that the respective time bases will be significantly less than a millisecond apart.

Analyses

It is estimated that approximately two man-years of effort will be required to do an off-line analysis of perhaps two different alignment transfer schemes. This would result in the equation definition and substantiating analysis.

Interfaces

As shown in Figure A-34, the digital data interfaces for the payloads exist. However, the interfaces with the MDM's must be shared with other experiments, as all attached experiments are hardwired into the system. Reservations approximately one year in advance must be made to obtain the required data interfaces. Details of these and other interfaces such as electrical power, environmental control, data systems, etc., are defined in ICD 2-19002 "Shuttle Orbiter/Cargo Standard Interfaces".

After selection of the P/L IMU/Computer system, design of the P/L bay attachment mechanism is required. An alternative is to obtain space on one of the OFT pallets. If this is feasible, it will then be the responsibility of the pallet developers to allocate part of their support system budget to the P/L IMU/Computer.

Mission Procedures

Timelines and procedure development on the Shuttle Mission Simulator (SMS) will be required. A preliminary operational sequence for the IBM alignment scheme would include the following:

- Crew commands Orbiter IMU in-orbit alignment via the GN&C computer.
- Before the first star sighting is taken, the SM computer is placed in the P/L system alignment mode by keyboard command.
- Thereafter, four collections of Orbiter and P/L data sets take place, as the Orbiter maneuvers and takes four star tracker sightings (on two stars) as previously described. Data would be automatically taken and transferred to the P/L flight computer.
- The SM computer would provide appropriate outputs to the CRT displays for crew monitoring of the alignment process.

Cost

Based on the above requirements, the Orbiter GPC software cost will be approximately $30K. The cost of the payload computer software development was based on an estimate from MDAC-HB of approximately $100/word for the Delco computer. For the 500 to 2000 word estimate, this would amount to $50K to $200K. Two man-years of analytical effort amounts to $120K.

4.0 MISSION IMPACT

A small amount of RCS propellant will be required for the on-orbit attitude maneuvers. Depending on the desired maneuver rate, approximately 100 to 150 pounds of propellant will be required.

Approximately 30 minutes of time during a given orbit will be required for the alignment procedure and the data collection.

EXPERIMENT N: ALIGNMENT VARIATION - REFERENCE TO CARGO BAY

1.0 Background and Objectives

This experiment was suggested by Rockwell International, Space Division. A method was proposed whereby the amount of alignment error between the Orbiter nav base and an experiment located in the payload bay could be measured.

The Orbiter on-orbit FCS will be used to orient the spacecraft to point various payloads to their desired targets. The RCS deadband can be set to maintain attitude within $\pm.05°$ of the IMU reference. This reference will be quite accurate assuming a recent star tracker alignment where IMU drift (about $.035°$/hr 1σ) may be neglected. Star tracker errors and star tracker-to-IMU alignment uncertainties will still be present (about $.028°$ 1σ).

Problems arise in maintaining accurate payload pointing because of Orbiter structural deformations caused by earth-to-space environmental changes. Although pressure changes and zero g may have some effect, the largest contributor is expected to be non-uniform thermal conditions. Unfortunately, these structural deformations due to non-uniform thermal effects will vary as a function of time at a particular solar aspect. The structure and environment have been modeled and the deformations simulated primarily to determine their impact on P/L bay door operations. Using this model, experiment pointing errors of up to $\pm 2°$ have been postulated.

These alignment errors introduced between when the instrument is aligned preflight and on-orbit operation obviously impact payload requirements. A section in the payload planning information questionaire addressing pointing accuracy requirements states that if greater than $\pm 2°$ accuracy is required other provisions must be made. These include either designing a star tracker/platform type control system into the payload or interfacing it with a NASA-provided experiment pointing module. Both would result in substantial weight and cost penalties, possibly making some experiments unfeasible. But if the estimated uncertainties are conservative, all this may be unnecessary. Similar analysis for Skylab instrumentation was discovered conservative, in some cases by an order of magnitude. An OFT experiment to measure alignment variations could answer these payload requirements questions.

In order to choose the best approach for measuring these pointing errors, some attention should be paid to how the payloads are attached and how the Orbiter structural deformations affect alignment (Figure A-35). This is necessary in order to draw conclusions about misalignments for future payload configurations.

The payloads are attached to the Orbiter main longerons and keel. Attachment fixtures can interface and be secured to these structural members at almost any point along the member. The pallets or payloads themselves have pins which are inserted into holes in the attachment assemblys. These pins are horizontal for the longerons and vertical for the keel attach points and may slide in and out of the holes a small amount. In this manner, loads in the X-Z plane are carried by the longerons and loads in the X-Y plane are supported at the keel.

The payload itself will be load bearing and thus can affect the amount of deformation present with that payload configuration. Furthermore, redundant attach points will be used on some payloads to provide additional

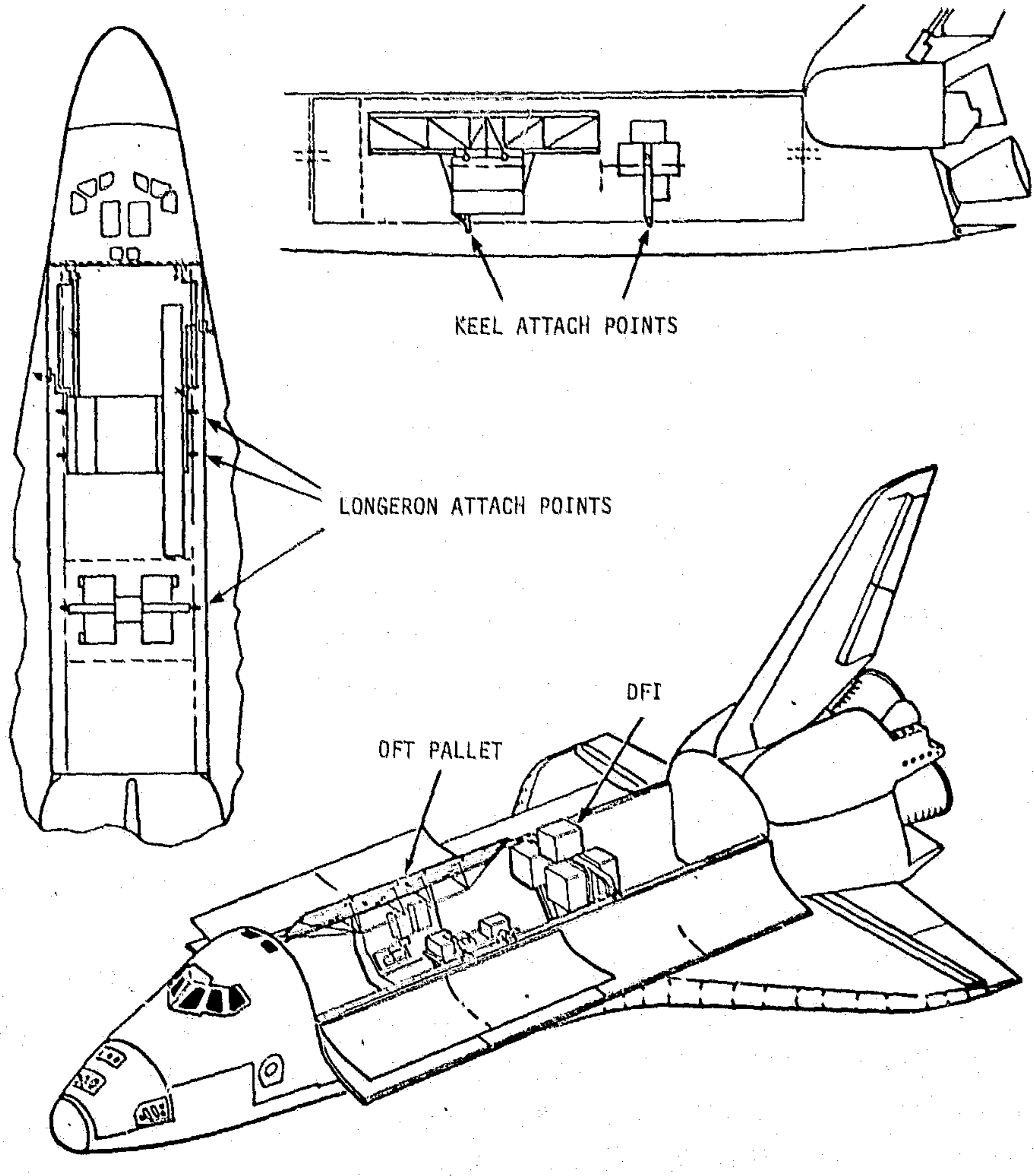

FIGURE A-35

EXAMPLE PAYLOAD INSTALLATION (OFT 2)

load paths. All this leads to the conclusion that deformation data taken on the longerons and keel with one payload configuration cannot be directly applied to another configuration. However, this data can be used to validate the simulation models which predict misalignments in the integrated configuration. Feedback of this sort will determine whether excessive conservation exists in the estimates and whether payload hardware or software requirements may be relaxed. This is our objective.

2.0 Feasibility

Several approaches in instrumenting this experiment have been proposed by the experiment suggestor, the study contractor, or the NASA JSC. They include:

1) Star Tracker
2) Photogrammetry
3) Laser Techniques
4) Crew Optical Alignment Sight (COAS)
5) Theodolite

The suggestor proposed that star trackers (Orbiter test units or spares) mounted on an OFT pallet be used for this purpose. Differencing the orientations of the Orbiter star trackers (mounted on the nav base extension) and that of the star trackers in the payload bay would certainly give very accurate misalignment information. This approach was originally proposed by the Orbiter star tracker vendor in connection with a NASA conceptual study of this type done four years ago. At that time it was decided that limited information would be obtained since only that pallet's misalignment would be measured. The attach point translations responsible for the error would not be uniquely defined and application of the data to other payload configurations could not be made.

The most elegant (and costly) approach proposed by the NASA study involved mounting several cameras in the payload bay looking in different directions. The cameras would have to be qualified for the payload bay environment and would have power and data bus interface requirements. These precision cameras would resolve angular displacements on the order of a few arc seconds. Using this photogrammetric technique and combining data from more than one camera, the deformations could be determined in three dimensions postflight.

One of the laser techniques cursorily examined used holographic interferometry. Another involved splitting the laser's output into many beams and projecting them onto targets attached to the points of interest in the bay. The displacement could be read directly off of the target.

During investigation of the feasibility of mounting a theodolite adjacent to the aft crew station payload bay window, it was discovered that a COAS (crew optical alignment sight) was recently baselined for use in this fashion in connection with payload bay door operations. The COAS will be fixed to the window sill with a suction mount and can be oriented to align its scribe marks with any payload reference. On OFT 1, one door will be shut and then the COAS marks aligned with the edge intended to mate with the other door. Any warpage of sufficient magnitude to prevent successful closing and latching of the door can be detected. This instrument has no magnificaton but can measure angles within its field of view to about .1° accuracy. For measuring the types of payload misalignments of interest here, an instrument of superior accuracy is desireable.

The theodolite concept previously alluded to is felt to be the most feasible approach when the required accuracies are considered. The theodolite and installation are illustrated in Figure A-36. It is hoped that alignments could be measured within .1°. Although theodolites of the most modest design would give sufficient accuracy, the better designs read to the arc second and can be estimated to the tenth of an arc second. If three targets five feet apart were fixed to a payload forty feet away, the translations in the Z-Y plane could be measured to less than 3 thousandths of an inch assuming 1 arc second error. Translations along the line-of-sight could be measured by finding the change in angle between the targets to an accuracy of less than 40 thousandths of an inch. Alignment changes in the Z-Y plane would be measurable to roughly 20 arc seconds. Misalignments in the X-Y or X-Z planes are less discernible and only about .5° accuracies can be guaranteed. Shorter distances to the array, larger distances between targets, better than 1 arc second theodolite accuracy (probably obtainable), or a four element array would be necessary to achieve the .1° goal.

Inferring the misalignment between the Orbiter nav base and a payload is dependent upon two things: First, the theodolite alignment with respect to the nav base must be known because its mounting jig would be expected to be subject to some deformations. Its alignment error may be determined by taking star sightings at the same time as the payload readings. Second, once the nav base to theodolite transformation is known, the theodolite to payload alignment is all that is needed. This could be measured absolutely if the distances between targets are known. Therefore, the targets cannot move appreciably with respect to each other due to payload warpage. This would be insured by connecting them with a rod of a material of low coefficient of thermal expansion. An array of three targets would be fixed to the ends of a "T" made of a material such as "Invar" whose expansion rate

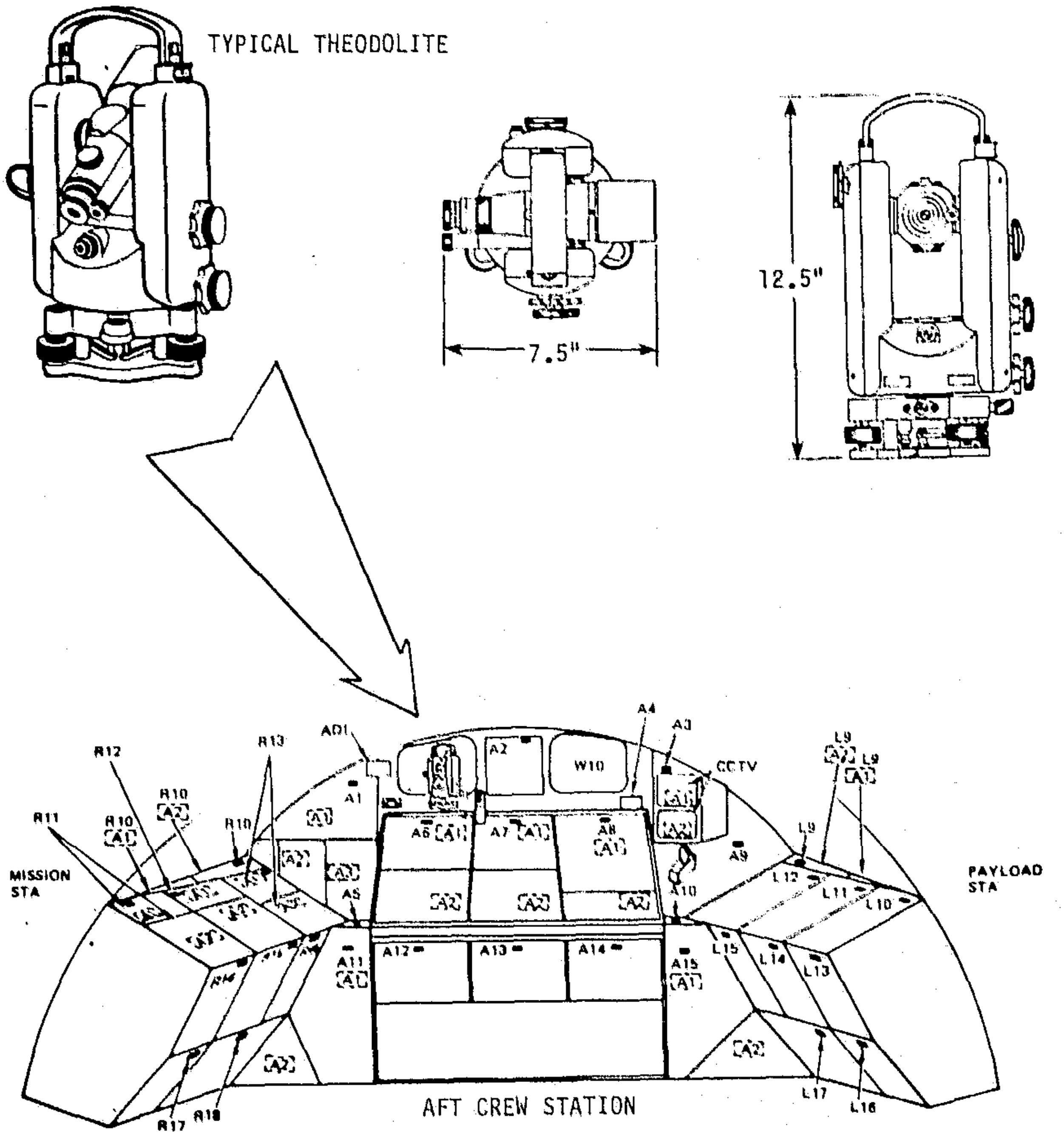

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FIGURE A-36

THEODOLITE AND ORBITER INSTALLATION

is 1/30 that of carbon steel. An "H" would be required for four elements. This frame would be fixed to the payload at one point and "slip" mounted at others in order that the payload may warp without affecting the targets.

This approach to measuring misalignments would not yield any more information than the star tracker approach but its data could be used real time if desired by feeding the theodolite star and target readings into a hand-held calculator and obtaining the misalignment angle. The correction angle could be applied via keyboard entry to the on-orbit FCS to adjust for the pointing bias. This is a potential operational application.

To accomplish our objective of refining the misalignment prediction model, the arrayed target approach wouldn't be necessary. The main thing of interest is the change in reading from preflight of single targets located along the members to which payloads will be attached. However, the theodolite reference translations would have to be backed out postflight from the readings taken on-orbit because the distances between targets may vary.

The basis for feasibility of the theodolite approach lies in the fact that these techniques have long been applied and proven in connection with numerous terresterial applications. The hardware is lightweight, self-contained, and simple to learn to operate and use. The postflight reduction of data into target displacements involves simple trigonometry. However, the preflight and postflight integrated structural modeling to determine optimum target placement and to apply the experimental data to deformation model refinement could become complicated. Structural analysts contacted believe this also feasible and the data would lend itself to other deformation concerns.

The other potential benefits of having a 30X instrument located at this station are many. The shifting of payloads could be monitored. This information is required to fully establish the feasibility of the "Alignment Transfer from Orbiter to Payload Bay" experiment. It could also be used as a tool to investigate problems in the payload bay such as possible damage during ascent or involving deployment/stowage of antennas, solar cells, or remote-sensing equipment.

3.0 Requirements

The basic requirements are:

1) Mechanical interfaces
   - Window sill
   - Targets in bay
2) Hardware
   - Mounting jig
   - Theodolite and box
   - Targets
3) Software development
4) Postflight analysis
5) Crew training

The only Orbiter interfaces required are mechanical since the theodolite is manually operated and readings could be recorded on the crew's tape recorder. The theodolite contains a small battery pack to illuminate the scale for reading.

A jig must be manufactured and fixed to the window sill in order that the instrument may be easily secured when removed from its box and readied for use.

The targets should be phosphorescent and adhesive backed so that they can be applied to tabs which must be oriented properly and secured to the points of interest. If the alignment measurement approach using the target array and Invar rods is desired, interfacing with the pallet(s) or payload(s) is required.

The total weight of thé experiment hardware is estimated at between 15 and 20 pounds. This includes the theodolite (10 pounds), stowage container, mounting jig, and targets. Recording equipment is already aboard. The total hardware costs are estimated at between 10 and 15 thousand dollars. Commercial theodolites can be obtained for about 6 thousand dollars.

No Orbiter software requirements exist unless the real-time angular bias correction technique is used. The ground software for postflight reduction of the observed phenomenon is simple but application to the deformation model refinement is largely an unknown. This should be fully addressed by experts in that field before a decision to baseline the experiment. Requirements for software develoment for postflight reduction and model application along with additional structural analysis preflight are the source of the largest impact of this experiment. Roughly one man-year would be required.

Preflight training of one crew member in the installation, operation, and recording of data could be accomplished in just a few hours.

4.0 Mission Impact

Besides the minimal weight impact, no other effects on the mission have been identified other than crew timeline. OFT 4 appears promising due to the thermal conditioning experiments planned. The payloads on that flight will include an OFT pallet and DFI.

At four times during the mission corresponding with peak thermal deformation predictions, the crew member would be required to take star sightings and sightings to about ten targets. The theodolite would be slewed to a reference angle corresponding to the preflight target position and the target will be within its field of view. About 30 seconds is required to slew, center the cross hairs, and record the reading for each target. Possibly ten minutes would be necessary at initial setup and stowage and about five for each set of star/target sightings between.

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APPENDIX B

BRIEF DESCRIPTION OF REJECTED EXPERIMENTS

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BRIEF DESCRIPTION OF REJECTED EXPERIMENTS

The following paragraphs give a brief description of the suggested experiments that were eliminated from further study by the contractor and the reason for the rejection. This will perhaps provide any future reviewer enough information to:

1. Determine whether it is worthwhile to pursue additional information because the experiment has more value to the reviewer than judged by the study contractor.
2. Determine that a detail study of the suggestion may be worth performing even through an Orbiter flight is not required to establish its feasibility.

The number associated with the experiment is that shown in the summary of Section 3.

3. Optimal Control Blending - Rejected because adequate data was not available to study contractor.

This new technique is based upon using all of the control actuators and surfaces in a totally integrated and coordinated manner. This approach would be implemented in the following manner: For example, consider that the vehicle has 7 control surfaces and the autopilot is controlling three angular motions and one vertical motion. The equations of the response would be written in the following matrix form:

(4 x 7)	(1 x 7)	=	(1 x 4)
vehicles effectiveness matrix	surface deflection		angular torques vertical force

or $CD = E$

For the normal control system application, matrix C is deterministic based upon the current operating condition and E is the required control effort for proper vehicle operation. The solution for D defines the required deflections. Unfortunately, C is not a square matrix and inverting it to solve for D is not straightforward. An approach called the pseudo-inverse can be used to solve for D in the following manner:

$$D = \frac{C^T E}{C C^T}$$

This solution has the property that D is generated to give the minimum mean value of command.

The onboard computer can readily store the data required to generate the C matrix as a function of flight condition and actuator position. The excess of actuators over control outputs is used deliberately to expand the system effectiveness and improve system reliability. This same approach is used to control skewed momentum exchange devices on spacecraft with very good performance predicted. This system implementation approach offers performance advantages with the primary cost being additional software for operation. Mission flexibility and reliability improvements are also benefits of this approach.

4. Blended Use of Ailerons and Rudder for Improved Lateral/Directional Control - Suggested by a number of companies. Rejected because only adequate data available was the MDTSCO scheme which has been incorporated into the baseline system. (If MDTSCO scheme hadn't been baselined, experiment would still have been rejected since simulations to prove feasibility would be adequate without actual Orbiter flight.)

Preliminary studies have indicated that an entry flight control scheme could be devised to extend all aero-surface lateral/directional flight control to higher angles of attack, where reaction jet augmentation is presently utilized. Uncertainties in pre-flight estimates (based on wind tunnel data) of certain aerodynamic coefficients forced the use of reaction jets for lateral/directional augmentation. A reduction in the utilization of reaction jets during entry could provide significant savings in reaction control system fuel, as well as reduce undesirable effects of firing jets in the atmosphere.

A modified configuration for the entry lateral/directional flight control system for the Orbiter vehicle suggested by MDTSCO offered the following advantages:

1. It had the same configuration for all entry flight regimes.
2. It had better performance and trim capability in the Mach region from Mach 5 to Mach 1.5.
3. It had more capability to handle aero variations.

Once the aerodynamic characteristics of the Orbiter has been adequately defined, this flight control system could be refined to:

1. Further extend all aero control capability
2. Reduce RCS requirements throughout flight regime and thus increase payload capability
3. Provide invaluable insight into the development of control systems for future vehicles.

6. <u>Bending Mode Suppression</u> - Rejected because adequate data was not available to study contractor.

The coupling between the varying bending modes and the flight control stability loops is a subject of uncertainty. Consequently, flight control experiments directed toward reducing this model uncertainty are worthy of investigation. Advanced methods which may be investigated include the use of adaptive control and learning control techniques. For example, one technique proposed in the past is the use of variable complex zeros placed in the neighborhood of a variable body-bending pole. The adaptive controller tracks "movements" of the body-bending poles and causes the complex zeros of the digital filter to follow the poles, thereby reducing their residue and the pole's contribution to the dynamic response of the body-bending mode. Another attractive technique, termed a "learning control system," uses flight control feedback sensors placed within the vehicle at different body stations. The learning control system seeks to select sensors at body stations that reduce the coupling from those body-bending modes which become a threat to stability.

9. <u>Control of Large Space Structures</u> - Rejected because it didn't fall into general category of study experiments.

This suggestion by LEC (Houston) dealt with control of on-orbit solar power systems. Two control problem areas were mentioned: 1) control and aligning solar array towards sun and 2) pointing microwave antenna (power transmitter) at ground collector. There was no concept at the time of the suggestion of any possible Orbiter experiment.

10. Criteria for FCS RCS/AERO Work Load - Rejected because adequate concept was not available to study contractor.

The objective of this task is to develop a procedure for monitoring and displaying to the pilot the status of the control system. The pilot must have sufficient information to ascertain the flight control system workload so as to avoid creating an intolerable situation for the system and to allow a reduction in system stress when appropriate. From the pilot's viewpoint, it is easy to specify how hard he works by requiring various levels of handling qualities in the specified flight envelope. But, it appears that with highly augmented systems, this is not sufficient. The vehicle can exhibit Level 1 handling qualities right up to the point where loss of control results. For example, if while exhibiting Level 1 handling qualities the control surface shows severe rate limiting and the yaw jets are on a 75 percent duty cycle, margin is actually small and the vehicle is on the ragged edge of going out of control. This may not and probably will not be apparent to the pilot. Thus, there needs to be some way to specify how hard the flight control system has to work.

11. Shuttle Pointing With CMG's - Rejected because feasibility of this was proven on Skylab. No reason for an experiment.

12. Remote Manipulator and CMG Control Blending - Rejected because adequate data was not available to study contractor. Also reasoned that if problems develop with RM deployment, CMG's may be a potential fix for the problems, however this would not classify as an experiment.

13. Closed-Loop Arm Control - Rejected because adequate data was not available to study contractor.

Current remote manipulator has open loop automatic and manually directed control capability. The suggestion was to employ a closed-loop design with arm effector position sensing.

20. Strake Vortex Visualization - Rejected because it didn't fall into general category of study experiments.

Previous flow visualization studies on strakes have been limited to strakes with small leading edge radii and at speeds up to about Mach 2 to 3. Since the Orbiter has a strake with a large leading edge radius, its flight flow characteristics may be significantly different from wind tunnel results due to Reynolds number effects. Therefore, the current flight state of the art data base is limited to low supersonic speeds and small leading edge radius strakes. The approach would be to inject a marking fluid at or near the strake leading edge at several locations. Camera(s) located within the fuselage and/or vertical tail would then photograph the vortex development and the flow over the wing.

21. Estimation of Orbiter Inertial Properties With RM Deployed - Rejected because it didn't fall into general category of study experiments. Also reasoned that this is a mainline Orbiter function.

22. Synchronized Mid-Value Select Averaging - Rejected because actual Orbiter flight is not required to prove feasibility.

Suggestor was told that the Shuttle system can experience transients when the mid-value selector (MVS) selects a new LRU or fault detection/isolation

(FDI) rejects one LRU and brings another into the MVS calculation. They feel this could be remedied by going to a scheme which uses all four LRU's and selects lower mid-value. In addition, the scheme they are familiar with used equalization to drive all LRU's towards the selected value. Hence switching transients were minimized. Again, this is probably not in the experiment classification. If switching transients cause objectionable pilot comments, that is only one possible way it can be corrected.

23. Voting With LRU Not In Common Location - Rejected because adequate data was not available to study contractor.

This is a big concern in military aircraft, where the LRU's are purposely separated in order to reduce vulnerability to enemy fire. This experiment would investigate and demonstrate some potential solutions of problems created in the redundancy management logic by using non-common signals from separated LRU's. This perhaps could be demonstrated by using the Orbiter and ACIP rate gyros.

24. RCS FDI Using Onboard Vehicle State Estimates and/or Release Plane Switching Lines - Rejected because actual Orbiter flight is not required to provide feasibility.

Work at Draper Laboratories and at LMSC indicates that detection and isolation of reaction jet failures can be accomplished using onboard vehicle state estimates and/or phase plane switching lines. The algorithms are mechanized in software, using available inertial sensor data. Such a technique for the Orbiter would greatly reduce the complexity of the hardware/software interface, which currently involves sampling and processing of data from a multitude of temperature, pressure, and valve command sensors.

(See: Journal of Spacecraft and Rockets "Maximum Likelihood Failure Detection Techniques Applied to the Shuttle RCS Jets", dated February 1976).

26. Real-Time Trajectory Generation - Rejected because adquate data was not available to study contractor.

This scheme would simulate the vehicle dynamics in real time onboard in response to pilot inputs. The program would include simulations of vehicle dynamics, controls, airframe, etc. Various parameters such as accelerations, rates, surface deflection or any intermediate point could be compared to actual measured parameters as a means for detecting failures and possibly reducing number of required LRU's in a given set. Experiment objectives would be to determine needed complexity of program, need for periodic updates, initialization requirements and actual flight tests. Somewhat related to "Analytical Redundancy" experiment.

27. Flat Surface Display Technology - Rejected because actual Orbiter flight is not required to provide feasibility or pilot acceptability.

The suggestor states, "The advances in flat surface display technology within the next few years promise methods which will replace the current CRT display technology. The flat surface display technologies are better matched to the new microprocessor technology than high-voltage CRT displays."

28. Advance Display Design - Rejected because actual Orbiter flight is not required to prove feasibility or pilot acceptability.

A stereo display concept was proposed for optimizing the pilot role. Symbology would be similar to PAFAM (Performance and Failure Assessment Monitor) used on DC-10 which has a runway and horizon presentation which grows in size during final approach. It does not include a roll degree of freedom.

29. System Monitor Display - Rejected because actual Orbiter flight is not required to prove feasibility or pilot acceptability.

This is similar in concept to the PAFAM which was developed by Douglas Aircraft Co.

30. Helmet Sight, Display, and Pointing in Zero G - This is a helmet sight system developed by Honeywell. It is envisioned that it would be useful in the deployment of Shuttle payloads. Basically it consists of two features: 1) a CRT assembly and associated optics which projects the image from the CRT face onto the helmet visor, and 2) a sensor electronics assembly with head-position sensors, used in conjunction with a helmet sight system to measure the direction of the user's line of sight as defined by the center of the helmet display's field of view. Feasibility and pilot acceptability of the visor display can be established via simulation. Information on the head position sensors was obscure, hence it was difficult to judge the merit of this capability. However, present applications are mostly related to providing pilot target tracking aid in high performance aircraft by having the vehicle or seeker antenna slaved to the pilot's line of sight. Application of high-performance response for Shuttle is unwarranted.

31. Estimation Techniques for Data Smoothing - Rejected because actual Orbiter flight is not required to prove feasibility.

This is the application of blending/complimentary filters for data reconstruction post flight. The experiment would have merit were the Shuttle data acquisition system not so extensive. As it currently stands, this technique could be used for filling in for data that is sparse -- i.e., using rate data and integrating to get attitude data between one second updates. Cost in this application would be low.

32. Terminal Area Sensing - Rejected because Orbiter is not a logical vehicle to test any type of terminal sensor because of the limited time per flight.

The suggestor envisioned some type of advanced onboard radar for this experiment. Infrared sensing was also mentioned in the suggestion. Lockheed has also attempted, and then discarded, this technique during the development of the L1011.

35. Wind Estimation - Rejected because Orbiter is not a logical vehicle to test a scheme to aid terminal approach guidance.

Work being done at ARC by Dr. George Meyer to obtain wind estimate based on non-linear equations of motion. Use knowledge to reduce trim error on final approach.